Report No. NA-69-420

Sabreliner Maintenance Manual

Model NA 265-70
Model NA 265-80

ORIGINAL ISSUE: 1 MAR 1970
REVISED: 14 MARCH 2003 — REVISION NO. 41

SABRELINER CORPORATION

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CRITICAL PART DISCLAIMER

CRITICAL PARTS LIST — A

WARNING

Installation of this part must be accomplished utilizing the highest quality workmanship, with particular attention being directed to alignment and fit. Proper installation of this part is critical to the continued airworthiness of the aircraft. Experience has shown tooling and installation drawings are typically required during the installation process. Therefore, Sabreliner Corporation accepts no responsibility for any performance or structural degradation or any other ensuing consequences which may occur when part installation is accomplished by anyone other than Sabreliner Corporation.

I have read and understand the preceeding disclaimer and hereby release Sabreliner Corporation of any responsibility or consequences resulting from the installation of this part in my aircraft.

Signed: _____

Date: _____

Title: _____

Aircraft Serial Number: _____

Company: _____

Part Number: _____

Invoice Number: _____

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RECORD OF REVISIONS

MFG REV NO	DESCRIPTION	ISSUE DATE	ATP REV DATE	INSERTED BY
39	See Highlights	7/31/00	12/21/00	ATP/VDR
40	See Highlights	6/14/02	10/24/02	ATP/VP
41	See Highlights	3/14/03	5/8/03	ATP/JB

REVISION LOG

Manual Control No:_____ Assigned To: _____

Manual Located At: _____

REV. NO.	Basic/ Rev. Date	Insertion Date	Inserted By (Initial)	Remarks
0	Mar 01/70			
1	Aug 6/71			
2	Jul 6/73			
3	May 1/74			
4	Jun 28/74			
5	Oct 31/74			
6	May 30/75			
7	Jul 21/75			
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9	Jun 30/76			
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25	Jan 10/86			
26	Aug 1/86			

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REVISION LOG

Manual Control No:_____ Assigned To: _____

Manual Located At: _____

REV. NO.	Basic/ Rev. Date	Insertion Date	Inserted By (Initial)	Remarks
27	Oct 31/86			
28	May 29/87			
29	Jan 15/88			Chapter 5 Reissue
30	Sep 2/88			General Revision
31	Jun 12/89			General Revision
32	Jan 19/90			General Revision
33	Jul 1/91			General Revision
34	Oct 4/91			General Revision
35	Apr 4/92			General Revision
36	Oct 5/92			General Revision
37	Nov 12/93			General Revision
38	Feb 29/96			General Revision
39	Jul 31/00			Chapter 5 Revision
40	Jun 14/02			Chapter 5 Revision
41	Mar 14/03	5/8/2003	JB	Chapter 5 Revision

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RECORD OF TEMPORARY REVISIONS

TEMP REV NO	DESCRIPTION	ISSUE DATE	ATP REV DATE	INSERT BY	DATE REMOVED	REV INCOR	REMOVE BY
	Service Publication List:1, 2, 5-10-04;Pgs 7, 8, 11, 12	7/25/01	9/13/01	ATP/MG	10/24/02	40	ATP/VP
	5-00-00:1/2, 3, 4	9/20/02	10/24/02	ATP/VP	5/8/03	41	ATP/JB

TEMPORARY REVISION LOG

Manual Control No: _____ **Assigned To:** _____

Manual Located At: _____

Temporary Rev. No.	Temporary Rev. Date	Insertion Date	Inserted By (Initial)	Remarks
5-10-04	7/25/01	9/13/01	ATP/MG	superceded by Rev No. 40
5-00-00	9/20/02	10/24/02	ATP/VP	superceded by Rev No. 41

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IMPORTANT

READ THIS NOTICE ! ! !

REVISION NOTICE

14 MARCH 2003

SABRELINER MAINTENANCE MANUAL

MODEL NA 265-70
MODEL NA 265-80

REVISION INSTRUCTIONS

This is the latest revision dated 14 March 2003, Revision No. 41 to your Sabreliner Maintenance Manual, Report No. NA-69-420. Insert latest revised pages, remove and destroy superseded pages.

This is a general revision to the manual, affecting the following chapters: 5. Significant to this revision are the following:

1. Requirement change on Sundstrand APU.
2. Incorporation of all Temporary Revisions.

When revising your manual the List of Effective Pages appearing in the front of each section or chapter should be used as a cross reference for removing obsolete pages and inserting the new and/or revised material. This ensures that only the latest material remains in the manual.

Please note that your maintenance manual has been revised in accordance with ATA specifications as closely as practical; this includes the page number/identification system. Revised pages are arranged and paginated to identify the chapter, section, and subject using the ATA 6-digit number system. To illustrate: 32-10-0 is 32-10-00, 5-20-1 is 5-20-01, 53-00-2 is 52-00-02. This number system is also used on revised L.E.P. and T.O.C. pages.

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SERVICE PUBLICATION LIST

The following is a list of Service Publications that are incorporated in this manual. They consist of Service Bulletins (S/B), Service Letters (S/L), and Optional Kits (S.O.K.).

Listed below are the Sabreliner Service Bulletins (S/B) incorporated in this manual.

SERVICE BULLETINS	SUBJECT
1	Inspection of Cockpit Enclosure — Center Beams, LH and RH.
11	Electrical — DC Power Circuit Breaker Rating Upgrade and Cooling Provision.
14	Thrust Reverser (Aeronca) — Inspection of 232-20258-505, -506, -507, and -508 Lever Arms.
17	Rudder — Static Balance Check.
29	Electrical Wiring Change to Landing Gear Downlock Relays K1069 and K1070 (Thrust Reverser Interlocks).
40	Electrical — Modification of Emergency Windshield Heat System.
41	Thrust Reverser — Improved Follow-Up Cable Mechanism and Improved Bearing Shoes on Pinion Gear Housing.
48	Elevator — Inspection for Trailing Edge Delamination.
49	Fowler Flaps — Replacement of Flap Flex Drive Shafts.
71-1	Wing — Inspection for Corrosion in Fuel Drains in Lower Wing Skin.
71-5	Installation of Auxiliary Power Unit (Solar T-62-T39 Gas Turbine) — NA 265-60, NA 265-70 and NA 265-80 Series.
72-6	Landing Gear — Installation of MLG Strut Door Position Sensing Switch.
72-8	Landing Gear — Removal of Main Landing Gear Inflatable Wheel Seal System.
73-7	Wiring Change — Main Landing Gear Load Switches.
74-1	FAA Airways Inspection/Checker Aircraft.
74-6	Flight Controls — Installation of Trim Disconnect.
74-7	Flight Control — Installation of Audio Signal for Horizontal Stabilizer Trim.
74-8	Thrust Reverser — Installation of Automatic Stow System — Sabreliner Models: NA 265-40, NA 265-60 & NA 265-70.
74-9	Electrical — Addition of Diodes to Protect Audible Warning Signal Generator.
74-10	Thrust Reverser — Installation of Improved Bogie, P/N 140-5484-1.
74-15	Thrust Reverser (Aeronca) — Rework of Actuator Link Lever Arms.
74-19	Thrust Reverser (Aeronca) — Inspection for Clearance of Blocker Doors.
74-20	Nose Landing — Uplock Stop Ice Breaker.
74-22	Electrical — Battery Temperature Indicating System Mod.
74-23	Fuel System — Improved Draining Provision.
74-27	Main Entrance Door — Replacement of Counterbalance Bungees.
75-1	Electrical — Installation of Rectifier in Main Landing Gear Control System.
75-2	Engine (G.E. — CF-700) — Installation of Fuel Recovery System.

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SERVICE PUBLICATION LIST (Continued)

Listed below are the Sabreliner Service Bulletins (S/B) incorporated in this manual.

SERVICE BULLETINS	SUBJECT
75-7	Main Entrance Door — Modification of Centerline Link Cover Door.
75-10	Thrust Reverser (Aeronca) — Inspection of Engine Aft Fairing Installation.
75-14	Thrust Reverser (Aeronca) — Replacement of Existing P/N 232-20258-501 or -503, and -502 or -504 Lever Arms for Blocker Doors.
75-17	Thrust Reverser (Aeronca) — Installation of Automatic Stow System — Sabreliner Model: NA-265-80.
75-20	Electrical — Installation of "Essential Bus Circuit Breaker Out" Caution Light and Test Circuit.
75-22	Flight Control — Modification of Trim System.
75-29	Electrical — Landing Gear Warning Horn Delay on Take-Off.
83-7	Solar Service Bulletin No. T-62 T-49-22 "Periodic Inspection of Fuel Tube Assembly."
85-2	Landing Gear — Nose Wheel Steering Test System.
86-1	Fuselage Entrance Door — Addition of Inspection Panels to Allow Corrosion Inspection of Internal Door Structure.
86-6	Landing Gear — Inspection of Nose Landing Gear Sleeve Assembly, Barrel Assembly, and Related Components.
87-3	Fuselage Structure — Modification/Inspection of Under Door Fuselage Structure Sta. 150 to 166.25.
87-5	Escape Hatch — Inboard Skin Modification and Inspection of Sealed Areas.
88-7	Fuselage Structure — Inspection of Engine Support Beam Interior.
92-4	Fuselage Structure — Inspection of Upper Main Entrance Door Stop Flange, P/N 370-311286-5.
93-1	Landing Gear — Inspection of Nose Landing Gear Axle and Outer Wheel Bearing Seal.
94-1	Fuselage Structure — Inspection Below Luggage, Galley and Forward Lavatory/Storage Area FS 143 to 206 RH Side and FS 174 to 206 LH Side, WL -21 to WL -29.

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SERVICE PUBLICATION LIST (Continued)

Listed below are the Sabreliner Service Letters (S/L) incorporated in this manual.

SERVICE LETTERS	SUBJECT
7	Fuselage — Deletion of Pressurized Nose Compartment.
10	Deletion of Fuselage Left Hand Inverter Ram Air Duct.
19	Nose Landing Gear — Improvement of Steering Cylinder Attachment to Steering Bellcrank.
30	Electrical/Hydraulic — New Nose Gear Steering System Control Box (P/N 57730) Interchangeability, and New Manifold (Valve) Assembly (P/N 57640) Interchangeability.
32	Hydraulic — Heavy Duty Hydraulic Motor – Pump Unit.
37	Thrust Reverser — Aeronca Inc. Service Bulletin No. S75-78-30-010C, dated 12 January 1977 "Blocker Door Actuator Link Lever Assemblies, Part Nos. 232-20258-503 -504, -505 -506, -507 -508, -509/-510 Assemblies.
39	Engine Pylon — Installation of Retaining Plate for Fire Detector Connector Receptacle.
49	Hydraulic System — Resetting of the Hydraulic Pump Motor Time Delay Relay (RDI TM 500).
60	Thrust Reverser (Aeronca) — Extension of Time Between Overhaul (TBO) Inspection, Cleaning and Repair.
64	Electrical System — Modification of the Secondary Bus Relay (K568) Electrical Wiring.

NEW

SERVICE PUB. LIST

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SERVICE PUBLICATION LIST (Continued)

Listed below are the Sabreliner Optional Kits (SOK) incorporated in this manual.

OPTIONAL KITS	SUBJECT
5	Environmental — Inflatable Door Seal System Modification.
7	Auxiliary Power Unit (APU) Addition of Check Valve in Pressure Sensing Line.
9	Engine — Installation of Stall Warning Horn Inhibitor Switch Bracket Assy. on Aircraft Equipped with G.E. CF700-2D-2 Engines
10	Fuselage — Installation of Dorsal Fin Ram Air Bird Stopper.
11	Hydraulic — Auxiliary Power Assist for Emergency Landing Gear Protection.
17	Electrical — Parallel/Series Battery Engine Start System Mod.
18	Power Plant — Teleflex Power Lever Control Cable Installation, RH Engine.
19	Engine — Addition of Engine Mounting Vibration Isolation System (CF 700-2D-2 Engine).
24	Landing Gear — 45 Degrees Nose Steering.
25	Electrical — Decrease Electrical System Affect of Hydraulic Pump Starting Transient.
26	Fuselage — Installation of Flotation Kits — FAA Airways Inspection Aircraft.
27	Electrical — Audio Pitch Trim Time Delay Relay Installation.
34	Aft Fuselage — Canted Bulkhead (F.S. 457) Mod.

NEW

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FRONTAL

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FRONTAL-L.E.P.	1/2	*	Mar 14/03
CH.-LIST.	1/2	*	Mar 14/03
SERVICE PUBLICATION LIST	1		Jun 14/02
	2		Jun 14/02
	3		Nov 23/84
	4		Nov 23/84

* The asterisk indicates pages revised, added, or deleted by the current revision.

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CHAPTER LISTING

Chapter/ No.	Title	Date
-*	Frontal Matter	Mar 14/03
-	Introduction	Jan 19/90
5*	Time Limits/ Maintenance Checks	Mar 14/03
6	Dimensions and Areas	May 15/81
7	Lifting and Shoring	Oct 15/82
8	Leveling and Weighing	Jul 6/73
9	Towing and Taxiing	Aug 1/77
10	Parking and Mooring	Jul 6/73
11	Required Placards	May 15/81
12	Servicing	Oct 5/92
20	Standard Airframe Practices	Oct 31/86
21	Air Conditioning	Feb 29/96
1 22	Autopilot (70)	Oct 31/86
2 22	Autopilot (80)	Oct 31/86
23	Communications	Nov 12/93
24	Electrical Power	Nov 12/93
25	Equipment/Furnishings	Sep 20/85
26	Fire Protection	Oct 5/92
27	Flight Controls	Nov 12/93
28	Fuel System	Feb 29/96
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1 71	Power Plant (70)	Feb 29/96
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1 80	Starting (70)	Jul 7/78
2 80	Starting (80)	Aug 1/86

1 Airplanes S/N 370-2 thru -6, -8 and -9

2 Airplanes S/N 380-1 and Subs

* The asterisk indicates chapters revised, added, or deleted by the current revision.

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INTRODUCTION

LIST OF EFFECTIVE PAGES

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INTRO-L.E.P.	1/2*	Jan 19/90
INTRO-CONTENTS	1/2	Oct 31/86
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	3/4*	Jan 19/90

* The asterisk indicates pages revised, added, or deleted by the current revision.

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INTRODUCTION

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General	1
Format	1
Identification of Subject Matter	1
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Warnings, Cautions and Notes	2

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INTRODUCTION

GENERAL

The function of this maintenance manual is to acquaint maintenance personnel with the systems and components of the SABRELINER and to direct them in the proper procedures for maintaining the airplane in an airworthy condition. Only the installations made in the airplane during manufacture have been reflected in this manual. Changes made after delivery from the manufacturer to the completion center or operator may be referenced in the Service Publication List, at the front of this manual. Operators may find it necessary to reference the logbooks of each airplane to verify changes made by service publications not included in this manual.The ability of maintenance personnel is recognized, and those procedures which are considered common to all aircraft have been briefly referenced or omitted.

This manual provides coverage for the NA 265-70 airplanes and the NA 265-80 airplanes. Maintenance procedures may be considered typical for both groups of airplanes except as noted. This manual is to be used in conjunction with the Wiring Diagram Manual, Report No. NA-69-421 or SR-73-26.

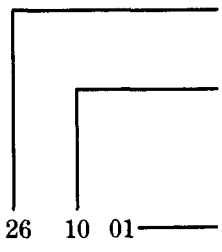
FORMAT

The manual has been prepared in accordance with Air Transport Association (ATA) Specification No. 100. A functional breakdown is employed whereby all data pertaining to a given system or component may be found in one chapter with a minimum of cross-referencing to other chapters.

The ELECTRICAL POWER and HYDRAULIC POWER chapters cover only the power sources and distribution equipment for these systems. Details of individual branch hydraulic or electrical systems will be found in the applicable chapters.

IDENTIFICATION OF SUBJECT MATTER

A three-dash number system is employed to identify subject matter. The first dash number identifies the chapter, the second dash number the section of the chapter, and the third dash number the subject of the section. The following example illustrates how the numbering system is used in the FIRE PROTECTION chapter:


26 10 01
Identifies the FIRE PROTECTION chapter
Identifies that section (group of related subjects) which provides coverage for the "Fire Detection System" portion of the FIRE PROTECTION chapter.
Identifies a specific subject of the "Fire Detection System"

The chapter number followed by a "-00" segregates that material covering the complete system. The chapter-section number followed by a "-00" is used for further details covering the subsystem.

PAGE NUMBER IDENTIFICATION

Page number blocks are assigned to separate the subject matter into the following categories:

General Coverage and Unit Description	1 - 100
Trouble Shooting	101 - 200
Maintenance Practices:	
If all subtopics are brief, they are combined into one topic. All such combined topics are page-numbered	201 - 300
If individual subtopics are so lengthy that a combination would require several pages, each subtopic is treated individually. Individual subtopic arrangement is	
Servicing	301 - 400
Removal/Installation	401 - 500
Adjustment/Test	501 - 600
Inspection/Check	601 - 700
Cleaning/Painting	701 - 800
Approved Repairs	801 - 900

Each new subject starts with page 1, 101, 201, etc., and continues through the page number block assignment as necessary.

FIGURE IDENTIFICATION

Figures (illustrations) are numbered consecutively within each topic (subject) as follows:

Figures in Description	1, 2, 3, 4, 5, etc.
Figures in Trouble Shooting	101, 102, 103, etc.
Figures in Maintenance Practices	
When not subdivided	201, 202, 203, etc.
When subdivided	
Servicing	301, 302, 303, etc.
Removal/Installation	401, 402, 403, etc.
Adjustment/Test	501, 502, 503, etc.
Inspection/Check	601, 602, 603, etc.
Cleaning/Painting	701, 702, 703, etc.
Approved Repairs	801, 802, 803, etc.

TABLE OF CONTENTS

The manual contains a general table of contents listing all the chapters in the manual; in addition, each chapter is prefaced with a table of contents identifying the subject matter within the chapter in the order of presentation. The table of contents is arranged with the following headings: DESCRIPTION, TROUBLE SHOOTING, and MAINTENANCE PRACTICES.

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INTRODUCTION (Continued)

LIST OF EFFECTIVE PAGES

A list of effective pages is provided with each chapter. It is an inventory of the pages in each chapter with the latest revision date for each page.

PART NUMBERS

This manual must not be used for identifying spare parts by number. Consult the SABRELINER Illustrated Parts Catalog for this information. Part numbers are used in this manual only as a means of identification when the nomenclature alone is inadequate.

REVISIONS

The LIST OF EFFECTIVE PAGES with each chapter serves as a tool for maintaining the manual. The LIST OF EFFECTIVE PAGES will be revised whenever the material in the affected chapter is revised. When revision pages are received, the list should be used as a cross reference for inserting the new/revised material and deleting obsolete material. This ensures that all obsolete material is destroyed and the manual is kept current.

Revisions to text are indicated by change bars (vertical lines) in the margin adjacent to the revised material. Revisions to illustrations may be indicated by change bars, zip-a-tone shading ▒▒▒ on electrical schematics, and pointing fingers ☞ on mechanical, hydraulic, and pneumatic diagrams.

When more than 50 percent of a page is changed, a major change symbol (MAJOR CHANGE) is placed in the lower outside corner of the page 1/4" above the top line of pagination.

When a new page is being added to an existing manual or section that has all new information included, a new change symbol (NEW) is used.

A REVISION LOG page is provided for recording revisions by revision date and the dates they were inserted in the manual.

A TEMPORARY REVISION LOG page is provided for recording Temporary Revisions (yellow sheets) by revision number, issue date, and the dates they were inserted in the manual.

WARNINGS, CAUTIONS, AND NOTES

Warnings, cautions and notes are used to set off instructions that require other than ordinary precautions. Definition and examples of each of the three types of precautionary notices follow:

1. Note — An essential clarification of a following operating procedure or condition.

   EXAMPLE:

   NOTE

   One complete turn of the adjustment screw will raise or lower the pressure approximately 1 psi.

2. Caution — Operating procedures or practices that if not strictly observed will result in damage to the equipment.

   EXAMPLE:

   CAUTION

   Do not operate engines with cowling or cowl panels removed.

3. Warning — Operating procedures or conditions that if not strictly observed will result in personal injury or loss of life.

   EXAMPLE:

   WARNING

   Ingition switch must be in the OFF position when rotating the propellers by hand.

NON-SABRELINER APPROVED PARTS

WARNING

Only Sabreliner parts or Sabreliner approved parts obtained from approved sources are to be used in connection with the maintenance and repair of Sabreliner airplanes.

Genuine Sabreliner parts are manufactured and inspected under rigorous procedures in accordance with a Production Certificate (PC) to insure airworthiness and suitability for use in Sabreliner aircraft application. Parts purchased from sources other than Sabreliner, or its approved vendors, may not have the required tests and inspections performed and may be different in fabrication techniques and materials. A non-approved part may seriously compromise the safety of the aircraft.

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INTRODUCTION (Continued)

WARNING (Cont'd)

Government surplus parts, salvaged airplane parts, reworked parts obtained from non-Sabreliner approved sources, or parts, components or structural assemblies which do not have proper documentation may have a service history which is unknown or cannot be authenticated. These parts may have been subjected to unacceptable stresses or temperatures or have hidden damage which is not discernible through routine visual or usual non-destructive testing techniques. This may render the part, component or structural assembly unsuitable and unsafe for aircraft use, even if such part, component or structural assembly were originally manufactured by Sabreliner or an approved source.

Sabreliner Corporation expressly disclaims any responsibility for malfunctions, failures, damage or injury caused by the use of non-Sabreliner approved parts.

NOTE

Sabreliner Corporation expressly reserves the right to supersede, cancel and/or declare obsolete, without prior notice, any part, part number, kit or publication referenced in this manual.

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CHAPTER

05

TIME LIMITS/ MAINTENANCE CHECKS

CHAPTER 5

TIME LIMITS/MAINTENANCE CHECKS

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Starter/Generator Replacement	5-50-00	1
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CHAPTER 5

TIME LIMITS/MAINTENANCE CHECKS

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* The asterisk indicates pages revised, added, or deleted by the current revision.

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CHAPTER 5

TIME LIMITS/MAINTENANCE CHECKS

GENERAL

This chapter provides recommended time limits for servicing, adjustment/test and inspection/check; also, this chapter establishes mandatory replacement schedules for Sabreliner NA 265-70 and NA 265-80 airplanes. These items are refined and changed during the service life of the airplane by continually evaluating the performance of the equipment, results of scheduled maintenance checks and a thorough study of factual operating data. The interval between the accomplishment of the recommendation is intended to be the longest period of time that an item or component can safely operate without an inspection or maintenance check. No attempt is made in this chapter to sequence the order of accomplishment. The tables in 5-10-01 are intended as an aid for referencing scheduled requirements.

Local conditions (special utilization, geographical locations, etc.) may dictate **more frequent inspection** or replacement, or more thorough inspections. Therefore, the operators are expected to exercise their prerogative to increase the frequency or scope of any recommendation. It is recommended that hourly intervals up to and including 200 hours cannot be exceeded by more than 10% of the hourly limitation and by not more than 5% on hourly limited items over 200 hours.

It is recommended that the calendar time limitations not be exceeded. Service life limited parts listed in the F.A.A. Type Certificate Data Sheet No. A2WE must be retired in accordance with the prescribed schedule.

The replacement schedule recommends replacement of items at a specific time whose failure would compromise safety of flight beyond reasonable limits or definitely cause a flight to abort. Items not listed will be known as "on condition" items and will be replaced only when necessary.

Operators not utilizing the Service Maintenance Inspection Program (SMIP) must maintain maintenance records to accurately control the length of service life on all major airplane components (equipment and structure) to ensure that the retirement of components is accomplished within the required time period. Such records will make note of the complete service-life hours of any component which could be interchanged from one airplane to another, and thus possibly reduce the over-all life of an airplane.

NOTES

The following effectivities are used in chapter 5 to explain conditions:

- **A** See Special Inspection, 5-10-07.
- **B** APU Hours.
- **C** See Unscheduled Maintenance Checks, 5-50-00.
- **D** 12 month initial inspection from the issue date of TR 90-2 (dated June 22/90); see 36, 48 or 60 month intervals for recurring inspection requirements.
- **E** Interval may be extended to six (6) months with the use of PRIST™ (or equivalent) additive a minimum of ten (10) consecutive days each month. Fuel must be treated or inspected for microbe contamination.
- **F** Except airplanes with teflon bearings installed, P/N KNDB10-22 or equivalent.
- **G** Repair may require Aux Fuel Cell removal.
- **H** Providing P&W S/B 2005 is incorporated not later than 2,200 hours.
- **I** See Illustrated Parts Catalog (NA-69-424) or Sabreliner Corporation for any alternate part number hose assembly that may exist in this location.
- **J** Hose assemblies are part of fuel drain tank kit installed by STC number SA1162CE.
- **K** Certify in accordance with the applicable F.A.R.
- **L** Repair may require F.S. 333 interior removal.
- **M** See Special Inspection, 5-10-07, chapter 5 — Corrosion Control/Inspection.
- **N** Hydraulic Pump/Motor Assembly, P/N 265-580623, overhaul is "On Condition."
- **P** New or remanufactured Hose Assy.

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TIME LIMITS

SCHEDULED MAINTENANCE CHECKS AND INSPECTIONS

The following tables are intended to be used as a reference/guide for recommended frequency of an area of scheduled maintenance. For example, if an airplane has 150 hours of time in service, the servicing and inspection/check areas each have at least one applicable item at the 150 hour frequency. Preflight, Post Flight and Special Inspections are not included in the following tables. See 5-20-01 or 5-10-07 for these items.

Table 1. Hour Frequency

Hour Frequency	Servicing	Adjustment/ Test	Inspection/ Check	Overhaul	Retirement
50		X	X		
100	X		X		
150	X	X	X		
200			X		
250	X		X		X
300	X	X	X		X
400	X		X		X
500			X		
600	X	X	X		X
900			X		X
1000		X	X		
1100	X		X		
1200	X	X	X	X	X
1500			X	X	X
2000				X	X
2100			X		X
2200			X	X	X
2400	X		X		
3000			X	X	X
4000				X	
10,000*					X
30,000*					X

*Also applicable to flights.

Table 2. Calendar Frequency

Calendar Frequency	Servicing	Adjustment/ Test	Inspection/ Check	Overhaul	Retirement
1 Month	X		X		
3 Months			X		
6 Months			X	X	
12 Months	X	X	X	X	
24 Months	X	X	X	X	
36 Months			X		
48 Months			X		
60 Months			X		X
84 Months			X	X	
15 Years*					X
24 Years*					X

* Also applicable at each 4380 Bottle Pressurizations.

Table 3. Landing Frequency

Landing Frequency	Servicing	Adjustment/ Test	Inspection/ Check	Overhaul	Retirement
150*	X				
1000*			X		
2400			X		
24,050					X
26,000					X
30,000					X

* Thrust reverser cycles

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TIME LIMITS

SERVICING

FREQUENCY	ITEM	EFFECTIVITY	REFERENCE/ CHAPTER
100 Hours	Change Oil — Three Wheel Cooling Turbine Assembly, AiResearch P/N 572365-1, -2, and -3	370-1 and -7 380-1 and Subs	21
150 Hours or 150 Cycles, whichever occurs first	Lubricate — Aeronca Thrust Reversers	370-1 amd -7; 380-1 and Subs	78
150 Hours	Lubricate — MLG and Doors	All	12
	Lubricate — NLG and Door Linkages	All	12
	Lubricate — Brake Control Valve	All	12
	Lubricate — Speed Brake	All	12
	Lubricate — Wing Flap Actuators	All	12
	Lubricate — Aileron Trim Tab	All	12
	Lubricate — Rohr Thrust Reversers	370-2 thru -6, -8 and -9	78
150 Hours or 12 Months, whichever occurs first	Lubricate — Main Entrance Door, Ground Escape Hatch, and Access Doors	All	12
300 Hours	Lubricate — Elevator Pushrods	All	12
	Lubricate — NLG Torque Arm Pin Bearings	All with grease fitting	12
	Lubricate — RH and LH MLG Downlocks	All	12
	Lubricate — Horizontal Stabilizer	All	12
	Lubricate — Pilot's Sliding Window	All	12
	Lubricate — Aeronca Thrust Reverser Lever Spring Plunger Assembly	370-1 and -7 380-1 and Subs	12
	Application of ICEX II to Aileron Cable Sectors A	All	27
300 Hours or 12 Months, whichever occurs first	Change — Engine Oil, P & W JT12A-8	370-2 thru -6, -8 and -9	79

A See Special Inspection, 5-10-07.

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SERVICING (Continued)

FREQUENCY	ITEM	EFFECTIVITY	REFERENCE/ CHAPTER
600 Hours	Lubricate — MLG Sidebrace Idler Link Springs	All	12
	Lubricate — NLG Uplock Hook Needle Bearings	All	12
	Lubricate — NLG Torque Arm Pin Bearings	All without grease fitting	12
	Lubricate — Engine Power Lever Cross Shaft, P & W JT12A-8	370-2 thru -6, -8 and -9	12
	Service Oil — Two Wheel Cooling Turbine Assembly, AiResearch P/N 203400-2	370-2 thru -6, -8 and -9	21
	Lubricate — Starter/Generator Shaft Splines, Bendix P/N 30B58-3B	370-2 thru -6, -8 and -9	24
	Change — Engine Oil — G.E. CF700	370-1 and -7 380-1 and Subs	79
1100 Hours	Lubricate — Rohr Thrust Reverser Bearings, P/N 140-5410-1 or -3	370-2 thru -6 -8 and -9	78
1200 Hours	Lubricate — Flight Control Hinges	All	12
1500 Hours	Lubricate — Aileron Sector Pushrods	All	12
2400 Hours	Lubricate — Flight Control Bearings	All	12
1 Month E	Treat — Fuel System with Biocide Agent	All	28
12 Months	Change — APU Oil A	All changed by S/B 71-5 (Inst of APU)	49
24 Months	Lubricate — Wing to Fuselage Fwd Attach Bolts	380-1 thru -4 F and 370-1 thru -9 not changed by SOK 81-3 (Service Life Extension)	57

A See Special Inspection, 5-10-07.

E Interval may be extended to six (6) months with the use of PRIST™ (or equivalent) additive a minimum of ten (10) consecutive days each month. Fuel must either be treated or inspected for microbe contamination.

F Except airplanes with teflon bearings installed, P/N KNDB10-22 or equivalent.

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TIME LIMITS

ADJUSTMENT/TEST

FREQUENCY	ITEM	EFFECTIVITY	REFERENCE/ CHAPTER
50 Hours	Functional Test — Nose Wheel Steering System	All not changed by S/B 85-2 (Nose Wheel Steering Test Sys)	32
150 Hours	Functional Test — Nose Wheel Steering System	All changed by S/B 85-2 (Nose Wheel Steering Test Sys)	32
300 Hours or after replacement	DELETED		
300 Hours	Operational Test — AC Power Generation System	All	24
	Resistance Test — Engine Nacelle Fire Detector Elements	All	26
	Operational Test — Battery Temperature System A	380-1 and Subs. 370-1 thru -9 with automatic control system installed	24
	Functional Test — Essential Bus 28 VDC Circuit breakers (Ref. AD 75-08-11)	370-1 thru -9; airplanes not changed by S/B 11 (DC Pwr CB Rating Upgrade with Cooling Provisions). S/B 75-20 (Inst of Essen Bus CB Out Caution Light and Test Circuit).	24
	Operational Test — Fuel Heater System	370-1 and -7; 380-1 and Subs	28
	Operational Test — Windshield Anti-icing System	All	30
	Operational Test — Environmental System Ram Air Inlet Electrical Heating System	All	30
	Functional Test — Anti-skid Fail Safe System	370-2 thru -6, -8, and -9	32
	Functional Test — Landing Gear System	All	32
	Functional Test — Anti-skid Brake System	All	

A See Special Inspection, 5-10-07.

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ADJUSTMENT/TEST (Continued)

FREQUENCY	ITEM	EFFECTIVITY	REFERENCE/ CHAPTER
300 Hours (Cont'd)	Operational Test — Stall Warning Vane Heaters	All	34
	Operational Check — Thrust Reverser Control System	All	78
300 Hours or 12 Months, whichever occurs first	Functional Test — Nose Wheel Steering Feedback Potentiometer Redundant Cable System	All	32
600 Hours	Functional Test — Fire Extinguisher System Electrical Circuit	All	26
	Functional Test — Fire Detection System	All	26
	Operational Test — Fuel Jettison System	370-2 thru-6, -8 and -9	28
	Functional Test — Fuel Transfer Ejectors [A]	All	28
	Operational Test — Hydraulic Power System	All	29
	Resistance Test — APU Fire Detector Elements	All changed by S/B 71-5 (Inst of APU)	49
	Functional Test — APU Fire Detector System	All changed by S/B 71-5 (Inst of APU)	49
	Functional Test — APU Fire Extinguishing System Electrical Circuit	All changed by S/B 71-5 (Inst of APU)	49
1000 Hours	Torque — Aeronca Thrust Reverser Translating Structure	370-1 and -7; 380-1 and Subs	78
1200 Hours	Operational Test — Airspeed Pitot and EPR Heater Systems	All	30
12 Months	Flight Test — Cabin Leakage	All	21
	Resistance Test — Static Dischargers [C]	All	23
	Flight Test — Landing Gear Position and Warning System	All	32
	Flight Test — Crew and Passenger Oxygen System	All	35
24 Months	Calibration/Certification — Altimeters	All	34 [K]
	Calibration/Certification — ATC Transponder(s),	All	— [K]
	Calibration/Certification — Air Data Computer	When ADC Sys is Installed	— [K]
	Test — Pitot and Static System Leakage	All	34 [K]
	Calibration/Certification — Altimeters (Standby)	All as equipped	34

[A] See Special Inspection, 5-10-07.
[C] See Unscheduled Maintenance Checks, 5-50-00.
[K] Certify in accordance with the applicable FAR.

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TIME LIMITS

INSPECTION/CHECK

FREQUENCY	ITEM	EFFECTIVITY	REFERENCE/ CHAPTER
50 Hours [B]	DELETED		
50 Hours	Inspect — Aeronca Thrust Reverser — Blocker Door Lever Arms, P/N 232-20258-505, -506, -507, -508, -509 and -510	370-1 and -7; 380-1 and Subs	78
100 Hours	Inspect — Entry Door Seal Sensing Lines Dehydrator	All	21
100 Hours or 3 Months, whichever occurs first	Inspection and Performance Check — Emergency Power Supply, JET PS-800 in accordance with JET Service Letter S/L-4 and Instruction Manual TP202 (4/1/74) or latest revision	All as equipped	34
	Inspection and Performance Check — Emergency Power Supply, JET PS-823 in accordance with JET Service Letter SL-8P (4/23/86) or latest revision	All as equipped	34
150 Hours	150 Hour Maintenance Check	All	5-20-01
	Inspect — Rohr Thrust Reverser	370-2 thru -6, -8 and -9	78
	Inspect and Clean — Engine Oil Strainer P&W JT12A-8	370-2 thru -6, -8 and -9 not changed by P&W S/B 2005	79
200 Hours	Inspect and Cycle — Airplane Batteries	All	24
300 Hours	Inspect — Air Temperature Sensor/Blower	All	21
	Inspect and Clean — Water Separator Condenser	All	21
	Inspect — DC Starter/Generator Brushes [A] Bendix P/N 30B58-3B	370-2 thru -6, -8 and -9	24
	Inspect — DC Starter/Generator Brushes [A] G.E. P/N 2CM82J4	370-1 and -7; 380-1 and Subs	24
	Inspect — Cockpit and Cabin Heater Pads [M]	All not changed by S/B 85-3, Part I (Mod of Floor Heater Pad)	25
	Inspect — Stabilizers	All	25

[A] See Special Inspection, 5-10-07.

[B] APU Hours

[M] See Special Inspection, 5-10-07, Chapter 5 — Corrosion Control/Inspection.

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INSPECTION/CHECK (Continued)

FREQUENCY	ITEM	EFFECTIVITY	REFERENCE/ CHAPTER
300 Hours (Cont'd)	Inspect — Elevator/Rudder Control Cable Tension M	All	27
	Inspect and Drain — Engine Fuel Heaters, G.E. CF700	370-1 and -7; 380-1 and Subs	28
	Inspect — NLG Steering Bellcrank	All	32
	Inspect — MLG Downlock Sidebrace Idler Rod End A	All with hollow or brass filled shank rod ends, and all others not marked per S/B 81-7 (MLG Downlock Sidebrace Idler Rod End Insp.)	32
	Inspect — Aft Fuselage Interior M	All	53
	Inspect — Lower Fuselage Interior (FS 264 to 357) M	All	53
	Inspect — Aft Fuselage Interior, Canted Bulkhead FS 457 to 485 M	All	53
	Inspect — Fuselage Nose Section Interior M	All	53
	Inspect — Clips, FS 76 M	All without heavy duty clips A	53
	Inspect — Pylon and Nacelle M	All	54
	Inspect — Power Plant G.E. CF700	370-1 and -7; 380-1 and Subs	71
	Inspect — Power Plant, P&W JT12A-8	370-2 thru -6, -8 and -9	71
	Inspect — Engine Mount Support Bolts A P&W JT12A-8	370-2 thru -6, -8 and -9	71
	Inspect and Clean — Fuel Control Screen, P&W JT12A-8	370-2 thru -6, -8 and -9	73
	Inspect and Clean — Fuel Control Filter, G.E. CF700	370-1 and -7; 380-1 and Subs	73
	Inspect and Clean — Main Fuel Supply Filter, G.E. CF700	370-1 and -7; 380-1 and Subs	73
	Inspect — Fuel Pressurization and Dump Valve Inlet Screen, P&W JT12A-8	370-2 thru -6, -8 and -9	73
	Inspect and Clean — Fuel Pump Filter, G.E. CF700	370-1 and -7; 380-1 and Subs	73
	Inspect — Engine Igniter Plugs, G.E. CF700	370-1 and -7; 380-1 and Subs	74
	Inspect — Engine Igniter Plugs, P&W JT12A-8	370-2 thru -6, -8 and -9	74

A See Special Inspection, 5-10-07.

M See Special Inspection, 5-10-07, Chapter 5 — Corrosion Control/Inspection.

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INSPECTION/CHECK (Continued)

FREQUENCY	ITEM	EFFECTIVITY	REFERENCE/ CHAPTER
300 Hours (Cont'd)	Inspect — Aeronca Thrust Reversers	370-1 and -7; 380-1 and Subs	78
	Inspect and Clean — Engine Oil Strainer P&W JT12A-8	370-2 thru -6, -8 and -9 H	79
	Inspect — Magnetic Drain Plug in Transfer and Accessory Gear Boxes, G.E. CF700	370-1 and -7; 380-1 and Subs	79
300 Hours or after replacement	Check — Regulation Setting of Fluid Pressure Regulating Valve, Three Wheel Cooling Turbine	370-1 and -7; 380-1 and Subs	21
500 Hours or 6 Months, whichever occurs first	Inspect — Elevator Honeycomb Trailing Edge A	All as installed	27
600 Hours	Inspect — DC Starter/Generator Shaft Splines, Bendix P/N 30B58-3B	370-2 thru -6, -8 and -9	24
	Inspect — Cockpit and Cabin Heater Pads A M	All changed by S/B 85-3, Part I (Mod. of Floor Heater Pads)	25
	Check — NLG Strut Steering Collar Clearance	All	32
	Inspect — NLG Axle P/N 4010745-3 (Terminated when -5 installed)	All	32
	Inspect — NLG Steering Link	All	32
	Inspect — MLG Actuating Cylinder Rod End and Clevis Pin	All	32
600 Hours or 12 Months, whichever occurs first	Inspect — Aileron Cables Without Nylon Jackets (AD71-25-04) P/N 276-523006-11, 276-523008-11 and 276-523005-11	380-1 thru -6; 370-1 thru -9 not changed by S/B 73-15 (Nylon Jacketed Aileron Cables)	27
600 Hours or 24 Months, whichever occurs first	DELETED		
900 Hours	Inspect — Cabin Air Dump and Safety Valve Area	All	21
	Inspect — Cabin Air Pressure (Outflow) Valve Area	All	21

A See Special Inspection, 5-10-07.

H *Providing P&W S/B 2005 is incorporated not later than 2,200 hours.*

M See Special Inspection, 5-10-07, Chapter 5 — Corrosion Control/Inspection.

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INSPECTION/CHECK (Continued)

FREQUENCY	ITEM	EFFECTIVITY	REFERENCE/ CHAPTER
900 Hours (Cont'd)	Inspect and Clean — Rohr Thrust Reverser Hydraulic Filter, P/N 24814	370-2 thru -6, -8 and -9 changed by SOK 82-11 (Addition of Hyd. Filter to System)	29
1000 Hours	Inspect — Lavatory Waste Receptacles (A.D. 74-08-09) M	All	25
	Inspect — Engine Hot Section, G.E. CF700	370-1 and -7; 380-1 and Subs	Refer to GE CF-700 Maint. Manual
	Inspect — Upper Main Entrance Door Stop Flange — S/B 92-4	All	53
	Torque — Thrust Reverser Translating Structure	370-1 and -7; 380-1 and Subs	78
	Inspect — Engine Hot Section, P & W JT12A-8 A	370-2 thru -6, -8 and -9	71
1000 Hours or 1000 Cycles, whichever occurs first	Inspect — Aeronca Thrust Reversers, A P/N 234-23000 (Includes inspection of flexible drive shafts, P/N 121460-2)	370-1 and -7; 380-1 and Subs	78
1100 Hours	Inspect — Rohr Thrust Reverser Bearings, P/N 140-5410-1 or -3	370-2 thru -6, -8 and -9	78
	Inspect — Rohr Thrust Reverser Rigging	370-2 thru -6, -8 and -9	78
	Inspect — Rohr Thrust Reverser Bogies, P/N 140-5441 or 140-5413-1	370-2 thru -6, -8 and -9 not changed by S/B 74-10 (Inst. of Improved Bogie P/N 140-5494-1)	78
1200 Hours	Inspect — Flight Control Hinge Pin	All	12
	Inspect — Elevator Universal Joints	All	27
	Inspect and Clean — Nose Steer Control Valve Filter Element, Sterer P/N 35582	All Changed by S/L 30 (New Nose Steer Manifold)	32
	Inspect — NLG Axle and Outer Bearing Seal (Ref. S/B 93-1) M	All	32
1500 Hours	Inspect — Aileron Cables with Nylon Jackets	380-7 and Subs. 370-1 thru -9 and 380-1 thru -6 changed by S/B 73-15 (Nylon Jacketed Aileron Cable)	27

A See Special Inspection, 5-10-07.
M See Special Inspection, 5-10-07, Chapter 5 — Corrosion Control/Inspection.

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INSPECTION/CHECK (Continued)

FREQUENCY	ITEM	EFFECTIVITY	REFERENCE/ CHAPTER
1500 Hours (Cont'd)	DELETED		
1500 Hours B or 3600 Cycles	Inspect/Clean APU PCD Filter (on APU's with Modulating Bleed Air Valve)	All with APU	49
	Inspect/Clean APU Fuel Pump Internal Filter	All with APU	49
2100 Hours and each 1200 Hours thereafter	Inspect — Wing Flap Actuator Jackshafts	All	27
2200 Hours	Inspect — Battery Ground Cables/Bolts	370-1 and -7; 380-1 and Subs	24
2400 Hours	Inspect — Flight Control Bearings	All	12
2400 Hours or 2400 Landings, whichever occurs first	Inspect — MLG Strut Outer Cylinder P/N 1127L001-3, (LH) P/N 1127L001-4, (RH) P/N 1127L001-5, (LH) P/N 1127L001-6, (RH)	All	32
3000 Hours or next engine removal, whichever occurs first	Inspect and Clean — DC Generator Shunt and Ground Jumper Leads, P&W JT12A-8	370-2 thru -6, -8 and -9	24
1 Month E	Inspect — Fuel System for Microbe Contamination	All	28
3 Months	Periodic Check — Emergency Power Supply, JET PS-835 in accordance with JET Service Letter SL-80 (6/1/84) Item 1.B. or latest revision	All as equipped	34
6 Months	Weight Check — Portable Fire Extinguisher Agent Container (CO_2) (Cockpit Installation Only) P/N 870904, 870905, and 870906 type 2TB, 4TB, or 5TB A	All as installed	—
	Inspect — APU External Check (Ref. Sundstrand SB-T-62T-49-77) (Tables I and II)	All changed by S/B 71-5 (Inst. of APU)	49
12 Months	Weight Check — Engine Fire Extinguisher Agent Containers A	All	26
	Weight Check — Portable H_2O Cabin Fire Extinguisher A	All	—
	Inspect — NLG Strut Piston	All	32

A See Special Inspection, 5-10-07.

B APU Hours.

E Interval may be extended to six (6) months with the use of PRIST™ (or equivalent) additive a minimum of ten (10) consecutive days each month. Fuel must be treated or inspected for microbe contamination.

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INSPECTION/CHECK (Continued)

FREQUENCY	ITEM	EFFECTIVITY	REFERENCE/ CHAPTER
12 Months (Cont'd)	Capacity Test — Emergency Power Supply, JET PS-835 in accordance with JET Service Letter SL-80 (6/1/84) Items 2 and 3, or latest revision	All	34
	Inspect — APU, General Annual Check A (Ref. Sundstrand SB-T-62T-49-77) (Tables III and IV)	All changed by S/B 71-5 (Inst. of APU)	49
	Inspect — Emergency Ground Escape Hatch M	All	52
	Inspect — Escape Hatch (Inactive) M	370-1 and Subs; 380-1 thru -68	53
	Inspect — Lower Flange and Skin F.S. 13 to 76, Lower Cap Angle of Nose Gear Beam B.P.9 and Lower Electronics Bay D M	All	53
	Inspect — Splice Between Lower Center Skin and Side Skin F.S. 99 to 156.25 LH and RH D M	All	53
	Inspect — Skin Splice on Intermediate Longerons, LH and RH. Inspect Longeron Above Main Entrance Door. Inspect LH and RH Lower Longeron D M	All	53
	Inspect — Skin Splice at F.S. 143 Lower Side Skin, LH and RH, Cockpit Glass Support Structure, Windshield Bow Supports, Intermediate Longerons and Skin Splice LH and RH, Cockpit Floors and Lower Longerons LH and RH Side D M	All	53
	Inspect — Lower Longeron Along Threshold of Main Entrance Door and Lower Door Stop D M	All	53
	Inspect — Speed Brake Inner Skin and Doubler, and Sonic Doublers — Ailerons and Flaps D M	All	53
	Inspect — Upper Longeron Splice F.S. 206 LH and RH D M	All	53
	Inspect — All Cabin Floors, Floor Support Beams, and Lower Main Intermediate Longerons on LH and RH Side D M	All	53
	Inspect — Floor Support Beam F.S. 264 and Wing Attach Fittings LH and RH on Fuselage D M	All	53
	Inspect — Strap Below Escape Hatches D M	All	53
	Inspect — Aileron Bellcrank Support and Pulley Support — Landing Gear Emergency Release D M	All	53
	Inspect — ADF Loop and ADF Sense Antenna Support Structure and Upper Main Intermediate Longerons D M	All	53

A See Special Inspection, 5-10-07.

D 12 month initial inspection from the issue date of TR 90-2 (dated Jun 22/90); see 36, 48, or 60 month intervals for recurring inspection requirements.

G Repair may require Aux Fuel Cell removal.

L Repair may require F.S. 333 interior removal.

M See Special Inspection, 5-10-07, Chapter 5 — Corrosion Control/Inspection.

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INSPECTION/CHECK (Continued)

FREQUENCY	ITEM	EFFECTIVITY	REFERENCE/ CHAPTER
12 Months (Cont'd)	Inspect — Horizontal Stiffeners on Fwd Side of F.S. 333 and Bulkhead to Side Skin Attach Angle D G M	All	53
	Inspect — Vertical Hat Sections on Aft Side of FS 333, Floor of Aux Fuel Bay, Pylon Castings and Upper Intermediate Longerons LH and RH FS 333 to 357 (Especially Near the Center of the Bulkhead) D L M	All	53
	Inspect — Web of Bulkhead F.S. 298.5 D M	All	53
	Inspect — Aft Aux Fuel Cell Floor Support Structure D G M	All	53
	Inspect — Intermediate to Aft Upper Longeron Splice F.S. 357 LH and RH D M	All	53
	Inspect — Pylon Match Angles and Fuselage Skin G M	370-2 thru -6, -8 and -9 not in compl. with TR 90-2 Corr. Insp. (see recurr. insp. at 60 mos.)	53
	Inspect — Radio Altimeter Antenna Interior and Exterior Doublers, and Fuselage Skin In Area D M	All not in compl. with TR 90-2 Corr. Insp. (see recurr insp. at 36 mos.)	53
	Inspect — Pylon Attach Fitting F.S. 333 D M	370-2 thru -6, -8 and -9	54
	Inspect — Vertical Stabilizer Rear Spar Cap D M	All	55
	Inspect — Wing Tips, Wing Front Spar, Leading Edge Skin, Rib and Stiffeners D M	All	57
24 Months	Inspect — NLG Sleeve Assembly, Barrel Assembly, and Related Components (Ref S/B 86-6) M	All	32
	Inspect — MLG Downlock Assembly A	All not changed by S/B 83-10, Part 2 (Insp. and Mod. MLG Downlock Assy.)	32
	Inspect and Test — Pitot/Static Systems	All	34
	Inspect — APU, Periodic Maintenance (Ref. Sundstrand SB-T-62T-49-77) (Table V)	All changed by S/B 71-5 (Inst. of APU)	49
	Inspect — Under Door Fuselage Structure F.S. 150 to 166.25 (Ref. S/B 87-3)	All	52
	Inspect — Lower Forward Fuselage Interior (F.S. 182 to 242) M	All	53
	Inspect — Engine Support Beam Interior (Ref. S/B 88-7) M	370-2 thru -6, -8 and -9	54

A See Special Inspection, 5-10-07.

D 12 month initial inspection from the issue date of TR 90-2 (dated Jun 22/90); see 36, 48, or 60 month intervals for recurring inspection requirements.

G Repair may require Aux Fuel Cell removal.

L Repair may require F.S. 333 interior removal.

M See Special Inspection, 5-10-07, Chapter 5 — Corrosion Control/Inspection.

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INSPECTION/CHECK (Continued)

FREQUENCY	ITEM	EFFECTIVITY	REFERENCE/ CHAPTER
24 Months (Cont'd)	Inspect — Cockpit Enclosure Center Support Beams	370-1 thru -9; 380-1 thru -19, -30 and -32	56
	Inspect — Wing Front and Rear Spars Per S/B 83-2 (AD 73-18-03) M	370-1 thru -9; 380-1 thru -72	57
	Inspect — Lower Wing Fuel Drain Valve Cavities	All	57
	Inspect — Wing Joint Rib Assembly M	All	57
	Inspect — Wing Flap Track Outboard Spar Lug (Ref. S/B 87-7, AD 71-03-03)	370-2 and Subs 380-1 and Subs	57
	Inspect — Main Entrance Door In Accordance with S/B 86-1	All	
36 Months	Inspect — Speed Brake Inner Skin and Doubler and Sonic Doublers — Ailerons and Flaps M	All not changed by S/L 98-2	27
	Inspect — MLG Downlock Support Forging and Downlock Assembly M	All	32
	Hydrostatic Check — Oxygen Bottle, Class DOT 3HT 1850 A	All	35
	Inspect — Lower Flange and Skin F.S. 13 to 76, Lower Cap Angle of Nose Gear Beam B.P. 9 and Lower Electronics Bay M	All not changed by S/L 98-2	53
	Inspect — Splice Between Lower Center Skin and Side Skin F.S. 99 to 156.25 LH and RH M	All not changed by S/L 98-2	53
	Inspect — Skin Splice on Intermediate Longeron, LH and RH Side. Inspect Longeron Above Main Entrance Door. Inspect Lower Longerons, LH and RH Side M	All not changed by S/L 98-2	53
	Inspect — Lower Longeron Along Threshold of Main Entrance Door and Lower Door Stop M	All not changed by S/L 98-2	53
	Inspect — Floor Support Beam at F.S. 264 and Wing Attach Fittings LH and RH on the Fuselage M	All not changed by S/L 98-2	53
	Inspect — Aft Aux Fuel Cell Floor Support Structure G M	All not changed by S/L 98-2	53
	Inspect — Radio Altimeter Antenna Interior and Exterior Doublers, and Fuselage Skin In Area M	All not changed by S/L 98-2	53
	Inspect — Intermediate to Aft Upper Longeron Splice F.S. 357 LH and RH M	All not changed by S/L 98-2	53

A See Special Inspection, 5-10-07.
G Repair may require Aux Fuel Cell removal.
M See Special Inspection, 5-10-07, Chapter 5 — Corrosion Control/Inspection.

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INSPECTION/CHECK (Continued)

FREQUENCY	ITEM	EFFECTIVITY	REFERENCE/ CHAPTER
36 Months (Cont'd)	Inspect — Fuselage Structure Below Luggage, Galley and Forward Lavatory/Storage Area. FS 143 to 206 RH Side and FS 174 to 206 LH Side, WL -21 to WL -29 (Ref: S/B 94-1) M	All	53
	Inspect — Pylon Attach Fitting F.S. 333 M	370-2 thru -6, -8 and -9 All not changed by S/L 98-2	54
	Inspect — Vertical Stabilizer Rear Spar Cap M	All not changed by S/L 98-2	55
	Inspect — Wing Tips, Wing Front Spar Leading Edge Skin, Rib and Stiffeners M	All not changed by S/L 98-2	57
48 Months	Inspect — Cockpit and Cabin Heater Pads A M	All changed by S/B 85-3, Part II (Heater Pad Mod.)	25
	Inspect — Skin Splice at F.S. 143 Lower Side Skin, LH and RH Side, Cockpit Glass Support Structures, Windshield Bow Supports, Intermediate Longerons and Skin Splice LH and RH, Cockpit Floors and Lower Longerons LH and RH Side. M	All not changed by S/L 98-2	53
	Inspect — Upper Longeron Splice F.S. 206 LH and RH M	All not changed by S/L 98-2	53
	Inspect — All Cabin Floors, Floor Support Beams and Lower Main Intermediate Longerons LH and RH Side M	All not changed by S/L 98-2	53
	Inspect — Lower Cap Angle in Aft Well F.S. 298+21 (Ref. S/B 91-3) M	All	53
	Inspect — Strap Below Escape Hatches M	All not changed by S/L 98-2	53
	Inspect — Aileron Bellcrank Support and Pulley Support — Landing Gear Emergency Release M	All not changed by S/L 98-2	53
	Inspect — ADF Loop and ADF Sense Antenna Support Structure and Upper Main Intermediate Longerons M	All not changed by S/L 98-2	53
	Inspect — Web of Bulkhead F.S. 298.5 M	All	53
60 Months	Hydrostatic Check — Engine Fire Extinguisher Agent Containers A	All	26
	Inspect — Fire Extinguisher System Double Check "T" Valve	All	26

A See Special Inspection, 5-10-07.

G Repair may require Aux Fuel Cell removal.

M See Special Inspection, 5-10-07, Chapter 5 — Corrosion Control/Inspection.

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INSPECTION/CHECK (Continued)

FREQUENCY	ITEM	EFFECTIVITY	REFERENCE/ CHAPTER
60 Months (Cont'd)	Inspect — Fire Extinguisher System Direction Valve	All	26
	Hydrostatic Check — Portable Fire Extinguisher Agent Container (Cockpit)	All	26
	Inspect — Speed Brake Inner Skin and Doubler and Sonic Doublers — Ailerons and Flaps M	All changed by S/L 98-2	27
	Inspect — MLG Strut Door for Corrosion — Ref: S/B 93-2	All	32
	Hydrostatic Check — Portable Oxygen Bottle, Class DOT 3AA, 1800 or 2100 A	All	35
	Inspect — Emergency Ground Escape Hatch Sealed Areas for Corrosion (Ref: S/B 87-5) M	All	52
	Inspect — Main Entrance Door M In Accordance with Corrosion Control Manual, SR-76-023	All	52
	Inspect — Vertical Hat Sections on Aft Side of F.S. 333, Floor of Aux Fuel Bay, Pylon Castings, and Upper Intermediate Longerons LH and RH F.S. 333 to 357 (Especially Near the Center of the Bulkhead) L M	All	53
	Inspect — Pylon Match Angles and Fuselage Skin G M	370-2 thru -6, -8 and -9 Changed by S/L 98-2	53
	Inspect — Inactive Escape Hatch Fittings and Mounting Pin Receptacles (S/B 87-4) M	All	53
	Inspect — Lower Flange and Skin F.S. 13 to 76, Lower Cap Angle of Nose Gear Beam B.P. 9 and Lower Electronics Bay M	All changed by S/L 98-2	53
	Inspect — Splice Between Lower Center Skin and Side Skin F.S. 99 to 156.25 LH and RH M	All changed by S/L 98-2	53
	Inspect — Skin Splice on Intermediate Longeron, LH and RH Side. Inspect Longeron Above Main Entrance Door. Inspect Lower Longerons, LH and RH Side M	All changed by S/L 98-2	53
	Inspect — Lower Longeron Along Threshold of Main Entrance Door and Lower Door Stop M	All changed by S/L 98-2	53
	Inspect — Floor Support Beam at F.S. 264 and Wing Attach Fittings LH and RH on the Fuselage M	All changed by S/L 98-2	53

A See Special Inspection, 5-10-07.
G Repair may require Aux Fuel Cell removal.
L Repair may require F.S. 333 interior removal.
M See Special Inspection, 5-10-07, Chapter 5 — Corrosion Control/Inspection.

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INSPECTION/CHECK (Continued)

FREQUENCY	ITEM	EFFECTIVITY	REFERENCE/ CHAPTER
60 Months (Cont'd)	Inspect — Aft Aux Fuel Cell Floor Support Structure G M	All changed by S/L 98-2	53
	Inspect — Radio Altimeter Antenna Interior and Exterior Doublers, and Fuselage Skin In Area M	All changed by S/L 98-2	53
	Inspect — Intermediate to Aft Upper Longeron Splice F.S. 357 LH and RH M	All changed by S/L 98-2	53
	Inspect — Skin Splice at F.S. 143 Lower Side Skin, LH and RH Side, Cockpit Glass Support Structures, Windshield Bow Supports, Intermediate Longerons and Skin Splice LH and RH, Cockpit Floors and Lower Longerons LH and RH Side. M	All changed by S/L 98-2	53
	Inspect — Upper Longeron Splice F.S. 206 LH and RH M	All changed by S/L 98-2	53
	Inspect — All Cabin Floors, Floor Support Beams and Lower Main Intermediate Longerons LH and RH Side M	All changed by S/L 98-2	53
	Inspect — Strap Below Escape Hatches M	All changed by S/L 98-2	53
	Inspect — Aileron Bellcrank Support and Pulley Support — Landing Gear Emergency Release M	All changed by S/L 98-2	53
	Inspect — ADF Loop and ADF Sense Antenna Support Structure and Upper Main Intermediate Longerons M	All changed by S/L 98-2	53
	Inspect — Horizontal Stiffeners on Fwd Side of F.S. 333 and Bulkhead to Side Skin Attach Angle G M	All	53
	Inspect — Pylon Attach Fitting F.S. 333 M	370-2 thru -6, -8 and -9 All changed by S/L 98-2	54
	Inspect — Vertical Stabilizer Rear Spar Cap M	All changed by S/L 98-2	55
	Inspect — Wing Tips, Wing Front Spar Leading Edge Skin, Rib and Stiffeners M	All changed by S/L 98-2	57
	DELETED		

G Repair may require Aux Fuel Cell removal.
M See Special Inspection, 5-10-07, Chapter 5 — Corrosion Control/Inspection.

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INSPECTION/CHECK (Continued)

FREQUENCY	ITEM	EFFECTIVITY	REFERENCE/ CHAPTER
5 Years or 5000 Hr. Ovhl. whichever occurs first	Inspect — Engine, G.E. CF700-2D-2 for Corrosion [A]	370-1 and -7 380-1 and Subs Not in compliance with G.E. S/B (CF700) 72-139	72 G.E. Maint. Manual SEI-187
10 Years or 5000 Hr. Ovhl. whichever occurs first	Inspect — Engine, G.E. CF700-2D-2 for Corrosion [A]	370-1 and -7 380-1 and Subs In compliance with G.E. S/B (CF700) 72-139	72 G.E. Maint. Manual SEI-187
Every Tire Change	Inspect — MLG and NLG Wheel Assemblies	All	32

[A] See Special Inspection, 5-10-07.

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TIME LIMITS

OVERHAUL

FREQUENCY	ITEM	EFFECTIVITY	REFERENCE/ CHAPTER
1200 Hours	Overhaul — DC Starter/Generator, G.E., P/N 2CM82J4	370-1 and -7; 380-1 and Subs	24
	Overhaul — Inverter (Rotary) Used for Windshield Heat (Rotary No. 2 LH, MGE23-400, MGE23-413, and MGH548-1) A	370-1 and Subs and 380-1 thru -3, -5, -7, -10, -12 and Subs without Solid State Inverters	24
	Overhaul — DC Starter/Generator, Bendix, P/N 30B58-3A or 3B	370-2 thru -6, -8 and -9	24
	Overhaul — Hydraulic Pump/Motor Assembly N	All single pump installations except P/N 265-580623 (Stratopower)	29
1500 Hours	Overhaul — Inverter (Rotary) Not Used for Windshield Heat (Rotary No. 1 RH, MGE23-400, MGE23-413, and MGH548-1) A	370-1 and Subs and 380-1 thru -3 -5, -7, -10, -12 and Subs without Solid State Inverters	24
	Overhaul — Fuel Boost Pump, P/N 201700-2	All (without "D" suffix S/N)	28
2000 Hours	Overhaul — Fuel Boost Pump, P/N 201700-2	All (with "D" suffix S/N)	28
	Overhaul — Aeronca Thrust Reverser, P/N 234-23000 A	370-1 and -7 and 380-1 and Subs not changed by S/L 60 (Extended 4000 hr Overhaul)	78
2200 Hours	Overhaul Engine, P&W JT12A-8 A	370-2 thru -6 -8 and -9 not changed by P&W S/B 2005	72

A See Special Inspection, 5-10-07.

N Hydraulic Pump/Motor Assembly, P/N 265-580623, overhaul is "On Condition."

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OVERHAUL (Continued)

FREQUENCY	ITEM	EFFECTIVITY	REFERENCE/ CHAPTER
2200 Hours (Cont'd)	Overhaul — Rohr Thrust Reverser Assembly, P/N 140-5400-1, -2, -3, and -4	370-2 thru -6, -8 and -9 not changed by S/B 74-10 (Improved Bogie P/N 140-5484-1) and S/B 74-11 (Improved Fwd Inbd Stang Brackets P/N 140-5483-1 RH and -2 LH) and 370-2 thru -6, and -8 not changed by S/B 72-7 (Replace Eng Exhaust Nozzle Assem Tailpipe)	78
	Overhaul — Rohr Thrust Reversers, P/N 140-5400-5 and -501, and -6 and -502	370-2 thru -6, -8, and -9 not changed by S/B 74-10 (Improved Bogie P/N 140-5434-1) and S/B 74-11 (Improved Fwd Inbd Stang Brackets P/N 140-5483-1 RH and -2 LH)	78
3000 Hours	Overhaul — Rohr Thrust Reverser Assembly, P/N 140-5400-5 and -6 (501 and 502 tailpipes)	370-2 thru -6, -8, and -9 changed by S/B 74-10 (Improved Bogie - P/N 140-5434-1) and S/B 74-11 (Improved Fwd Inbd Stang Brackets - P/N 140-5483-1 RH and -2 LH)	78
	Overhaul — Engine P&W JT12A-8 A	370-2 thru -6, -8, and -9 H	72
4000 Hours	Overhaul — Aeronca Thrust Reverser, P/N 234-23000 A	370-1 and -7 380-1 and Subs changed by S/L 60 (Extended 4000 hr Overhaul)	78
5000 Hours	Overhaul — Engine, GE CF700	370-1 and -7, 380-1 and Subs Complying with GE S/B 72-39 (Latest Revision)	72

A See Special Inspection, 5-10-07.

H Providing P&W S/B 2005 is incorporated not later than 2,200 hours.

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OVERHAUL (Continued)

FREQUENCY	ITEM	EFFECTIVITY	REFERENCE/ CHAPTER
60 Months	Perform Proof Pressure Test and Service in Accordance with Walter Kidde CMM 26-20-20 for Extinguisher (Water) P/N 892480	All	26
84 Months	Overhaul — Engine, P&W JT12A-8 A	370-2 thru -6, -8 and -9	71

A See Special Inspection, 5-10-07.

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TIME LIMITS

RETIREMENT

FREQUENCY	ITEM	EFFECTIVITY	REFERENCE/ CHAPTER
300 Hours	Replacement — Water Separator Delta P Sensor Filter Element	370-2 thru -6, -8 and -9,	21
	Replacement — Engine Fuel Pump Filter, P&W JT12A-8 Cartridge, P/N 49881	370-2 thru -6, -8 and -9	73
	Replacement — Engine Oil Filter,G.E. CF700 GE, P/N 3014T99P02 or Cuno, P/N 54371-01	370-1 and -7; 380-1 and Subs	79
400 Hours	DELETED		
600 Hours	Replacement — Entry Door Seal Sensing Lines Filter	As Installed	21
	Replacement — Hydraulic Reservoir Pressure Regulator Air Filter Element, P/N AN6237-1	All	29
900 Hours	Replacement — Hydraulic Filter Element (pressure and return), P/N AN6235-3A [A]	All not changed by S/L 83-10 (Improved Hyd. Filter Element)	29
	Replacement — Hydraulic Filter Element (pressure and return) P/N AC6091F-8Y82 [A]	All changed by S/L 83-10 (Improved Hyd. Filter Element)	29
	Replacement — Nose Steering Control Valve Filter Element, P/N AN6235-1A	All with Menasco Valve Assy, P/N 1176L001 not changed by S/L 30 (New Nose Steering Manifold)	32
1000 Hours [B], 2000 Cycles, or 3 Years, whichever occurs first	Replacement — APU Fuel Inlet Filter	All changed by S/B 71-5 (Inst. of APU)	49
	Replacement — APU Oil Filter, P/N 27570-0	All changed by S/B 71-5 (Inst. of APU)	49
1200 Hours	Replacement — Door Seal Hydraulic Reservoir Water Separator Filter Element, P/N 2488294	As Installed	21
1500 Hours	Replacement — Deicer Boot Water Separator Filter 2487866 Element, 2488294 Element, and O-Ring	Airplanes with deicer boots	30

[A] See Special Inspection, 5-10-07.

[B] APU Hours.

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RETIREMENT (Continued)

FREQUENCY	ITEM	EFFECTIVITY	REFERENCE/ CHAPTER
2000 Hours	Replacement — Flap Interconnect Coupling, P/N's Stewart SK6060A, S.S. White 2700-7579X-0D	370-1 and Subs and 380-1 thru -16, -18 thru -20, -22 thru -24, -26 thru -43, -45 thru -47, -51 thru -53, -55, -56, -58 and Subs	27
	Retirement — Wing Flap Flex Drive Shafts, LH and RH, P/N S.S. White 13468Y-115; Stewart LA-2672-115; 16125Y2	370-1 and Subs and 380-1 thru -16, -18 thru -20, -22 thru -24, -26 thru -43, -45 thru -47, -51 thru -53, -55, -56, -58 and Subs	27
2100 Hours	Replacement — Nose Compartment Air Pressure Regulator Filter Cartridge, P/N CU73927 or CU79030	370-1 thru -9 and 380-1 thru -4, -6 thru -9, -11, -13, -15, -17, -19, -21, -23 not changed by S/L 7 (Deletion of Pressurized Nose Compartment)	21
	Replacement — Cabin Air Pressure Regulator Filter Cartridge, P/N CU73927 or CU79030	All	21
2200 Hours	Retirement — Lower Door Links, Rohr Thrust Reverser, P/N 140-5409-1 or -3 [A]	370-2 thru -6, -8 and -9	78
3000 Hours	Retirement — Engine Mounts, G.E. CF700, P/N 93285-2 thru -6	370-1 and -7 and 380-1 and Subs changed by SOK 19 (Addition of Engine Mount Vibration Isolation)	71
10,000 Hours	Retirement — Rudder Center Hinge, P/N 265-240428	All	27
	Retirement — Outboard Flap Track, P/N 265-180051-21	All	27
10,000 Hours or 10,000 Flights, whichever occurs first	Retirement — Airframe	370-1 thru -9; 380-1 thru -72 not changed by SOK 81-3 (Service Life Extension)	5

[A] See Special Inspection, 5-10-07.

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RETIREMENT (Continued)

FREQUENCY	ITEM	EFFECTIVITY	REFERENCE/ CHAPTER
30,000 Hours or 30,000 Landings, whichever occurs first	Retirement — Airframe	All changed by SOK81-3 (Service Life Extension)	5
24,050 Landings	Retirement — MLG Outer Cylinder, P/N 1127L001-3, -4, -5, and -6	370-1 thru -9	32
26,000 Landings	Retirement — MLG Outer Cylinder, P/N 1127L001-3, -4, -5, and -6	380-1 thru -16, -18 thru -20, -22 thru -24, -26 thru -43, -45 thru -47, -51 thru -53, -55, -56, -58 and Subs	32
60 Months	Replacement — Fire Extinguisher Discharge Cartridge, P/N 30903819 (HTL) or 841155 (Kidde)	All	26
	Replacement — APU Fire Extinguisher Discharge Cartridge P/N 30903819 (HTL) 841155 (Kidde)	All changed by S/B 71-5 (Inst. of APU)	49
	Replacement — Flexible Fuel, Oil and Hydraulic Hoses in Pylon and Engine Nacelle as follows [I]:		
	Fuel Hose Assy, Stratoflex LH, Nacelle to Eng, P/N 111391 [I] (P&W JT12A-8) [P]	370-2 thru -6, -8 and -9	73
	Fuel Hose Assy, Stratoflex RH, Nacelle to Eng, P/N 111380-12-0166 [I] (P&W JT12A-8) [P]	370-2 thru -6, -8 and -9	73
	Eng Fuel Hose Assy P&D Valve Reservoir Drain Tank, P/N S01-3201-013, -015, -017, -031, -033, -035 and -037 (P&W JT12A-8) [J] [P]	370-2 thru -6, -8 and -9 with P&D Valve Reservoir Drain Tank Kit, P/N S01-3201-001 (LH), -003 (RH)	73
	Eng Fuel Supply Hose LH, P/N 130101J0134F343 (Titeflex), or 13010J01345-343 [I] G.E. CF700 [P]	370-1 and -7; 380-1 and Subs	73
	Eng Fuel Supply Hose RH, P/N 130101J0134F017 (Titeflex), or 13010J01345-017 [I] G.E. CF700 [P]	370-1 and -7; 380-1 and Subs	73

[I] See Illustrated Parts Catalog (NA-69-424) or Sabreliner Corporation for any approved alternate part number hose assembly that may exist in this location.

[J] Hose assemblies are part of fuel drain tank kit installed by STC number SA1162CE.

[P] New or remanufactured Hose Assy.

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RETIREMENT (Continued)

FREQUENCY	ITEM	EFFECTIVITY	REFERENCE/ CHAPTER
60 Months (Cont'd)	Tube Assy (Fuel Hose) RH, P/N 130S5014-0130 (Titeflex), or TR-5025CC4-0130 [I] G.E. CF700 [P]	370-1 and -7; 380-1 and Subs not changed by GE S/B 73-34	73
	Tube Assy (Fuel Hose) LH, P/N 130S5014-0150 (Titeflex), or TR-5025CC4-0150 [I] G.E. CF700 [P]	370-1 and -7; 380-1 and Subs changed by GE S/B 73-34	73
	Tube Assy (Fuel Hose) Pump Press Tap, P/N 130S5014-0170 (Titeflex), or TR-5025CC4-0170 [I] G.E. CF700 [P]	370-1 and -7; 380-1 and Subs	73
	Fuel Recovery Sys Hose, P/N R50200CC6-0160 (Resistoflex), or 124001-6CR-0160 [I] G.E. CF700 [P]	370-1 and -7; 380-1 and Subs	73
	Tube Assy (Fuel Hose) Fuel Pump to Filter, P/N R28174 (Resistoflex) [I] G.E. CF700 [P]	370-1 and -7; 380-1 and Subs	73
10 Years	Replacement — Fire Extinguisher Discharge Cartridge P/N 841155-1 (Kidde)	All	26
15 Years or 4380 Bottle Pressurization Cycles, whichever occurs first	Retirement — Oxygen Bottle ICC or DOT-3HT-1850 Cylinders identified as Special Permit, DOT-SP5967 [A]	All as equipped	35
24 Years or 4380 Bottle Pressurization Cycles, whichever occurs first	Retirement — Oxygen Bottle ICC or DOT-3HT-1850 NOT identified as Special Permit, DOT-SP5967 [A]	All as equipped	35
3600 APU Cycles	Replacement — APU Turbine Wheel [A]	All with APU	49
7200 APU Cycles	Replacement — APU Compressor Wheel [A]	All with APU	49

[A] See Special Inspection, 5-10-07.

[I] See Illustrated Parts Catalog (NA-69-424) or Sabreliner Corporation for any approved alternate part number hose assembly that may exist in this location.

[P] New or remanufactured Hose Assy.

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TIME LIMITS

SPECIAL INSPECTION INFORMATION

GENERAL

Special inspection information is listed in this section to further explain various servicing, adjustment/test, inspection/check, overhaul and retirement time limit requirements. Note A, listed in sections 5-10-02 thru 5-10-06 and 5-20-01 indicates time limit items which are explained in more detail in this section.

This Special Inspection information has been arranged in ATA 100 specifications, like that of the maintenance manual. This will assist the reader in locating data with its associated chapter.

CHAPTER	ITEM
5	• CORROSION CONTROL/INSPECTIONS

Effective corrosion control is accomplished in two phases: (1) preventive maintenance, and (2) corrective maintenance. Preventive maintenance includes frequent inspections and cleaning. Corrective maintenance includes the complete removal of corrosion products and renewal of protective coatings. Both phases are of equal importance in any successful program of corrosion control. Whenever any evidence of corrosion is found (i.e., paint chipping, bubbling of skin or seams, exfoliation, etc.) further investigation will be required. This may include, but is not limited to, interior removal and structural disassembly as required to inspect and take the necessary corrective action. Where corrosion has progressed unchecked, the resulting damage may require replacement of parts or major components and assemblies. Conscientious inspections will reveal scratches and/or abraded areas in protective coatings or plated surfaces, minor areas of corrosion, build up of corrosive materials (i.e., salt spray, dirt, chemicals, etc.). All these conditions can result in expensive repairs if allowed to continue untreated.

The Following Basic Procedures Are Essential to the Success of a Corrective Corrosion Maintenance Program:

1. All corrosion must be removed. It may seem acceptable to leave the bottom of corrosion pits to avoid removing the metal surrounding the pit, but this will only allow corrosion to continue. Pitting corrosion should be blended out all the way to the tip of the pit. Pits are sites for fatigue crack initiation. The bottom of a pit will be an area of stress concentration. The smaller the radius at tip of the pit, the higher the stress concentration factor.
2. Don't assume the corrosion stops where the corroded surface meets another part (i.e., corrosion may continue past visual areas into faying surfaces and hidden areas).
3. While specific areas, based on past history, have been identified for thorough inspection, all areas in and around the identified area should receive a close visual inspection. When indications exist and dictate, additional airframe disassembly must be accomplished to expose the corrosion and allow for proper corrosion removal, treatment, and/or repair, early recognition and treatment of corrosion can drastically reduce the costs associated with long term aircraft preventive maintenance.
4. The structure must be returned to its original strength if corrosion extends beyond allowable limits set by the Corrosion Control Manual. Repairs should be made in accordance with the Structural Repair Manual.
5. Clean the area thoroughly and apply the finish system to the part in accordance with the Corrosion Control Manual.

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SPECIAL INSPECTION INFORMATION (Continued)

CHAPTER	ITEM
5 (Cont'd)	**Purpose of Inspections**

Text and illustrations are provided to help operators of Sabreliner Aircraft establish an ongoing corrosion inspection program. This information is based upon the experience and extensive history compiled by the original aircraft manufacturer. Data from Service Publications, Material Reviews, Repair Station Discrepancy Workbooks, and Technical Manuals has been used in an attempt to address all areas previously identified as requiring special attention.

NOTE

The manufacturer does not represent this data as a complete and all inclusive Corrosion Inspection Program. It is therefore incumbent upon each operator to determine the required detail and frequency of inspection as dictated by the particular operational environment involved and increase the inspection requirements as necessary.

Personnel Qualifications

All personnel responsible for maintaining an aircraft must be familiar with the applicable maintenance practices as set forth by the manufacturer (i.e., Maintenance Manual, Corrosion Control Manual, Structural Repair Manual, Service Publications, etc.) and adequately trained to recognize evidence of corrosion.

21 • **FLUID PRESSURE REGULATING VALVE, 3-WHEEL COOLING TURBINE**

Reference Garrett AiResearch Report No. 4-273 for information regarding the proper procedure for checking the regulation setting of the fluid pressure regulating valve.

23 • **STATIC DISCHARGERS**

See Unscheduled Maintenance Checks, 5-50-00.

24 • **DC STARTER/GENERATOR BRUSH INSPECTION**

When the top of the brush(es) is even (flush) with the top of the brush holder, continued inspections at 150 hour intervals are recommended. When the top of the brush recedes to 1/4 in. below the top surface of the brush holder, the brushes must be replaced. Bendix starter/generator brushes require bench run-in, and are generally not field replaceable.

• **BATTERY TEMPERATURE TEST**

Airplanes that have a battery temperature system with automatic control isolation system installed after production should check system in accordance with the installing agency's instructions.

• **INVERTER REPLACEMENT**

When a P/N MGE 23-400, MGE 23-413, or MGH 548-1 Motor Generator (Inverter) is installed in the No. 2 (LH) position on airplanes with inverter powered windshield heating, the time between overhauls is 1200 hours. Otherwise it is 1500 hours.

• **STARTER/GENERATOR REPLACEMENT OR GENERATOR BRUSH REPLACEMENT**

Whenever a starter/generator is replaced or the generator brushes are replaced, an inspection of the brushes should be made after the first 100 hours of operation. (Refer to chapter 80).

25 • **COCKPIT AND CABIN HEATER PAD INSPECTION**

Airplanes not complying with S/B 85-3, Part I, should inspect heater pads at 300 hours. Airplanes complying with S/B 85-3, Part I, should inspect the heater pads at 600 hours. Airplanes complying with S/B 85-3, Part II, should inspect the heater pads every 48 months.

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SPECIAL INSPECTION INFORMATION (Continued)

CHAPTER	ITEM
26	• **ENGINE FIRE EXTINGUISHER CONTAINERS** DELETED.
	• **PORTABLE FIRE EXTINGUISHERS** Weight check all portable fire extinguishers per manufacturers' requirements, as indicated on bottle or accompanying instructions.
27	• **INSPECTION, ELEVATOR HONEYCOMB TRAILING EDGE** 1. The inspection shall be accomplished on new airplanes as follows: A. At the first 1000 hours or 12 months, whichever occurs first, and; B. Again at each following 500 hour or 6 month interval, whichever occurs first. 2. If the airplane has exceeded 1000 hours and/or 12 months, inspect as follows: A. As soon as possible, but no later than the next 150 hours or 30 days, and; B. Again at each following 500 hour or 6 month interval, whichever occurs first. 3. Delaminated trailing edges must be replaced. 4. If replacement trailing edges are honeycomb, reinspect as follows: A. At the first 1000 hours or 12 months, whichever occurs first, and; B. Again at each following 500 hour or 6 month interval, whichever occurs first. 5. Installation of non-honeycomb trailing edges shall terminate additional inspection requirements. 6. Reference Illustrated Parts Catalog (NA-69-424) for replacement of trailing edges.
28	• **FUNCTIONAL TEST FUEL TRANSFER EJECTOR FILTER SYSTEM TIME LIMITS** At the conclusion of the first 20 hours of airplane operation, perform an operational check of the ejector pumps. If the pumps are found to be operating properly and are free of contaminants, another inspection must be made at 100 hours of airplane operation. If pumps pass this second inspection, another inspection must be made at the conclusion of the next 200 hours of airplane operation. If the pumps are still operating properly at this time, the inspection may be performed every 600 hours thereafter. Procedures for checking fuel transfer ejectors for proper operation are given in chapter 28. Anytime an ejector pump is found inoperative because of contaminants in the nozzle, ejector operation should be checked after every 10 hours of airplane operation, until the ejectors are found free of contaminants. When the ejectors are found free of contamination, the inspection interval should revert back to 100 hours, then 200 hours, and then 600 hours as previously mentioned.
	• **WING JOINT RIB ASSEMBLY/FUEL DRAIN VALVES** The wing joint rib assembly corrosion inspection should be performed on a 24 month cycle or concurrent with inspecting Fuel Drain Valves for corrosion.

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SPECIAL INSPECTION INFORMATION (Continued)

CHAPTER	ITEM
29	• **HYDRAULIC FILTER ELEMENT (Pressure and Return)**

P/N AN6235-3A filter, a 10-micron filter, is the standard filter used in the hydraulic pressure and return system. P/N AC6091F-8Y82 filter, a 3-micron filter, replaces P/N AN6235-3A filter per S/L 83-10. Although the new 3-micron filter is a direct replacement for the 10-micron filter, it can be used only when installed per S/L 83-10. S/L 83-10 provides detailed infor-mation regarding the flushing of the hydraulic system, as well as installation of the 3-micron filter. Reference S/L 83-10 for complete details.

30 • **APPLICATION OF ICEX II TO AILERON CABLE SECTORS**

If severe environmental conditions exist, ICEX II should be applied every 300 hours.

32 • **BRAKE WEAR INSPECTION**

Dependent on the operational status of the airplane (flights of a short duration, consequently more landings), the brake wear inspection may be required at a more frequent interval. Refer to Chapter 32 for inspection procedures.

• **MAIN LANDING GEAR DOWNLOCK SIDEBRACE IDLER ROD END**

Once any solid rod end has been properly etched, per S/B 81-7, the 300 hour inspection requirement is deleted.

35 • **OXYGEN CYLINDERS**

1. DOT 3HT 1850 Class oxygen cylinders shall be hydrostatic tested every 3 years (36 months) following date of manufacture. They are Service Life limited to 24 years from manufacture date, or after 4380 bottle pressurization cycles, whichever occurs first.
2. DOT 3AA 1800 Class oxygen cylinders shall be hydrostatic tested every 5 years (60 months) following date of manufacture. There is no Service Life limit if the hydrostatic tests are satisfactory.
3. Oxygen cylinders manufactured under an identified (stamped on cylinder) "Special Permit, DOT SP-5967" are Service Life limited to 15 years from manufacture date. They shall be hydrostatic tested every 3 years (36 months) following date of manufacture.
4. If a cylinder is recharged more than an average of once every other day, an accurate record of the number of rechargings must be maintained by its owner or agent.

49 • **APU**

The inspection intervals are applicable to APU's operated less than 600 hours per year in accordance with Sundstrand Service Bulletin SB-T-62T-49-77. If usage exceeds 600 hours per year, refer to Sundstrand Maintenance Manual T-62T-39.

NOTE

Drain oil sump and replenish with new oil at 12 month intervals. The 12 month limitation is due to the anti-foam service life of MIL-L-23699. MIL-L-7308, or commercial equivalent oils.

In addition to the 3600 cycle turbine wheel or the 7200 cycle compressor wheel replacement, it is quite probable that other related components will be replaced. Any components of the APU, other than the turbine wheel or the compressor, are replaced on an as-needed basis. Reference Sundstrand Power System Service Bulletin SB-T-62T-49-76 (revised Feb 8/90 or later) for further details regarding the replacement of APU life limited components.

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SPECIAL INSPECTION INFORMATION (Continued)

CHAPTER	ITEM
53	• **INSPECT — CLIPS, F.S. 76**

Operators should check log books for evidence of heavy duty clips, P/N 306-311076-003 and -004.

71 • **ENGINE MOUNT BOLTS (S/N 370-2 thru -6, -8 and -9)**

Retorque and apply torque striping to main engine pylon trunnion mount bolts any time any engine is removed and reinstalled. Visually inspect for loose engine mount support bolts and/or torque striping at 300 hour intervals.

• **ENGINE OVERHAUL (P&W JT12A-8) (S/N 370-2 thru -6, -8 and-9)**

A maximum calendar period of seven years between overhauls is recommended for engines. This time is considered to be exclusive of any period the engine is properly preserved for long term storage.

JT12A-8 engines may be operated to 3000 hours or 84 months, whichever occurs first between overhauls, providing P&W S/B 2005 is incorporated not later than 2200 hours and the combustion area inspected interval does not exceed 1000 hours. The monthly interval may be extended provided a satisfactory inspection of the compressor section is accomplished in accordance with P&W inspection requirements.

Additionally, an engine which meets inspection criteria, per P&W S/B 1508, during the seventh calendar year can be allowed an extension of twenty-four (24) months from the time of this inspection to full overhaul providing the recommended disk hourly and cycle lives are not exceeded.

Thereafter, during the ninth, eleventh, etc. years (24 month intervals) recurring inspections can be done to gain additional 24 months calendar extension from the time of this inspection with the same provisions that the published disk lives are not exceeded.

• **ENGINE CORROSION INSPECTION (GE CF 700-2D-2) (S/N 370-1 and Subs and 380-1 and Subs)**

1. Five Year Corrosion Inspection Requirement.

   Engines not in compliance with the extended life compressor as defined in G.E. Service Bulletin (CF700) 72-139 which do not reach overhaul within a 5 year period from date of installation in aircraft, must be scheduled for a compressor corrosion inspection.

2. Ten Year Corrosion Inspection Requirement.

   Engines in compliance with GE Service Bulletin (CF700) 72-139 (Extended Life Compressor) must be scheduled for a compressor corrosion inspection every 10 years. Also, only engines which contain vane segments made from inco material qualify for a 10 year corrosion inspection. This inspection can be accomplished during replacement of cycle-limited parts, during engine overhaul or 10 year calendar.

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SPECIAL INSPECTION INFORMATION (Continued)

CHAPTER	ITEM
78	• **AERONCA THRUST REVERSER P/N 234-23000**

Any thrust reverser that has accumulated over 2000 hours (without compliance with S/L 60) or 4000 hours (with S/L 60 accomplished) must either be overhauled by Aeronca or accomplish a One Time Inspection and Rework by Sabreliner Corporation Repair Station No. JV2R940K (formerly 3001) at Perryville, Missouri. Recurring maintenance is required after complying with the initial requirements above.

1. A recurring 2000 hour or 4000 hour overhaul of the reverser (interval determined by compliance with S/L 60) by Aeronca is required after the initial overhaul by Aeronca. Overhauled reversers must also accomplish a 1000 hour recurring retorque of the translating structure.
2. A One Time Inspection and Rework of the reverser by Sabreliner Corporation will qualify the reverser for recurring 1000 hour/1000 cycle inspection by any certified mechanic or FAA approved repair station. The retorque of the translating structure is included in a 1000 hour/1000 cycle inspection.

79 • **THRUST REVERSER — LOWER DOOR LINKS (ROHR)**

The retirement interval for the lower door links for the thrust reverser, P/N 140-5409-1, designates the retirement for the entire assembly. Regardless of any dash number link in use, it becomes a -1 when the rod end balls are inserted into the link. This -1 refers to an assembly, and this retirement interval is 2200 hours. When a lower door link is retired, the bearings must be replaced as well.

HOSES — GENERAL

- The "service life" of any hose begins at the time of installation.

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SCHEDULED MAINTENANCE CHECKS

PRE-FLIGHT INSPECTION

INSPECTION PROCEDURES

1. Electrical power .....on
2. Wing FLAP handle ..... DN
3. Electrical power ..... off
4. Remove maintenance safety locks (if installed).
5. Remove pitot and engine covers (if installed).
6. Drain fuel sumps (first flight of the day only).
7. Inspect the following areas and items for:
   - Evidence of fuel leaks
   - Excessive or abnormal wear
   - Security of attachment
   - Skins for cracks or damage
   - General airplane condition

A. Fuselage Forward

Speed brake
OAT probe
Ice lights
Entrance door seal
Cooling air outlet
Cooling air inlet
Rotating beacon
Landing and taxi lights
Pitot tubes
Battery vents
Radome
Windows/windshields
Windshield wiper/wiper blades
AOA transmitter probe
Static vents

B. Nose Landing Gear

Hydraulic actuators (leaks and condition)
Tires (inflation/damage)
Wheels
Strut (extension/leaks)
Balance weight and bracket
Doors (linkage)
Steering collar/linkage

C. Main Landing Gear RH

Hydraulic actuators (leaks and condition)
Tires (inflation/damage)
Wheels
Brakes (hoses/wear/leakage)
Strut (extension/leaks)
Fire bottle (specified pressure)
Doors
Drag brace
Load switches and electrical wiring

D. Right Wing

Stall warning vanes
Static wicks
Single point refueling cap
Leading edge
Wing tip
Navigation/strobe/recognition lights
Flaps/fairings/tracks/screwjacks/flex shafts
Fuel filler cap (gravity fill)
Fuel vent
Aileron
Wing slats (operation)
Deicer boots (if installed)

E. Fuselage & Power Plant Aft RH

Engine air intake
Cooling air inlets
Cooling air outlets
Engine oil level
Engine cowling
APU inlet and exhaust ducts
Thrust reversers (doors & latches) C
Engine Exhaust
Fuel filler cap (fuselage gravity fill)
Aft fairing D
Pylon
Control cables
Cabin windows
Access doors (secured)
Engine fan inlet duct D

F. Empennage

Ram air inlet-obstructions (bird stopper door if installed)
Vertical fin (right side)
Horizontal stabilizer RH
Elevator RH

C S/N 370-2 thru -6, -8, and -9
D S/N 370-1 and -7; 380-1 and subs

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PRE-FLIGHT INSPECTION (Continued)

F. Empennage (Cont'd)

Fuel vent
Flame arrestor
Navigation/strobe lights
Rudder
Rudder tab
Rotating beacon/strobe light
Elevator LH
Horizontal stabilizer LH
Vertical fin (left side)
Static wicks
Lightning rods
Tail cone
Antennas

G. Fuselage & Power Plant Aft LH

Control cables
Fire bottle discharge indicators
Oxygen bottle discharge indicator
Thrust reversers (doors & latches)
Engine exhaust
Engine cowling
Cooling air inlets
Cooling air exhausts
Engine oil level
Engine air intake
Aft fairing D
Check hydraulic reservoir air pressure (35 psi)
Normal hydraulic accumulator pressure
Auxiliary hydraulic accumulator pressure
Thrust reverser hydraulic accumulator pressure C
Pylon
Cabin windows
Access doors (secured)
Engine fan inlet duct D

H. Main Landing Gear LH

Hydraulic actuators (leaks and condition)
Tires (inflation/damage)
Wheels
Brakes (hoses, wear, leakage)
Strut (extension/leaks)
Doors
Drag brace
Load switch and electrical wiring
Fire bottle (specified pressure)

I. Left Wing

Stall warning vanes D
Flap/fairings/tracks/screwjacks/flex shafts
Aileron
Aileron tab
Static wicks
Wing tip
Navigation/strobe/recognition lights
Fuel vent
Fuel filler cap
Leading edge
Wing slats (operation)
Deicer boots (if installed)

J. Cabin

Circuit breakers (closed)
Flashlights
Oxygen valve (open)
Oxygen (1800 psi)/regulators/hoses
Portable fire extinguisher
Emergency escape hatches
Seats and seat belts
Cabin furnishings
Galley
Lavatory
Lights (cockpit/cabin)
Windows
Instruments (slippage marks and range markings)

8. Close all access doors or panels if applicable.
9. Electrical power ..... on
10. Wing FLAP handle ..... UP
11. Check engine/flight controls for freedom of operation.
12. Electrical power ..... off

C S/N 370-2 thru -6, -9, and -9
D S/N 370-1 and -7; 380-1 and subs

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SCHEDULED MAINTENANCE CHECKS

POST-FLIGHT INSPECTION

INSPECTION PROCEDURES

1. Check engine oil level (within 1 hour after shutdown). Service as required.
2. Electrical power ..... on
3. Check the following lights:
   - A. Navigation
   - B. Recognition
   - C. Landing-Taxi
   - D. Anti-Collision
   - E. Strobe
   - F. Ice
4. Perform the following:
   - A. Wing FLAP handle ..... DN
   - B. SPEED BRAKE switch ..... OUT
   - C. Install ground maintenance safety locks as required.
   - D. HYDRAULIC PUMP switch ..... OFF & RESET
   - E. Dump all hydraulic accumulators.
5. Inspect the following areas and items for:
   - Evidence of fluid leaks
   - Excessive or abnormal wear
   - Security of attachment
   - Cracked or damaged skin
   - General airplane condition

A. Fuselage Forward

OAT probe
Ice lights
Entrance door seal
Cooling air outlet
Cooling air inlet
Rotating beacon
Landing and taxi lights
Pitot mast and static vents
Battery vents
AOA transmitter probe
Radome
Windows/windshields
Windshield wiper/wiper blades
Speed brake

B. Nose Landing Gear

Hydraulic actuators (leaks and condition)
Tires (inflation/damage)
Wheels
Strut (extension/leaks)
Balance weight and bracket
Doors (linkage)
Steering collar/linkage

C. Main Landing Gear RH

Hydraulic actuators (leaks and condition)
Tires (inflation/damage)
Wheels
Brakes (hoses/wear/leakage)
Strut (extension/leaks)
Fire bottle (specified pressure)
Doors
Drag brace
Load switches and electrical wiring

D. Right Wing

Single point refueling cap
Leading edge
Wing tip
Navigation/strobe/recognition lights
Fuel filler cap (gravity fill)
Fuel vent
Aileron
Flap/fairings/tracks/screwjacks/flex shafts
Static wicks
Wing slats (operation)
Stall warning vanes
Deicer boots (if installed)

E. Fuselage & Power Plant Aft RH

APU exhaust and inlet ducts
Engine air intake
Cooling air inlets
Cooling air outlets
Engine cowling
Thrust reversers (doors & latches) C
Engine exhaust
Fuel filler cap (fuselage gravity fill)
Aft fairing D
Cabin windows
Pylon
Access doors secured
Engine fan inlet duct D

C S/N 370-2 thru -6, -8, and -9
D S/N 370-1 and -7; 380-1 and subs

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POST-FLIGHT INSPECTION (Continued)

F. Open Aft Fuselage Access Doors and Inspect APU (if installed) and Air Conditioning Components for the following:

Electrical harness condition
Fuel and oil leaks
Air system leaks
APU oil level (if installed)
Air conditioning turbine oil level (3-wheel turbine)
APU and air conditioning components for security
Accessible flight control cables and servos
Hydraulic reservoir fluid quantity
Hydraulic reservoir air tank (35 psi)
Normal accumulator (1500 psi)
Auxiliary accumulator (1500 psi)
Thrust reverser accumulator (900 psi) C

G. Empennage

Ram air inlet - obstructions (bird stopper door if installed)
Vertical fin (right side)
Horizontal stabilizer RH
Elevator RH
Fuel vent
Flame arrestor
Navigation/strobe lights
Lightning rods
Rudder
Rudder tab
Rotating beacon/strobe light
Elevator LH
Horizontal stabilizer LH
Vertical fin (left side)
Antennas
Static wicks
Tail cone

H. Fuselage & Power Plant Aft LH

Aft fairing D
Fire bottle discharge indicators
Oxygen bottle discharge indicator
Thrust reversers (door & latches) C
Engine exhaust
Engine cowling
Cooling air inlets
Cooling air exhausts
Engine air intake
Pylon
Cabin windows
Access doors (secured)
Engine fan inlet duct D

I. Main Landing Gear LH

Hydraulic actuators (leaks and condition)
Tires (inflation/damage)
Wheels
Brakes (hoses/wear/leakage)
Strut (extension/leaks)
Doors
Drag brace
Load switch and electrical wiring
Fire bottle (specified pressure)

J. Left Wing

Stall warning vanes D
Flap/fairings/tracks/screwjacks/flex shafts
Aileron
Aileron tab
Wing tip
Navigation/strobe/recognition lights
Fuel vent
Fuel filler cap
Leading edge
Static wicks
Wing slats (operation)
Deicer boots (if installed)

K. Cabin

Circuit breakers (closed)
Oxygen valve (open)
Oxygen (1800 psi)/regulator/hoses
Portable fire extinguisher
Emergency escape hatches
Seats and belts
Galley
Lavatory
Windows
Cabin furnishings
Lights (cockpit and cabin)
Flashlights
Instruments (slippage marks and range markings)

6. Ground maintenance safety locks ..... remove
7. Hydraulic system ..... dump and check
8. Wing FLAP handle .....UP
9. SPEED BRAKE switch..... IN
10. Electrical power ..... off
11. Check flight and engine controls for freedom of operation.
12. Pitot and engine covers ..... install
13. Airplane..... service as required
14. Airplane..... secure as desired

C S/N 370-2 thru -6, -8, and -9
D S/N 370-1 and -7; 380-1 and subs

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SCHEDULED MAINTENANCE CHECKS

150 HOUR MAINTENANCE CHECK

INSPECTION PROCEDURES

1. 28 VDC electrical power .....on
2. Circuit breakers as follows:
   - A. LH ENG-MASTER .....out
   - B. RH ENG-MASTER .....out
   - C. LH FUEL-BOOST PUMP .....out
   - D. RH FUEL-BOOST PUMP .....out
   - E. LH & RH VIBRATORS .....out
   - F. EQUIP GRD BLO-SECOND CONT .....out
3. Perform the following:
   - A. SPEED BRAKE .....open
   - B. PARKING brake handle ..... set
   - C. Ground maintenance safety locks ..... install
   - D. HYDRAULIC PUMP switch ..... OFF
   - E. Wheel well dish pans ..... remove
   - F. FLAPS .....down
4. Engine ignition system check as follows:
   - A. LH & RH ENGINE MASTERS .....ON
   - B. IGNITION switch .....CONTINUOUS
   - C. AIRSTART switch ..... OFF
   - D. LH power lever..... IDLE
     Check for audible firing of igniter plug and IGNITION light on
   - E. IGNITION switch .....NORMAL
     Igniter plug firing should cease.
   - F. AIRSTART switch..... LEFT HAND ON
     Check both igniters for firing.
   - G. LH power lever..... CLOSE
     Igniter firing should cease.
   - H. IGNITION switch .....CONTINUOUS
   - I. RH power lever..... IDLE
     Check for audible firing of igniter plug and IGNITION light on.
   - J. IGNITION switch .....NORMAL
     Igniter plug should cease firing.
   - K. AIRSTART switch..... RIGHT HAND ON
     Check both igniters for firing.
   - L. RH power lever..... CLOSE
     Igniter firing should cease.
   - M. LH & RH ENGINE MASTERS ..... OFF
   - N. AIRSTART switch..... OFF
5. Check trim tab system in normal and alternate positions for operation and stabilizer trim on individual actuators.
6. Check flap actuators for operation on individual actuators. Leave flaps extended.
7. Inspect all flexible fuel, oil, and hydraulic hoses for the following:
   - Leakage
   - Stiffness
   - Discoloration
   - Kinking
   - Service life
8. Check the following for proper operation:

   Check trim disconnect system (airplanes changed by S/B 74-6).

   Navigation lights

   Landing - taxi lights

   Anti-collision lights

   Ice lights

   Pitot heat system

   EPR heat system

   Strobe lights

   Recognition lights
9. Inspect the following areas and items for:
   - Evidence of fluid leaks
   - Excessive or abnormal wear
   - Security of attachment (safetywire)
   - Bent push-pull rods
   - Cracked or damaged skins
   - Corrosion damage or distortion

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150 HOUR MAINTENANCE CHECK (Continued)

9. Inspect the following areas and items for: (Cont'd)
   - Loose bearings
   - Bent attaching brackets
   - Frayed control cables
   - Equipment furnishings for security
   - General airplane condition

A. Fuselage Aft LH

Access doors
Exterior
Skins
Fan discharge duct (oil mist)
Antennas
Fire bottle discharge indicators
Oxygen bottle discharge indicator
Interior
  APU (if installed)
  APU fire bottle (if installed)
  Air conditioning components
  Hydraulic components
  Flight controls and cables
  Electrical wiring
Pylon
Pylon access plates
Pylon drains
Pylon to nacelle seals
Cabin windows
Fuel drains
External hydraulic power fittings and access door
Fuel heater (exhaust duct and drain)
External electrical power receptacle (microswitch)

B. Power Plant LH

Thrust reverser
  Doors (latches, hinge points)
  Exhaust nozzle (cracks)
Actuator (hydraulic fluid leaks) C
Aft fairing, skin/cracks/corrosion
Aft fairing access doors, skin/cracks/corrosion
Engine cowling access doors and louvers
Engine drain manifold
Generator cooling air inlets
Generator cooling air exhaust (carbon brush dust)
Electrical wiring
Plumbing lines
Inlet area
  Inlet duct
  Fan Inlet D
  Variable inlet guide vanes D
  Free rotation of compressor and blade condition
  Bullet nose
  Inlet pressure port and sense probe
  Inlet temperature sensor C
  Splitter vanes C
  Inlet guide vanes C
Exhaust area
  Tailpipe and cone
  Struts
  Turbine blades for free rotation and condition C
  Fan for free rotation and blade condition
    EGT thermocouples and EPR probes D

C. Wing LH

Exterior
Deicer boots (if installed) D
Skins
Flap/tracks/screwjacks/flex shafts
Aileron, tab, and bonding wires
Hinges and pins (bolts)
Bellcranks, sector, cables, and pulleys
Static wicks
Navigation light
Anti-collision light/strobe light/recognition light
Fuel filler cap
Leading edge
Single-point refueling cap
Fuel drains
Required markings (legibility)
Wing tip fuel vent outlet
Wing slats, tracks, bearings, slat bumpers
Stall warning vane D

D. Main Landing Gear LH

Tire (inflation/damage)
Wheels
Brake/shuttle valve/hoses
Brake discs wear A
Static ground wire
Strut (extension/leaks)
Landing gear doors
  Torque tube
  Torque links
Actuating cylinders, linkages, and hoses

A See Special Inspection, 5-10-07.
C S/N 370-2 thru -6, -8 and -9
D S/N 370-1 and -7; 380-1 and subs

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150 HOUR MAINTENANCE CHECK (Continued)

D. Main Landing Gear LH (Cont'd)

Uplocks/linkage
Gear upstop support
Downlock assembly
Hydraulic lines
Electrical wiring and switches
Fire extinguisher bottle (specified pressure)
Fuel pump seal drain
Emergency landing gear cables, pulleys, and brackets
Fuel and hydraulic lines (leaks, chafing, security)

E. Fuselage Forward LH

Access doors
Exterior
Skins
Ice light
Antennas
Rotating beacon
Speed brake
Cabin door
Door seal
Fairing door
Cooling air inlets
Cooling air outlets (excessive carbon brush dust)
Cockpit windows
Windshield
Windshield wiper
Undersurface (fluid leakage)
Pitot tube (obstructions)
Static ports
Radome (condition of rain erosion coating)
Landing - taxi lights
Interior (nose)
Batteries and vent system
Fluid leaks
Fuel leaks
OAT probe

F. Nose Landing Gear

Tires (inflation/damage)
Wheels
Strut (extension/leaks, trunnion)
Landing gear doors/seals
Actuating cylinders/drag brace/hoses
Uplock
Downlock assembly
Nose steeering assembly
Hydraulic lines
Electrical wiring
Electrical switches/linkage
Torque links
Steering collars/latch/linkage
Wheel retaining nuts (torque, safetywire)
Balance weight and bracket
Vacuum pump

G. Fuselage Forward RH

Access doors
Exterior
Skins
Cockpit windows
Windshield
Windshield wiper
Pitot tubes (obstructions)
Static ports
Cooling air inlets
Cooling air outlets
Ice lights
Static wicks
Electrical ground power receptacle

H. Main Landing gear RH

Tires (inflation/damage)
Wheels
Brake/shuttle valve/hoses
Brake discs A
Strut (extension/leaks)
Landing gear doors
Torque links
Actuating cylinders, linkages, and hoses
Uplocks/linkage
Gear uplock support
Downlock assembly
Hydraulic lines
Electrical wiring and switches
Fire extinguisher bottle (specified pressure)
Fuel pump seal drain
Emergency landing cables, pulleys, and brackets
Fuel and hydraulic lines (leaks, chafing, security)
Strut door switch D

I. Wing RH

Exterior
Skins
Single point refueling cap
Leading edge
Navigation light
Anti-collision light/strobe light/recognition light
Static wicks
Fuel filler cap
Aileron and bonding wires

A See Special Inspection, 5-10-07.
D S/N 370-1 and -7; 380-1 and subs

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150 HOUR MAINTENANCE CHECK (Continued)

I. Wing RH (Cont'd)
- Hinges and pins (bolts)
- Bellcrank, sector, cables and pulleys
- Flap/fairings/tracks/screwjacks/flex shafts
- Fuel drains
- Required markings (legibility)
- Wing tip, fuel vent outlet
- Wing slats, tracks, bearings
- Deicer boots (if installed)
- Stall warning vanes

J. Power Plant RH
- Thrust reverser
  - Exhaust nozzle (cracks)
  - Doors (latches, hinge points)
- Actuator (hydraulic fluid leaks) C
- Aft fairing, skin/cracks/corrosion
- Aft fairing access doors, skin/cracks/corrosion
- Engine cowling access doors and louvers
- Engine drain manifold
- Generator cooling air inlets
- Generator cooling air exhaust (carbon brush dust)
- Electrical wiring
- Plumbing lines
- Inlet area
  - Inlet duct
  - Fan inlet D
  - Variable inlet guide vanes D
  - Free rotation of compressor and blade condition
  - Bullet nose
  - Inlet pressure port and sense probe
  - Inlet temperature sensor C
  - Splitter vanes C
  - Inlet guide vanes C
- Exhaust area
  - Tailpipe and cone
  - Struts
  - Turbine blades for free rotation and condition C
  - Fan for free rotation and blade condition
  - EGT thermocouples and EPR probes D

K. Fuselage Aft RH
- Access doors
- Exterior
- Skins
- Cabin windows
- Fuel filler cap
- Pylon
- Pylon access plates
- Pylon drains
- Fuel drains
- APU inlet and exhaust duct (if installed) D
- Fuel heat exhaust duct D
- Pylon to nacelle seals D

L. Empennage
- Ram air inlet (obstructions)
- EPR pitot masts
- Bird stopper door (if installed)
- Antennas
- Deicer boots (if installed) D
- Vertical fin
- Static wicks
- Rotating beacon
- Horizontal stabilizer (RH)
- Elevator (RH)
- Rudder
- Trim tab
- Navigation light
- Anti-collision light/strobe light
- Tail cone
- Flame arrester
- Fuel vent
- Lightning diverters
- Horizontal stabilizer (LH)
- Elevator (LH)

M. Cockpit
- Standby compass (correction card)
- Center pedestal
- Oxygen regulators, hoses, masks
- Instrument lines, wire bundles, and flex hoses (fwd of instrument panel)
- Seats
- Safety belts and harness/inertial reel (pull test seat belts if condition is questionable)
- Instruments (slippage marks and range marks)
- Switch panels
- Circuit breakers
- Lighting
  - Thunderstorm
  - Emergency/map
  - Floods, utility, caution warning
- Flight controls and engine controls (operation)
- Windshield and windows
- Gust lock
- Flashlights
- Portable oxygen bottle (1800 psi and weight date)
- Fire extinguisher bottle (weight/date)
- Placards (legibility)
- Parking brake

C S/N 370-2 thru -6, -8 and -9
D S/N 370-1 and -7; 380-1 and subs

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150 HOUR MAINTENANCE CHECK (Continued)

N. Entrance Way

Window dehydrators
Furnishings
Required markings (legibility)
Electrical equipment rack

O. Cabin

Cabin seats
Safety belts (pull test if condition is questionable)
Windows
Upholstery
Lighting
Emergency escape hatches
Oxygen bottle (1800 psi)
Portable fire extinguisher
Carpet
Air outlets
Lights and cabin signs
Emergency lighting
Oxygen masks stowed
Cleanliness (galley, lavatory)
Doors
Baggage restraints

10. Insure proper sevicing of hydraulic system and perform the following:
    A. Ground maintenance safety locks .... remove
    B. HYDRAULIC PUMP switch .....AUTO
    C. SPEED BRAKE switch ..... IN
    D. PARKING brake handle .....release
    E. WING FLAP handle .....UP
    F. HYDRAULIC PUMP switch ..... OFF & RESET
    G. HYDRAULIC AUX SYS switch .....ON
11. Slowly relieve normal system hydraulic accumulator pressure. Check that auxiliary system pressure does not follow.
12. Relieve auxiliary and thrust reverser system [C] hydraulic accumulator pressure. Check hydraulic reservoir for proper fluid level and pressure (minimum 35 psi).
13. Check the following hydraulic accumulators for specified air pressure.
    A. Normal ..... 1500 (±50 psi)
    B. Auxiliary ..... 1500 (±50 psi)
    C. Thrust Reversers .....900 (±50 psi) [C]
14. Perform the following:
    A. HYDRAULIC AUX SYS switch ..... OFF
    B. HYDRAULIC PUMP switch .....AUTO
    C. Check pump operating time from start through accumulator build-up to pump shut-off. Maximum 25 seconds. This check is performed with external 28 VDC power applied to the airplane.
15. Perform the following speed brake test:

WARNING

Do not enter speed brake area when electrical and hydraulic power is applied. Anyone caught by a moving speed brake panel may be seriously injured or killed.

    A. Electrical Power.....ON
       HYDRAULIC PUMP switch .....ON
    B. Extend Speed Brake
    C. HYDRAULIC PUMP switch ..... OFF
    D. Bleed pressure from normal and auxiliary hydraulic accumulators to simulate hydraulic failure.
    E. Speed Brake Switch..... OFF (neutral position of switch)
    F. Speed Brake Dump Switch ..... Dump

NOTE

The speed brake dump switch is spring loaded to the NORM position.

    G. While holding speed brake dump switch in the "dump" position, an assistant should be able to slowly push the speed brake toward the closed position.
    H. Clear all personnel from speed brake area.
    I. Allow speed brake dump switch to return to the NORM position.
    J. HYDRAULIC PUMP switch .....AUTO
    K. When hydraulic pressure reaches normal range — retract speed brake.
16. Perform the following:
    A. Open LH inspection plate (snap open type plate at bottom of FS 298 +21).
    B. Monitor indicator pointer on fire extinguisher direction valve when the RH FIRE PULL handle is pulled.
    C. Check that pointer moves and points to the LH engine (blocked engine).
    D. Push in RH FIRE PULL handle. Pointer returns to normal position (indicator points to RH engine).
17. Electrical power ..... off

[C] S/N 370-1 thru -6, -8, and -9.

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150 HOUR MAINTENANCE CHECK (Continued)

18. Circuit breakers:
    LH and RH ENG-MASTER ..... in
    LH and RH VIBRATORS ..... in
    LH and RH FUEL-BOOST PUMP ..... in
    EQUIP GRD BLO-SECOND CONT ..... in
19. Drain the following: C
    Fuselage water drains
    Fuel sump drains
    Pitot system
    Static system
    EPR system
20. Close all access doors if applicable.
21. Check all flight controls for freedom of movement, binding, and looseness.
22. Check bleed air shutoff valves for proper operation. This may be performed during engine run by selecting CABIN AIR for each engine and noting air flow at the cabin air outlets.
23. Check emergency escape hatch(es) for proper operation.
24. Wheel well dish pans ..... install
25. Airplane ..... service as required
26. Airplane ..... secure as desired

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UNSCHEDULED MAINTENANCE CHECKS

GENERAL INFORMATION

GENERAL

All maintenance checks and inspections on the airplane, its systems, and units that are dictated by special or unusual conditions which are not related to the time limits specified under Scheduled Maintenance Checks are included here.

BLOWN TIRES

After a blowout or a thrown tread of a main landing gear tire, inspect the wheel well area as follows:

CAUTION

Particular attention should be given to any bending of the actuator piston rod, as a slight bend places an asymmetrical load on the rod which will cause additional bending through repeated usage and cause the landing gear to jam during retraction. If bending is suspected, a retraction test should be accomplished.

1. Inspect main landing gear actuating cylinder for bent piston rod and/or damage to cylinder and attaching parts.
2. Inspect main gear uplock assembly for bent or broken springs, levers, arms, and/or hooks.
3. Inspect main gear strut doors for cracks, tears, and/or damage to attaching hardware.
4. Inspect all electrical wiring, fuel lines, and hydraulic lines in wheel well area for damage.
5. Inspect wing flaps lower skin for damage due to thrown tread.
6. Inspect brakes for abnormal wear or signs of binding.
7. Perform an operational test of the anti-skid system.
8. Inspect wheels, per ABS manual AP-426, for any resulting damage.
9. Inspect flaps and trailing edges for damage in the event of thrown tire tread.

HEAVY RAIN/MOISTURE/LIGHTNING STRIKE/"P" STATIC

Whenever the airplane has encountered rain or other heavy concentration of moisture, operate drain fitting in lower LH fuselage skin at FS 143 to permit removal of any water accumulation. Ensure that drain fitting is returned to closed position after water removal.

Whenever it is known or suspected that the airplane has sustained a lightning strike, perform the following:

1. Inspect the upper centerline of the radome for evidence of burn marks.
2. Check resistance of radome lightning diverter strip.
3. Inspect two lightning diverter rods for general condition, evidence of burned areas and security.
4. Check static dischargers subsequent to a flight through heavy rain per chapter 23.

INSTRUMENT REMOVAL/INSTALLATION

Whenever any installation or reinstallation of instrument system components in the pilot's or co-pilot's instrument panels has been performed, an inspection will be made to determine that adequate clearance exists between the instrument panel installations (plumbing, wiring, etc.) and the elevator control column throughout its full operating range. Special attention should be given to wire bundle installations in this area that may shift or dislodge because of airplane G-loading.

FIRE EXTINGUISHER SYSTEM DISCHARGE/PURGE

Whenever the fire extinguishing system has been discharged, the system should be purged with nitrogen. (Refer to chapter 26.)

STARTER/GENERATOR REPLACEMENT

DELETED

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GENERAL INFORMATION (Continued)

WING STRUCTURE

NOTES

The following is not required if:

- ***The fuselage/wing attach fittings have been replaced or if the fittings have been removed and inspected using the dye penetrant procedure and the adjacent structure inspected and found to be airworthy.***
- ***When wing attach fittings are removed, the fittings must be installed with the use of the jig fixture.***

Whenever the airplane wing has been struck with sufficient force to cause material damage of consequence, a special inspection must be accomplished to ascertain whether structural damage has also occurred in the wing-to-fuselage forward attach area. (See Figure 1 for procedures.) The wing-to-fuselage attach area should be reinspected every 200 hours up to a 600 hour total after the wing damage has occurred.

ENGINE OVERSPEED/OVERTEMP/HUNG START

If the engine has been subjected to an overspeed or overtemperature condition, a "hung" start, or shutdown from an aborted start in which residual fuel is burned inside the engine, inspect interior of thrust reverser assembly for hot spots, distortion, localized overheating, and presence of foreign material.

HARD LANDING

If the airplane has been subjected to a hard landing, it is recommended that the following inspection be performed:

1. Check that all landing gear downlocks are fully engaged.
2. Deplete the hydraulic system pressure.
3. Place the airplane on jacks.

NOTE

The landing gear will require partial retraction to accomplish step 4 thru 6. Refer to chapter 32 for inspection of landing gear.

4. Disconnect the main landing gear actuating cylinders from the struts and ensure that the hydraulic cylinder actuator rods are not bent. Reconnect and safety the actuator rod ends.
5. Perform an operational check of the landing gear.
   A. Visually check for proper chassis alignment.
   B. Check that strut cylinder and linkage binding does not occur.
   C. Check for proper landing gear indication.
6. Inspect main and nose landing gear attach points. Check nose landing gear drag brace attach points.
7. Inspect forward and aft wing to fuselage attach fittings and surrounding structure.
8. Check APU (if installed) mounting points.
9. Inspect fuselage skins and frames at FS 307.25, 333.625, and 357 in the area of the engine pylon attach points.

NOTE

If failure of any type is found at FS 333.625, internal inspection must be made of pylon-to-nacelle attachments at FS 318.586 and 333.625. (Attachments are between inner and outer skins in nacelle structure.) Refer to the Structural Repair Manual (NA-66-1032), for illustrations of inspection areas.

10. If any damage is detected at wing-to-fuselage area, disassemble MLG struts and inspect the metering pin and other related parts for distortion.
11. Inspect wheels, per ABS Manual AP-426, for any resulting or suspected damage.

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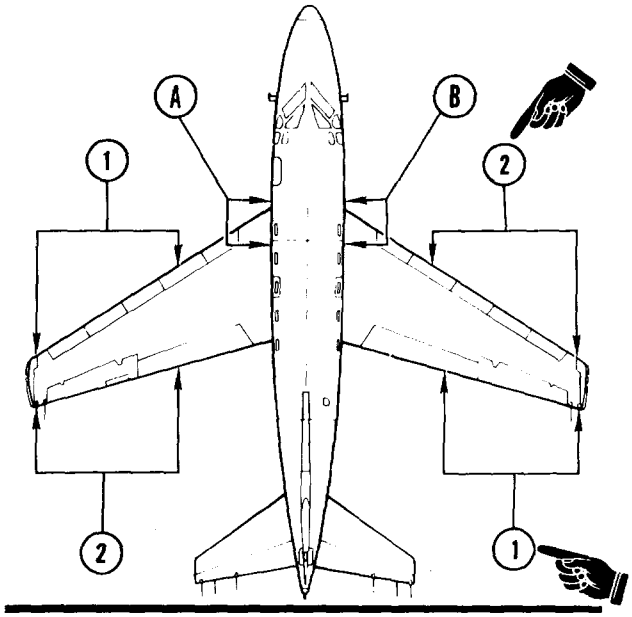

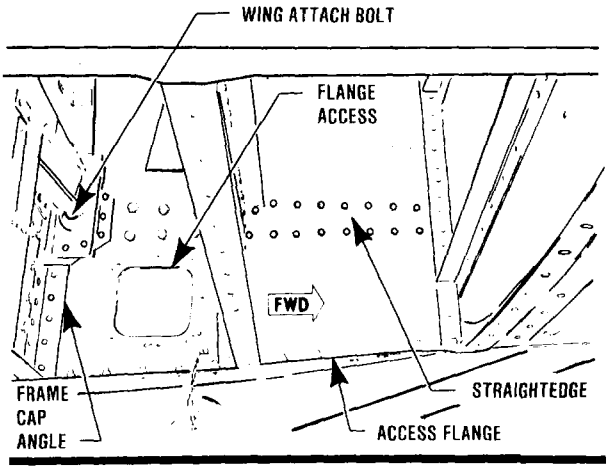

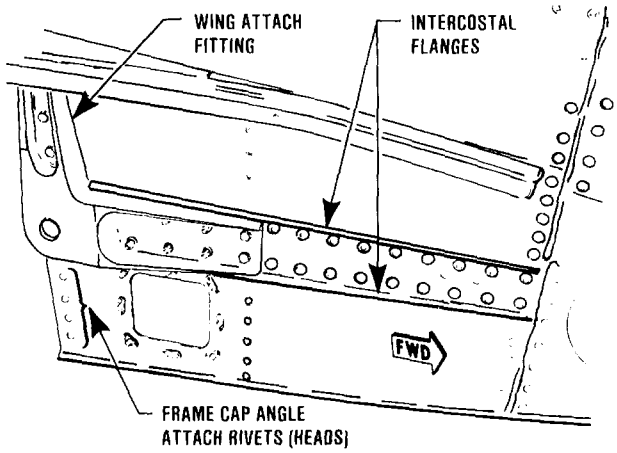

VIEW WITH WING REMOVED

Figure 1 — Wing Inspection

INSPECTION PROCEDURES

If impact damage has occurred in areas designated (1), inspection of the left wing-to-fuselage forward attach area (A) is of primary importance, and inspection of attach area (B) is of secondary importance. If impact damage has occurred in area designated (2), inspection of the right wing-to-fuselage forward attach area (B) is of primary importance, and inspection of attach area (A) is of secondary importance. Inspect areas, using the following criteria and procedures:

1 Remove access panels 22 and 32 from lower fuselage at wing forward attach area.

2 In forward outbd bay of access, lay a straight edge along rivet pattern that secures wing attach fitting intercostal to fuselage skin. Check for obvious bows and deformation in skin and for pulled attaching rivets. Check lower outbd access area flange for buckling and frame-to-fuselage attaching clips and support structure in access for pulled and cocked rivets.

3 In rear outbd corner of bay at wing attach bolt, check frame cap angle at station 206 +17 for cracks or deformation. Check for sheared, cocked, or pulled rivets securing frame cap angle to fuselage skin and frame web.

4 Remove small flanged access panel (secured with eight screws) from rear bay of access.

5 Insert a small inspection mirror through access and, using a bright flashlight, check flanges of intercostal for evidence of bows or buckling. The intercostal flanges should be straight within 1/32 inch. Check intercostal for cocked or pulled attach fasteners. Observe any deformation in paint pattern around rivet heads securing intercostal.

6 Through small access and below wing attach fitting, check frame cap angle for sheared, cocked, or pulled attaching rivets.

7 Repeat steps **2** through **6** for opposite attach area.

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CHAPTER 06

DIMENSIONS AND AREAS

CHAPTER 6

DIMENSIONS AND AREAS

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CHAPTER 6

LIST OF EFFECTIVE PAGES

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		3	Jul 6/73
		4	Jul 6/73
		5*	May 15/81
		6	Jul 6/73
		7*	May 15/81
		7A*	May 15/81
	Added	7B*	May 15/81
		8	Jul 6/73
		9	Jul 6/73
		10	Jul 6/73
		11*	May 15/81
		12	Jul 6/73
		13	Jul 6/73
		14	Jul 6/73
		15*	May 15/81

* The asterisk indicates pages revised, added, or deleted by the current revision.

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DIMENSIONS AND AREAS - DESCRIPTION

1. General

The Rockwell International Sabreliner is a high-speed, pressurized, passenger transport airplane, powered by two engines mounted aft of the wing on both sides of the fuselage in pod-type nacelles.

Replacement engines for the NA-265-80 airplanes shall have a "B" suffix letter following the engine serial number on the data plate.

The fuselage is divided into six separate compartments: a nose equipment compartment, a crew compartment, an entrance compartment, a passenger compartment, a fuel cell compartment, and an aft equipment compartment. The entrance to the airplane is on the left side of the fuselage, just aft of the crew compartment. The entrance door swings out and down to provide a stairway into the airplane. The crew compartment provides for a crew of two: pilot and copilot. The passenger compartment provides for a maximum of 10 passengers. Space is provided in the entrance compartment for baggage and for installation of a galley and lavatory. The aft equipment compartment contains components of the electrical, hydraulic, environmental, and flight control systems.

Three fuel tanks, one bladder-type cell in the fuselage and an integral fuel cell in each wing, contain a total usable capacity of 1106 US gallons. The elevators, ailerons, and rudder are conventionally operated and manually controlled through the use of cables, bell cranks, and push rods. Electrically operated flaps are provided on the inboard trailing edge of each wing, and aerodynamically operated slats are provided on the leading edge of each wing.

A hydraulically actuated speed brake, which can be operated at any airspeed, is on the bottom of the fuselage.

The tricycle landing gear has dual nose wheels and dual main gear wheels. The main gear retracts inboard into the fuselage, and the nose gear retracts forward into the nose section. The nose gear is steered hydraulically.

General dimensions are shown in figure 6-00-1. Stations are shown in figure 6-00-2. Access provisions are given in figure 6-00-3 or 6-00-4.

2. General Dimensions

A.	Span	44 ft 5 in.
B.	Length, maximum	47 ft 0 in.
C.	Height of vertical fin tip from ground	17 ft 3 in.
D.	Design maximum ramp and taxi gross weight.	21,700 lb* 23,500 lb†
E.	Design maximum landing gross weight.	18,500 lb* 22,000 lb†

*NA-370-2 through -6, -8 and -9
†NA-380-1 and later

3. Wing

A.	Type	Low
B.	Airfoil at root	64A series (NAA-modified) 11.30 pct thickness - chord ratio
C.	Airroil at tip	64A series (NAA-modified) 9.36 pct thickness - chord ratio
D.	Mean aerodynamic chord	8.38 ft
E.	Dihedral	3.14 deg
F.	Aspect ratio	5.77
G.	Area	342.05 sq ft
H.	Aileron area (total aft of hinge line)	16.42 sq ft
I.	Wing flap area (total)	40.26 sq ft
J.	Wing slat area (total)	36.34 sq ft

4. Fuselage

A.	Maximum breadth, outside	69.0 in.
B.	Total cabin volume	660 cu ft
C.	Speed brake area (total)	7.64 sq ft
D.	Speed brake deflection	50 deg

5. Landing Gear

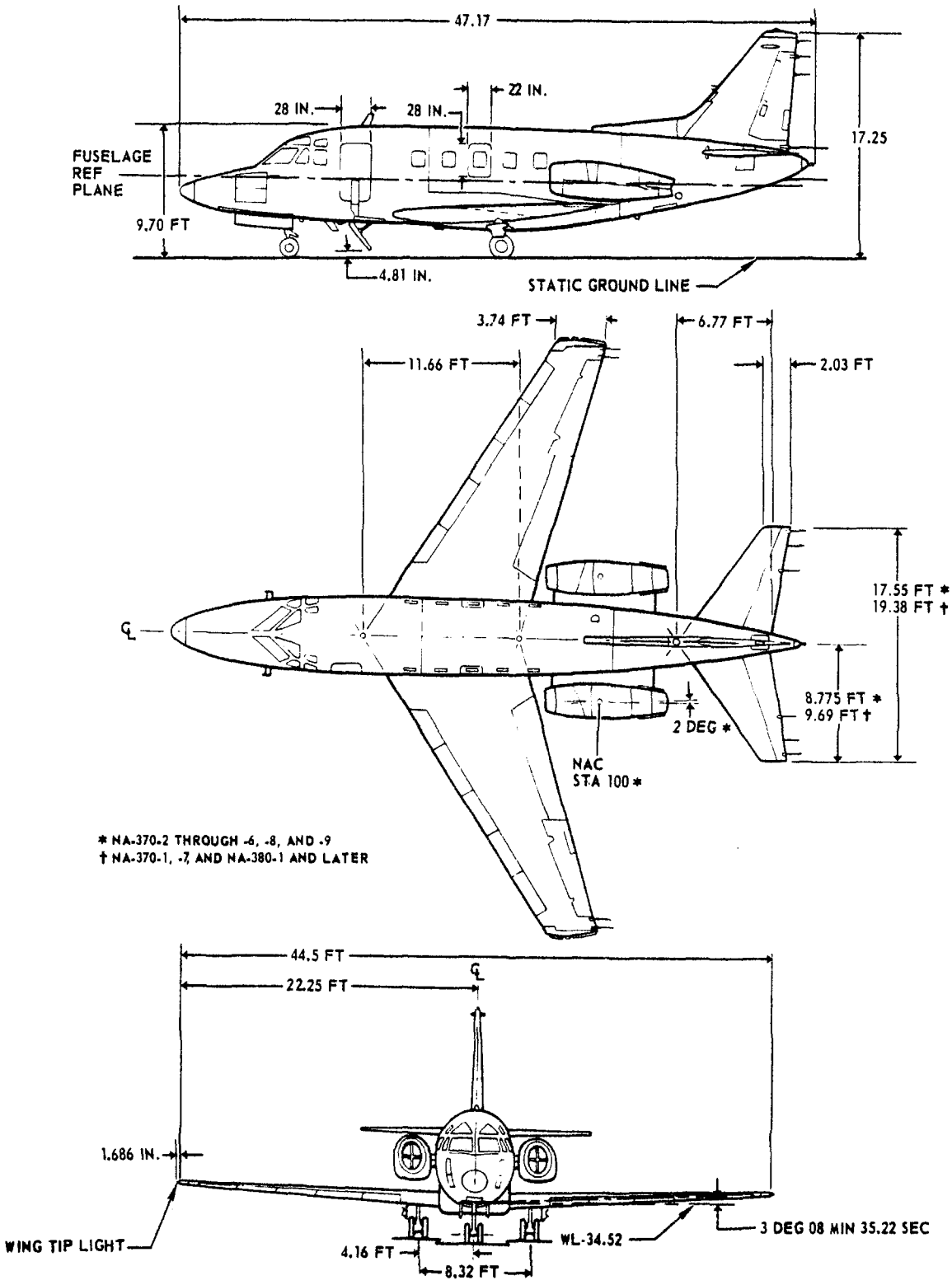
A.	Tread	8.33 ft
B.	Wheel base	14.50 ft
C.	Main gear tire size (tubeless)	22 x 5.75-12 10-ply rating
D.	Nose gear tire size (tubeless)	18 x 4.4, 6-ply, 18 x 4.4, 10-ply rating chine (deflector) tire*

*Use on runways with standing water.

6. Tail

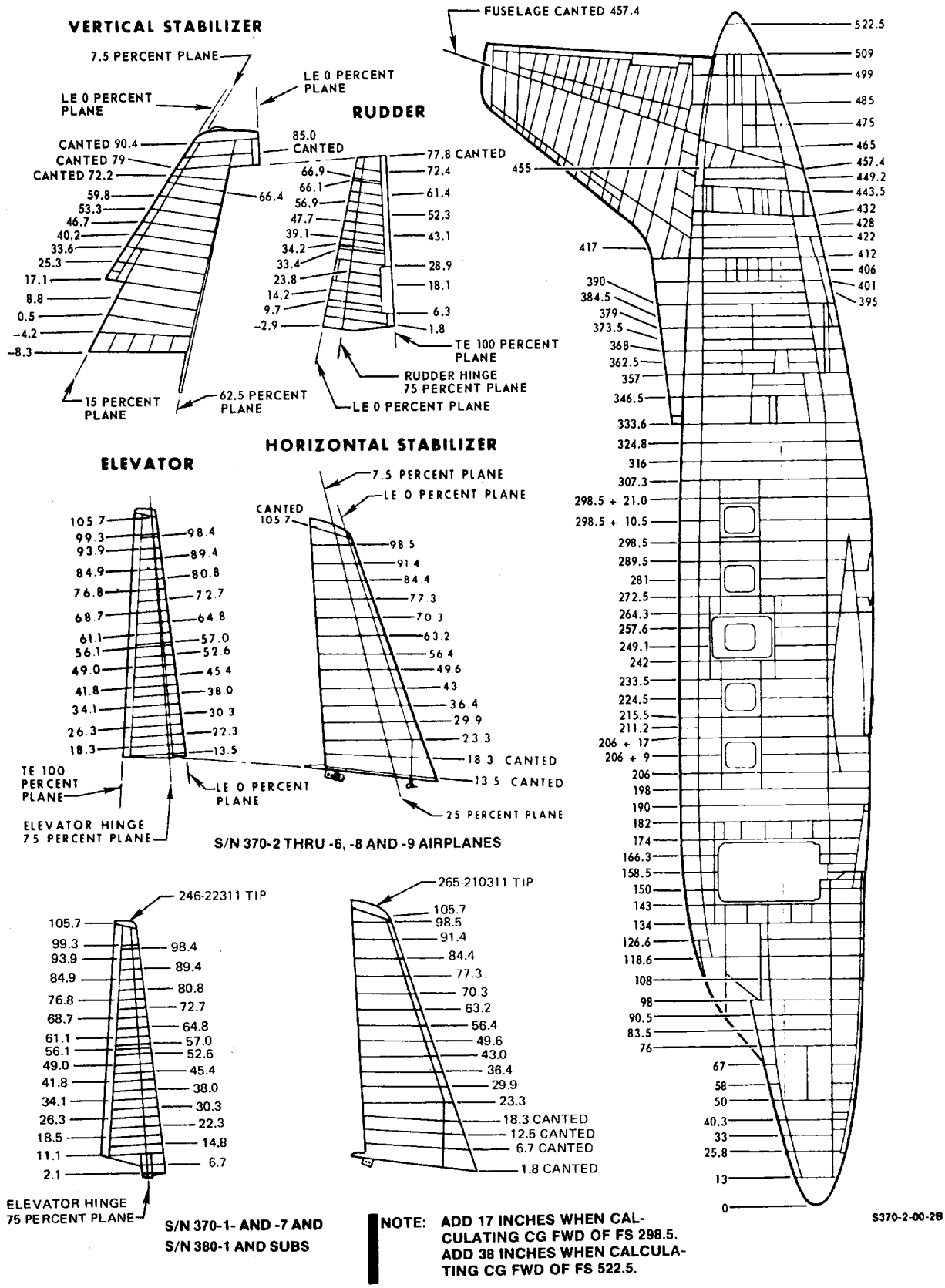
A.	Horizontal tail span	17.55 ft* 19.38 ft†
B.	Horizontal stabilizer area (total)	77 sq ft* 90.076 sq ft†
C.	Elevation area (total aft of hinge line) . .	16.52 sq ft* 19.425 sq ft†
D.	Vertical stabilizer area (total)	52.24 sq ft
E.	Rudder area (total aft of hinge line). . . .	9.75 sq ft

*NA-370-2 through -6, -8 and -9
†NA-370-1, -7 and NA-380-1 and Later



S370.2.00-1A

General Dimensions
Figure 6-00-1



Stations Diagram
Figure 6-00-2 (Sheet 1 of 4)

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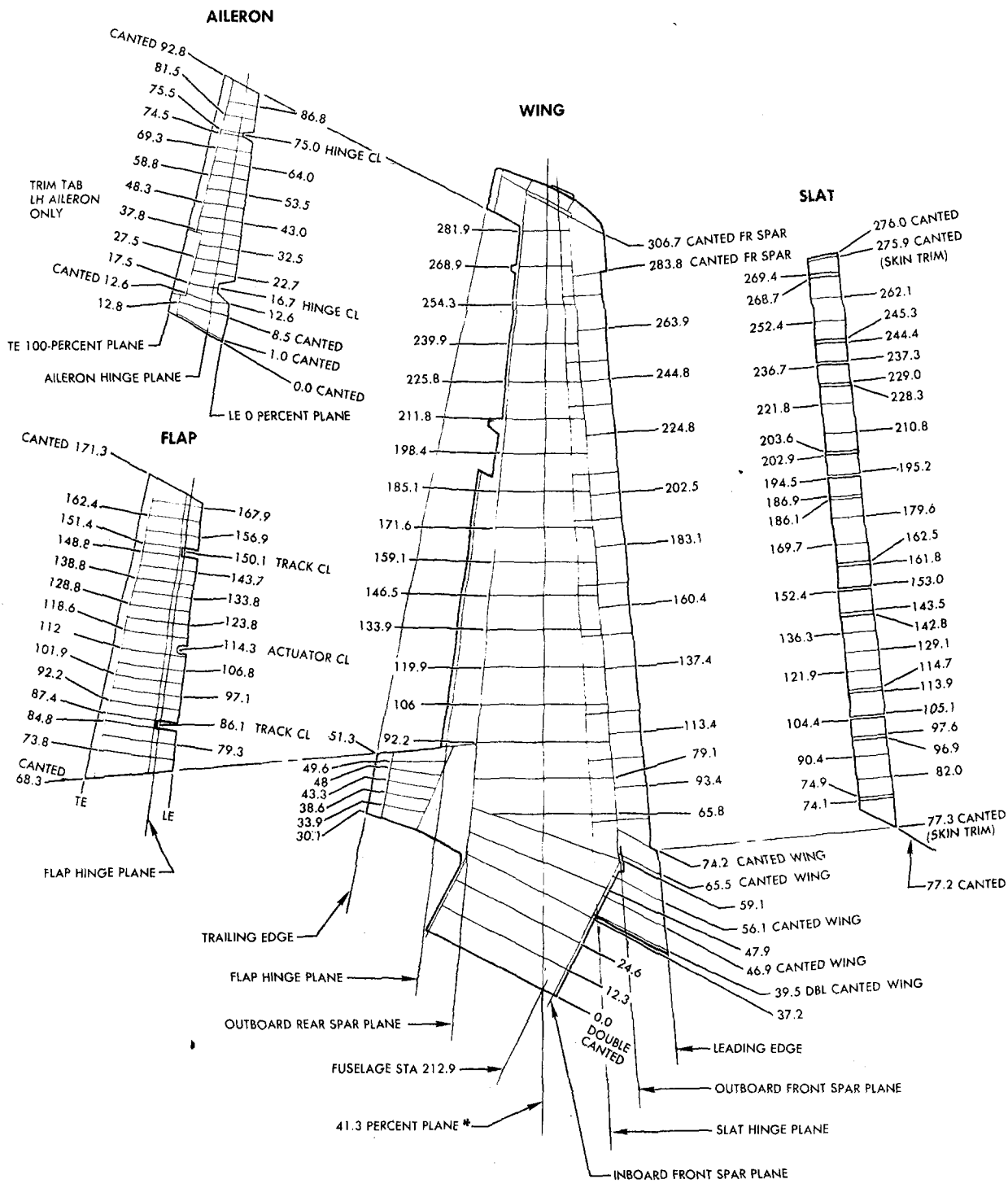
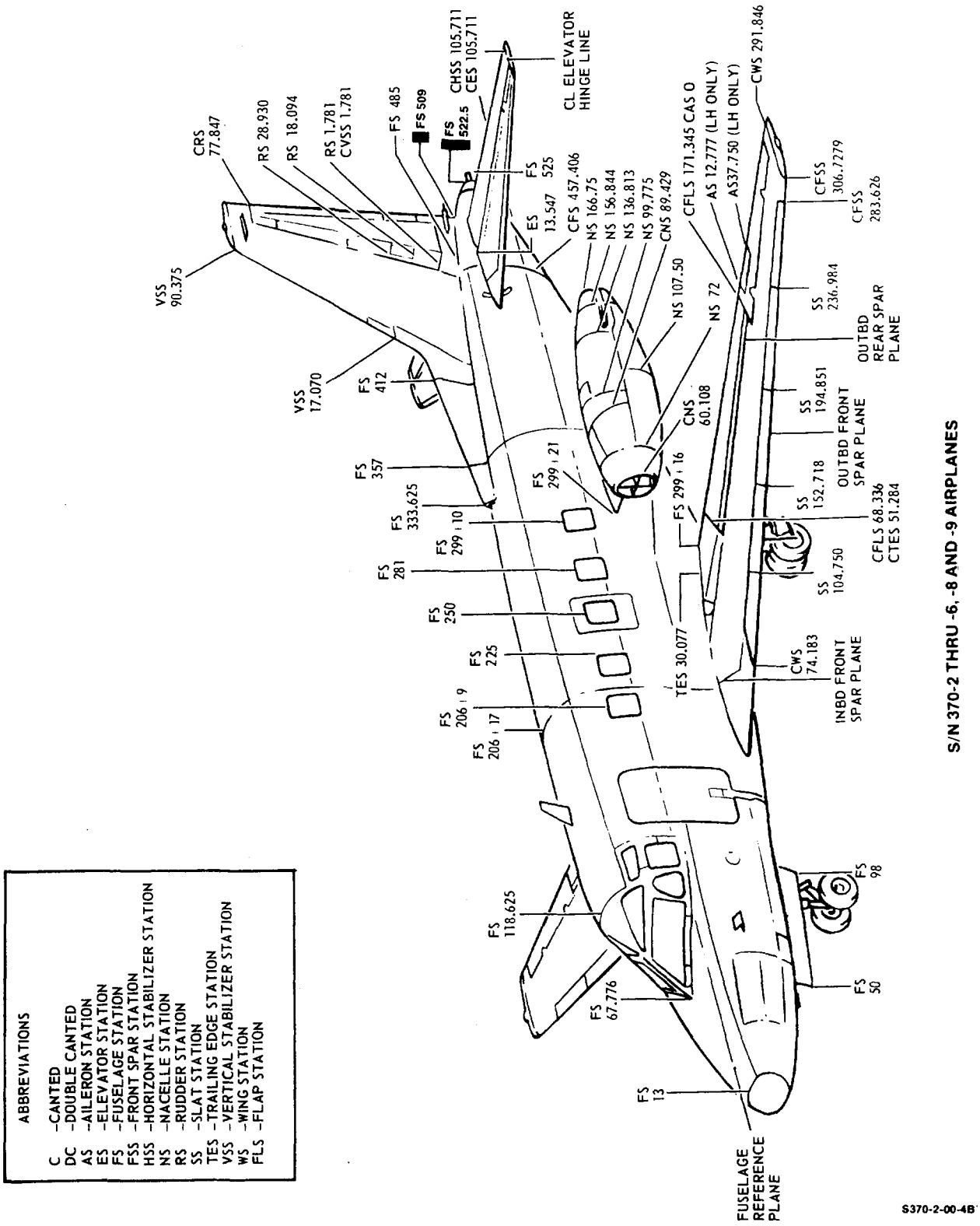


*ALL WING STATIONS ARE PERPENDICULAR TO 41.3 PERCENT ELEMENT PLANE

S370-2-00-3

Stations Diagram
Figure 6-00-2 (Sheet 2 of 4)



Stations Diagram
Figure 6-00-2 (Sheet 3 of 4)

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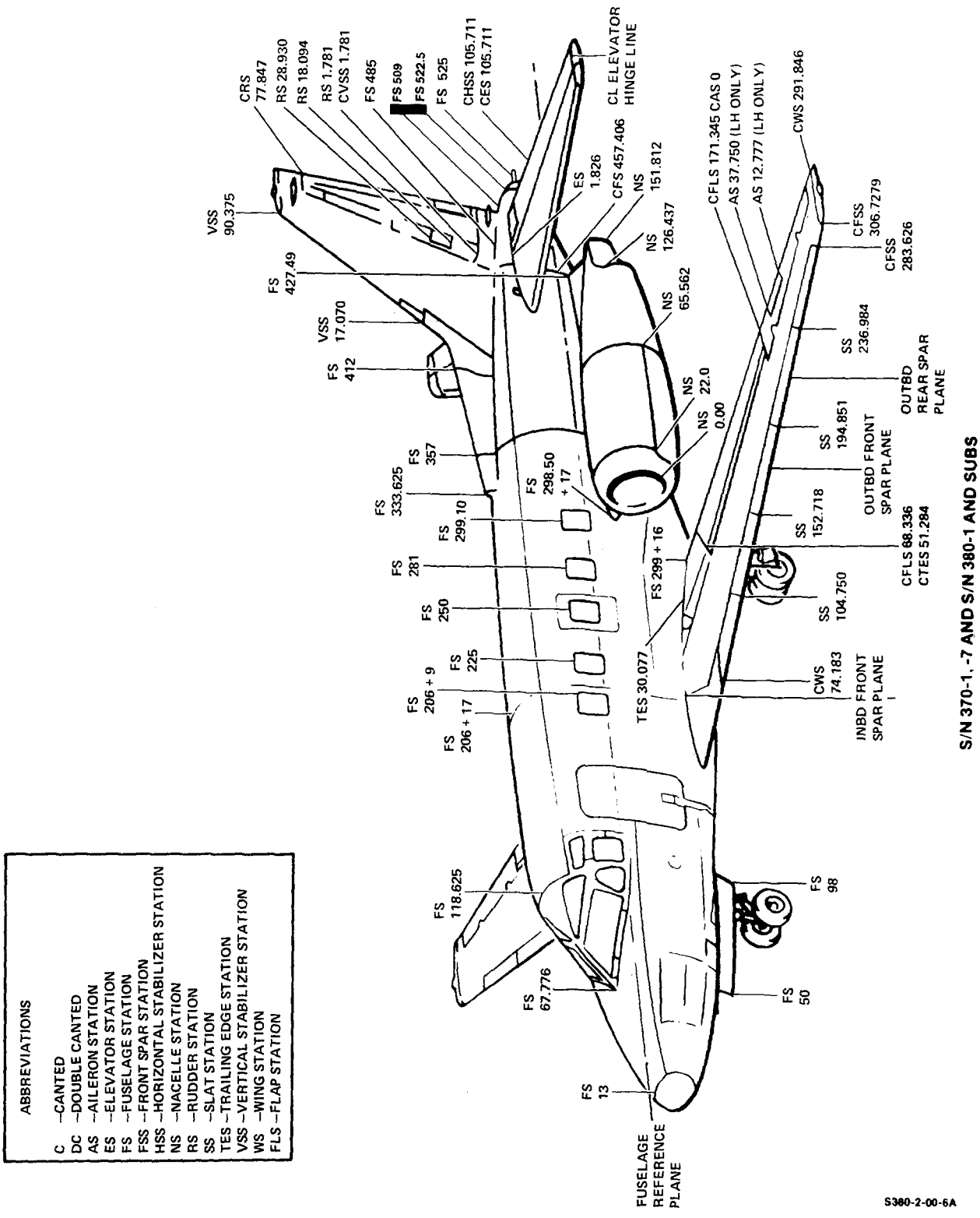

Stations Diagram
Figure 6-00-2 (Sheet 4 of 4)

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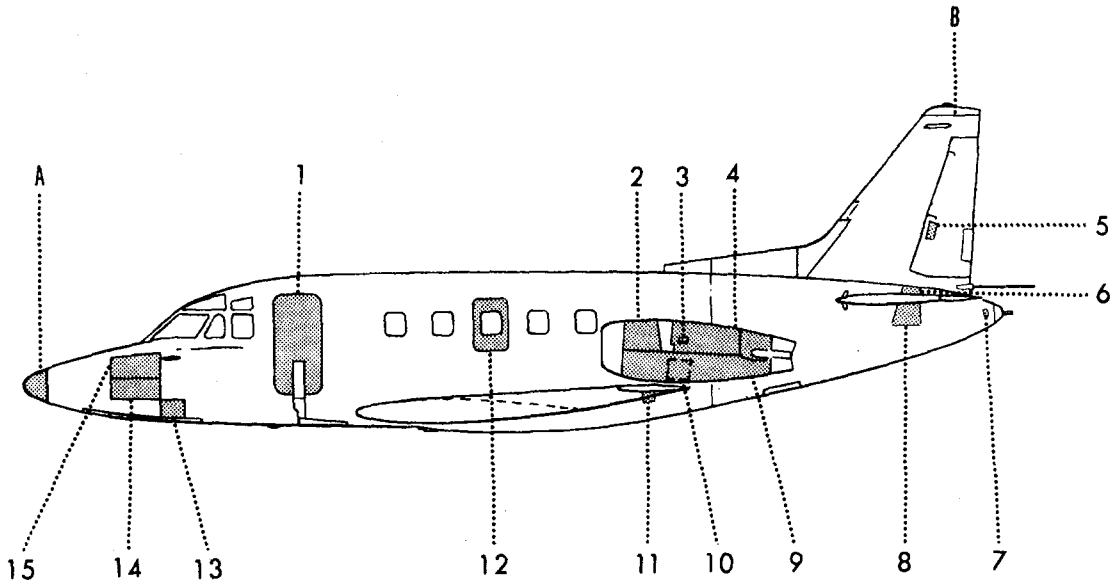


A RADOME

B VERTICAL STABILIZER TIP

1 ENTRANCE DOOR

2 ENGINE ACCESS

3 FIRE DETECTOR DISCONNECT

4 ENGINE ACCESS

5 RUDDER TRIM TAB ACTUATOR ACCESS

6,8 FLIGHT CONTROL ACCESS
FUEL VENT DISCONNECT
RUDDER RIGGING PROTRACTOR
ELEVATOR RIGGING PROTRACTOR

7 FUEL JETTISON AND VENT LINE DISCONNECTS

9 ENGINE ACCESS

10 ENGINE LINE ACCESS

11 DC ELECTRICAL POWER RECEPTACLES

12 EMERGENCY ESCAPE HATCH

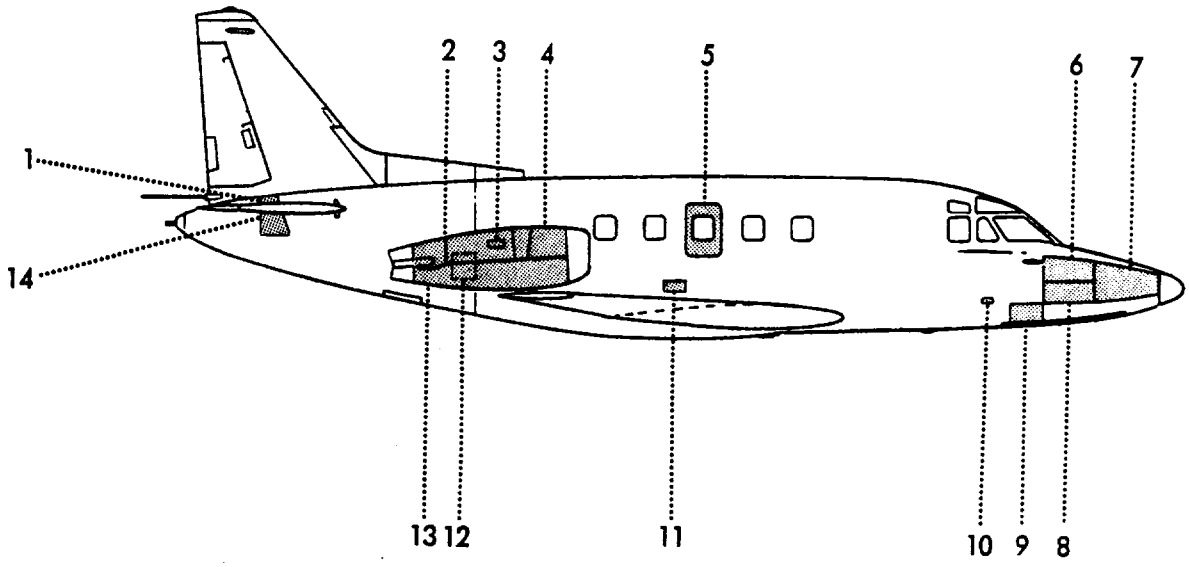
13 FLIGHT CONTROL ACCESS
CABIN PRESSURE SAFETY AND DUMP VALVE

14 INVERTER

15 COMMUNICATIONS AND NAVIGATION EQUIPMENT COMPARTMENT

5E70-2-00-5

Access Provisions - NA-370-2 Through -6, -8 and -9
Figure 6-00-3 (Sheet 1 of 4)



1 FLIGHT CONTROL ACCESS

2 ENGINE ACCESS

3 FIRE DETECTOR DISCONNECT

4 ENGINE LINE ACCESS

5 EMERGENCY ESCAPE HATCH

6 NOSE EQUIPMENT COMPARTMENT

7 COMMUNICATIONS AND NAVIGATION EQUIPMENT COMPARTMENT

8 INVERTERS

9 FLIGHT CONTROL RIG PIN ACCESS
CABIN PRESSURE REGULATOR

10 RIG PIN ACCESS

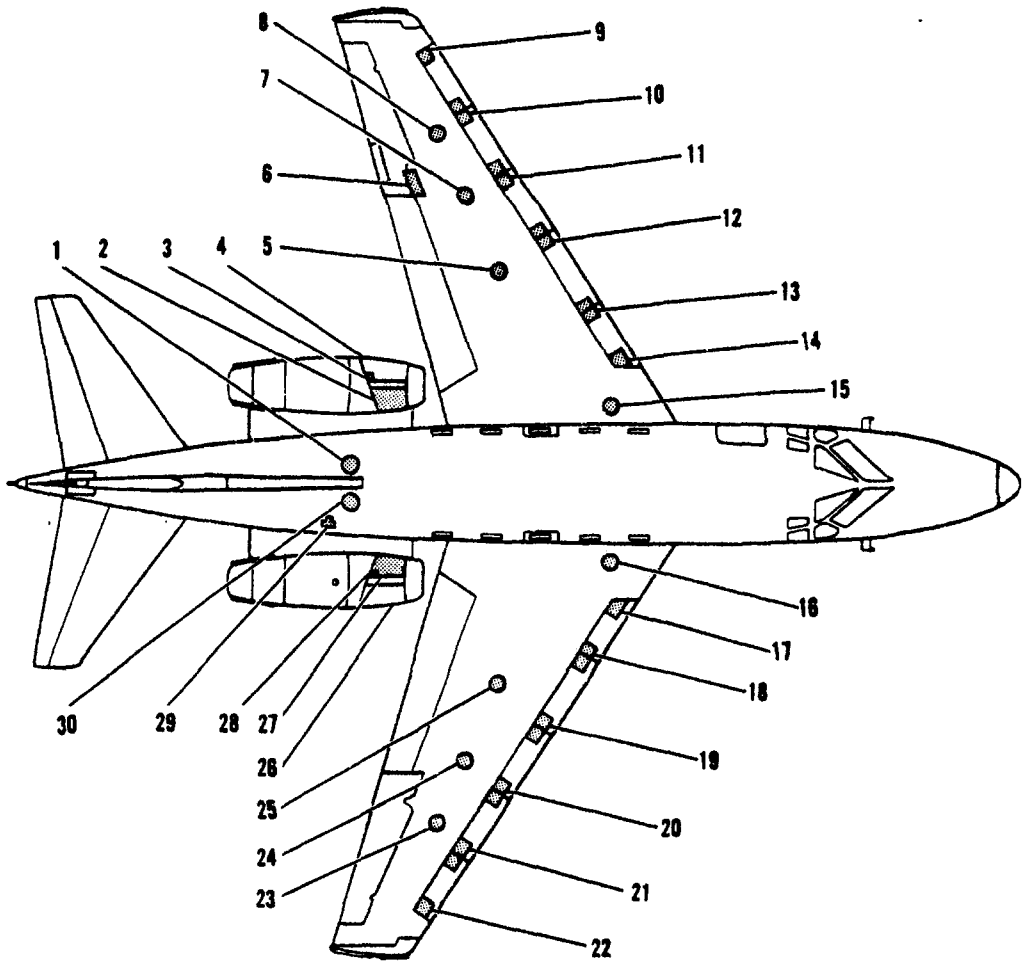
11 HYDRAULIC LINE ACCESS
FLIGHT CONTROL BELL CRANK ACCESS

12 ENGINE LINE ACCESS

13 ENGINE ACCESS

14 FLIGHT CONTROL ACCESS

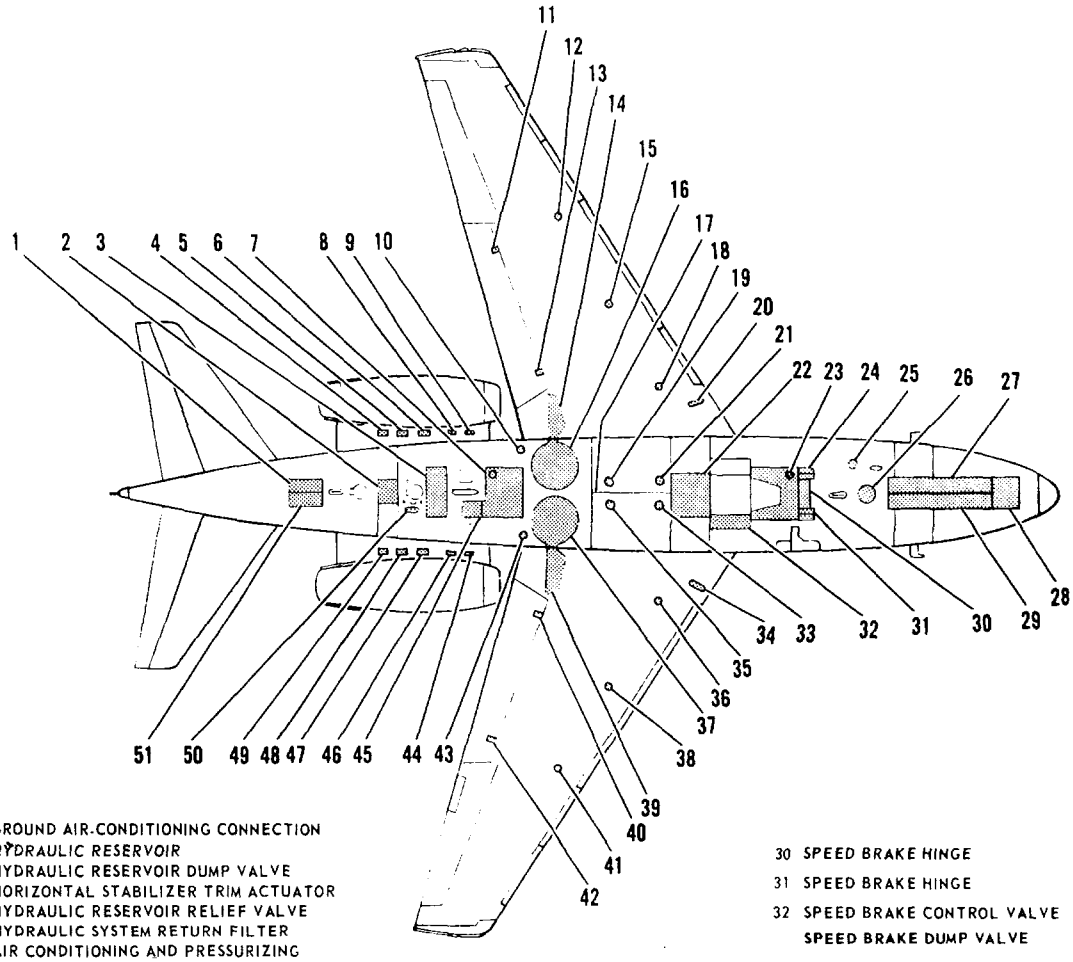
5370-2-00-6



1 FUSELAGE FUEL LEVEL TANK UNIT
2 ENGINE ACCESS
3 LEFT ENGINE OIL TANK FILLER
4 LEFT ENGINE
5 LEFT WING FUEL LEVEL TANK UNIT
6 AILERON TRIM TAB ACTUATOR
7 LEFT WING FUEL LEVEL TANK UNIT
8 LEFT WING FUEL LEVEL TANK UNIT
9 WING SLAT TRACK
10 WING SLAT TRACK
11 WING SLAT TRACK
12 WING SLAT TRACK
13 WING SLAT TRACK
14 WING SLAT TRACK
15 LEFT WING FUEL LEVEL TANK UNIT
16 RIGHT WING FUEL LEVEL TANK UNIT
17 WING SLAT TRACK
18 WING SLAT TRACK
19 WING SLAT TRACK
20 WING SLAT TRACK
21 WING SLAT TRACK
22 WING SLAT TRACK
23 RIGHT WING FUEL LEVEL TANK UNIT
24 RIGHT WING FUEL LEVEL TANK UNIT
25 RIGHT WING FUEL LEVEL TANK UNIT
26 RIGHT ENGINE
27 ENGINE ACCESS
28 RIGHT ENGINE OIL TANK FILLER
29 FUSELAGE FUEL TANK FILLER
30 FUSELAGE FUEL LEVEL TANK UNIT

S370-2-00-7

Access Provisions - NA-370-2 Through -6, -8 and -9
Figure 6-00-3 (Sheet 3 of 4)



1,51 GROUND AIR-CONDITIONING CONNECTION
HYDRAULIC RESERVOIR
HYDRAULIC RESERVOIR DUMP VALVE
HORIZONTAL STABILIZER TRIM ACTUATOR
HYDRAULIC RESERVOIR RELIEF VALVE
HYDRAULIC SYSTEM RETURN FILTER
AIR CONDITIONING AND PRESSURIZING SYSTEM UNITS

2 HYDRAULIC PRESSURE TRANSMITTERS

3 HYDRAULIC AUXILIARY SYSTEM SHUTOFF VALVE

4 ENGINE BLEED-AIR DUCT BELLOWS

5 ENGINE BLEED-AIR SHUTOFF VALVE

6 FUEL LINE CONNECTIONS
HYDRAULIC LINE CONNECTION

7 FUEL DRAIN

8 ENGINE PYLON ACCESS

9 *ENGINE PYLON ACCESS*

10 FUEL DRAIN

11 FLAP ADJUSTMENT ACCESS

12 FUEL DRAIN ACCESS

13 FLAP ADJUSTMENT ACCESS

14 MAIN LANDING GEAR STRUT DOOR
MAIN GEAR ACTUATING CYLINDERS

15 FUEL DRAIN ACCESS

16,37 MAIN GEAR ACTUATING CYLINDERS
MAIN LANDING GEAR INFLATABLE WHEEL SEAL
MAIN GEAR UPLOCK ACTUATING CYLINDERS
FUEL BOOST PUMPS
WING FLAP INTERCONNECT CABLE
BRAKE ANTISKID CONTROL VALVES

17 WING SPLICE JOINT ACCESS

18 FUEL DRAIN ACCESS

19 **FUEL DRAIN ACCESS**

20 FUEL CHECK VALVE
SINGLE POINT REFUELING EQUIPMENT

21 **FUEL DRAIN ACCESS**

22 **BATTERIES**

23 SPEED BRAKE
SPEED BRAKE LIMIT SWITCH

24 SPEED BRAKE HINGE

25 JACKING POINT ACCESS

26 HYDRAULIC LINE ACCESS

27,29 LANDING GEAR POSITION SELECTOR VALVE
ALTERNATE STATIC PORT
NOSE GEAR STEERING VALVES
BRAKE RESERVOIR
BRAKE CONTROL VALVES
PARKING BRAKE VALVE
NOSE GEAR ACTUATING CYLINDER
NOSE GEAR UPLOCK SWITCH

28 LANDING AND TAXI LIGHTS

30 SPEED BRAKE HINGE

31 SPEED BRAKE HINGE

32 SPEED BRAKE CONTROL VALVE
SPEED BRAKE DUMP VALVE

33 **FUEL DRAIN ACCESS**

34 FUEL CHECK VALVE

35 **FUEL DRAIN ACCESS**

36 *FUEL DRAIN ACCESS*

38 FUEL DRAIN ACCESS

39 MAIN LANDING GEAR STRUT DOOR
MAIN GEAR ACTUATING CYLINDERS

40 *FLAP ADJUSTMENT ACCESS*

41 FUEL DRAIN ACCESS

42 FLAP ADJUSTMENT ACCESS

43 FUEL DRAIN

44 ENGINE PYLON ACCESS

45 HYDRAULIC SYSTEM RELIEF VALVE
HYDRAULIC SYSTEM PRESSURE FILTERS
FUEL JETTISON VALVES
FUEL DRAIN VALVE
FIRE EXTINGUISHING BOTTLES

46 ENGINE PYLON ACCESS

47 FUEL LINE CONNECTIONS
HYDRAULIC LINE CONNECTION

48 ENGINE BLEED-AIR SHUTOFF VALVE

49 ENGINE BLEED-AIR DUCT BELLOWS

50 HYDRAULIC GROUND PRESSURE TEST

S370-2-00-8A

Access Provisions - S/N 370-2 Thru -6, -8 and -9
Figure 6-00-3 (Sheet 4 of 4)

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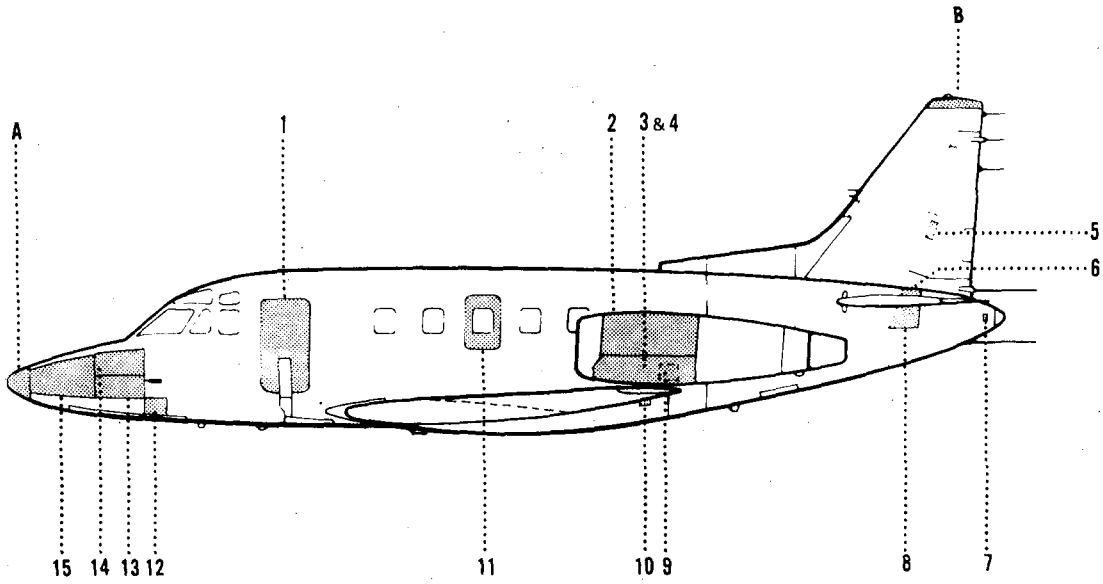


A RADOME

B VERTICAL STABILIZER TIP

1 ENTRANCE DOOR

2 ENGINE ACCESS (UPPER)

3 ENGINE ACCESS (LOWER INBOARD)

4 ENGINE ACCESS (LOWER OUTBOARD)

5 RUDDER TRIM TAB ACTUATOR ACCESS

6,8 FLIGHT CONTROL ACCESS

FUEL VENT DISCONNECT

RUDDER RIGGING PROTRACTOR

ELEVATOR RIGGING PROTRACTOR

7 FUEL VENT LINE DISCONNECTS

9 ENGINE LINE ACCESS

10 EXTERNAL HYDRAULIC POWER CONNECTIONS

11 EMERGENCY ESCAPE HATCH

12 FLIGHT CONTROL ACCESS

CABIN PRESSURE SAFETY AND DUMP VALVE

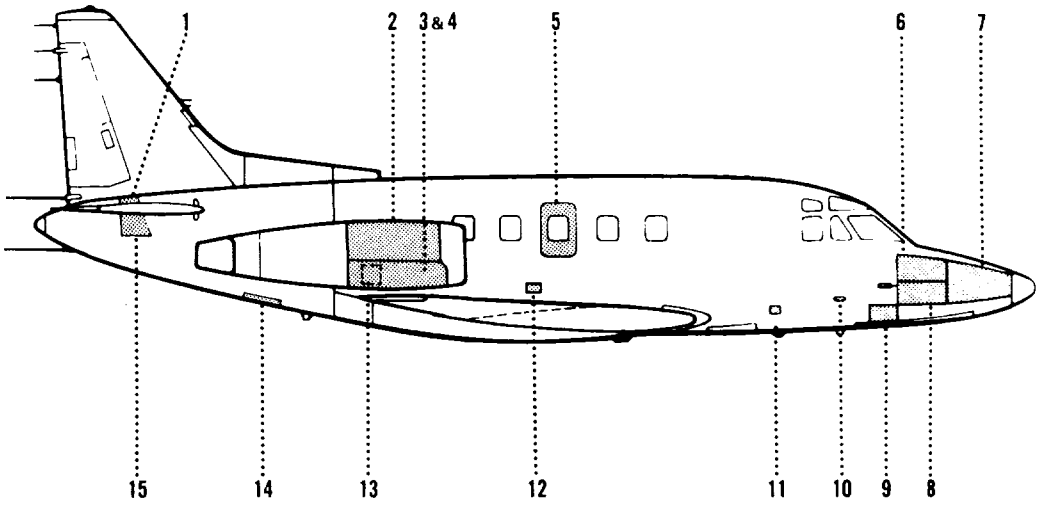
13 INVERTER

14 COMMUNICATIONS AND NAVIGATION EQUIPMENT COMPARTMENT

15 BATTERIES

S380-2-00-1

Access Provisions - NA-380-1 and Later
Figure 6-00-4 (Sheet 1 of 4)



1 FLIGHT CONTROL ACCESS

2 ENGINE ACCESS (UPPER)

3 ENGINE ACCESS (LOWER OUTBOARD)

4 ENGINE ACCESS (LOWER INBOARD)

5 EMERGENCY ESCAPE HATCH

6 NOSE EQUIPMENT COMPARTMENT

7 COMMUNICATIONS AND NAVIGATION EQUIPMENT COMPARTMENT

8 INVERTERS

9 FLIGHT CONTROL RIG PIN ACCESS
CABIN PRESSURE REGULATOR

10 RIG PIN ACCESS

11 DC ELECTRICAL POWER RECEPTACLES

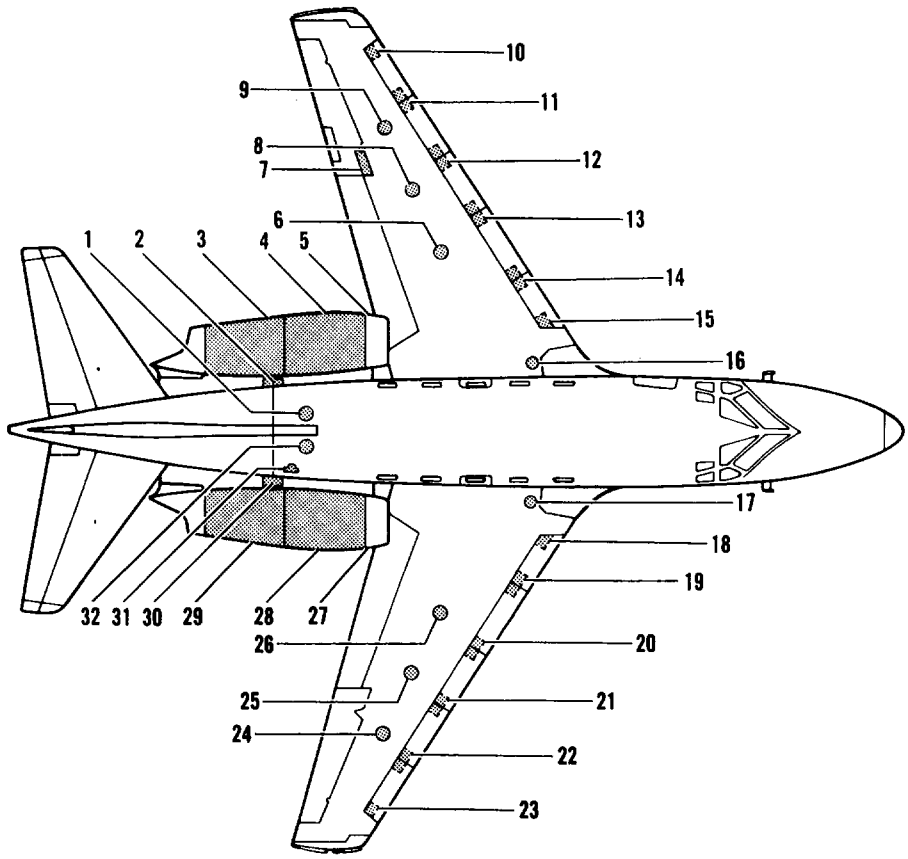
12 HYDRAULIC LINE ACCESS
FLIGHT CONTROL BELL CRANK ACCESS

13 ENGINE LINE ACCESS

14 AFT FUSELAGE ACCESS

15 FLIGHT CONTROL ACCESS

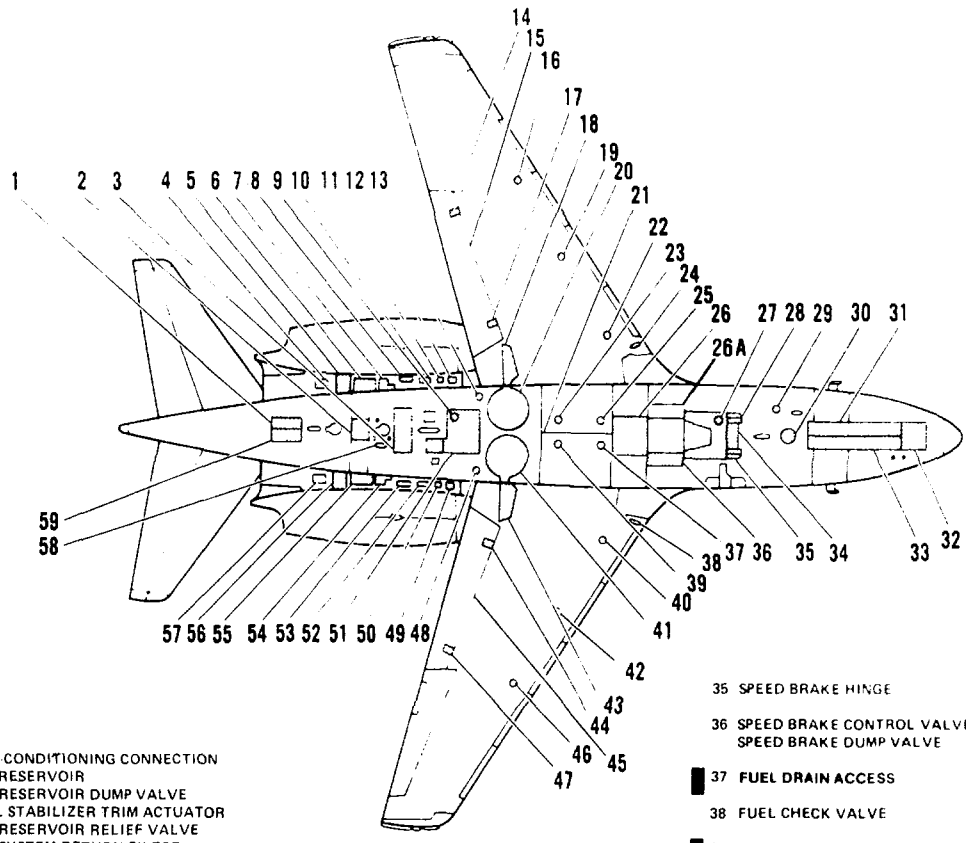
S380 2 00 2



1	FUSELAGE FUEL LEVEL TANK UNIT	17	RIGHT WING FUEL LEVEL TANK UNIT
2	ENGINE SUPPORT FITTING	18	WING SLAT TRACK
3	ENGINE AFT FAIRING	19	WING SLAT TRACK
4	ENGINE ACCESS (UPPER)	20	WING SLAT TRACK
5	LEFT ENGINE	21	WING SLAT TRACK
6	LEFT WING FUEL LEVEL TANK UNIT	22	WING SLAT TRACK
7	AILERON TRIM TAB ACTUATOR	23	WING SLAT TRACK
8	LEFT WING FUEL LEVEL TANK UNIT	24	RIGHT WING FUEL LEVEL TANK UNIT
9	LEFT WING FUEL LEVEL TANK UNIT	25	RIGHT WING FUEL LEVEL TANK UNIT
10	WING SLAT TRACK	26	RIGHT WING FUEL LEVEL TANK UNIT
11	WING SLAT TRACK	27	RIGHT ENGINE
12	WING SLAT TRACK	28	ENGINE ACCESS (UPPER)
13	WING SLAT TRACK	29	ENGINE AFT FAIRING
14	WING SLAT TRACK	30	ENGINE SUPPORT FITTING
15	WING SLAT TRACK	31	FUSELAGE FUEL TANK FILLER
16	LEFT WING FUEL LEVEL TANK UNIT	32	FUSELAGE FUEL LEVEL TANK UNIT

S380-2-00-3

Access Provisions - NA-380-1 and Later
Figure 6-00-4 (Sheet 3 of 4)



1,59 GROUND AIR CONDITIONING CONNECTION
HYDRAULIC RESERVOIR
HYDRAULIC RESERVOIR DUMP VALVE
HORIZONTAL STABILIZER TRIM ACTUATOR
HYDRAULIC RESERVOIR RELIEF VALVE
HYDRAULIC SYSTEM RETURN FILTER
AIR CONDITIONING AND PRESSURIZING SYSTEM UNITS

2 HYDRAULIC PRESSURE TRANSMITTERS

3 HYDRAULIC AUXILIARY SYSTEM SHUTOFF VALVE

4 ENGINE SUPPORT AFT

5 ENGINE BLEED AIR SHUTOFF VALVE
FAN SPEED SENSOR PLUG

6 ENGINE SUPPORT FITTING CENTER

7 FUEL LINES

8 FUEL HEATER

FUEL DRAIN

10 ENGINE ELECTRICAL CONNECTIONS

11 ENGINE POWER LEVER

12 ENGINE PYLON ACCESS

13 FUEL DRAIN

14 FLAP ADJUSTMENT ACCESS

15 FLAP ACTUATOR ROD END ACCESS

16 FUEL DRAIN ACCESS

17 FLAP ADJUSTMENT ACCESS

18 MAIN LANDING GEAR STRUT DOOR
MAIN GEAR ACTUATING CYLINDERS

19 FUEL DRAIN ACCESS

20, 41 MAIN GEAR ACTUATING CYLINDERS
MAIN GEAR UPLOCK ACTUATING CYLINDERS
FUEL BOOST PUMPS
WING FLAP INTERCONNECT CABLE
BRAKE ANTISKID CONTROL VALVES

21 WING SPLICE JOINT ACCESS

23 FUEL DRAIN ACCESS

24 FUEL DRAIN ACCESS

25 FUEL DRAIN ACCESS

26 FUEL CHECK VALVE
SINGLE POINT REFUELING EQUIPMENT

26A DEICER BOOT FILTER, EJECTOR FILTER

27 SPEED BRAKE
SPEED BRAKE LIMIT SWITCH

28 SPEED BRAKE HINGE

29 JACKING POINT ACCESS

30 HYDRAULIC LINE ACCESS

31, 33 LANDING GEAR POSITION SELECTOR VALVE
ALTERNATE STATIC PORT
NOSE GEAR STEERING VALVES
BRAKE RESERVOIR
BRAKE CONTROL VALVES
PARKING BRAKE VALVE
NOSE GEAR ACTUATING CYLINDER
NOSE GEAR UPLOCK SWITCH

32 LANDING AND TAXI LIGHTS

34 SPEED BRAKE HINGE PIN ACCESS

35 SPEED BRAKE HINGE

36 SPEED BRAKE CONTROL VALVE
SPEED BRAKE DUMP VALVE

37 FUEL DRAIN ACCESS

38 FUEL CHECK VALVE

39 FUEL DRAIN ACCESS

40 FUEL DRAIN ACCESS

42 FUEL DRAIN ACCESS

43 MAIN LANDING GEAR STRUT DOOR
MAIN GEAR ACTUATING CYLINDERS

44 FLAP ADJUSTMENT ACCESS

45 FLAP ACTUATOR ROD END ACCESS

46 FUEL DRAIN ACCESS

47 FLAP ADJUSTMENT ACCESS

48 FUEL DRAIN

49 ENGINE PYLON ACCESS

50 ENGINE POWER LEVER

51 HYDRAULIC SYSTEM RELIEF VALVE
HYDRAULIC SYSTEM PRESSURE FILTERS
FUEL DRAIN VALVE
FIRE EXTINGUISHING BOTTLES

52 ENGINE ELECTRICAL CONNECTIONS

53 FUEL HEATER

54 FUEL LINES

55 ENGINE SUPPORT FITTING CENTER

56 ENGINE BLEED AIR SHUTOFF VALVE
FAN SPEED SENSOR PLUG

57 ENGINE SUPPORT AFT

58 HYDRAULIC GROUND PRESSURE TEST

S380-2-00-4 A

Access Provisions - S/N 380-1 and Subs
Figure 6-00-4 (Sheet 4 of 4)

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CHAPTER

07

LIFTING AND SHORING

CHAPTER 7

LIFTING AND SHORING

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CHAPTER 7

LIFTING AND SHORING

LIST OF EFFECTIVE PAGES

Chapter/ Section	Page	Date
Chapter 7-L.E.P.	1/2 *	Mar 15/83
Chapter 7-CONTENTS	1	Mar 1/70
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	202	Mar 1/70

* The asterisk indicates pages revised, added or deleted by the current revision.

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LIFTING AND SHORING - DESCRIPTION

1. General

Provisions are made for hoisting either the entire airplane or the individual components. Hoist sling attach points are provided on each wing panel on the fuselage and on each horizontal stabilizer panel. (See figure 7-10-201.) Hoisting points are also provided on each engine. (Refer to Chapter 71.) Jacking points are provided on the nose and main landing gear struts, on each wing panel, and on the fuselage. An additonal jack point is provided at each engine pylon and at the center of the wing assembly.

CAUTION: PYLON AND CENTER WING JACK POINTS ARE TO BE USED ONLY WHEN THE WING IS NOT INSTALLED ON THE AIRPLANE. IF THESE POINTS ARE USED TO LIFT THE ENTIRE AIRPLANE, STRUCTURAL DAMAGE WILL RESULT.

AIRPLANE HOISTING - DESCRIPTION

1. General

When hoisting the entire airplane or any of the airplane components, it is recommended that hoisting slings, manufactured specifically for the airplane, be used. (See figure 7-10-201.)

NOTE: These slings are designed to lift the airplane or components from the approximate center of gravity. The fuselage hoist sling is easily adjusted to allow for different weight and center-of-gravity variations.

The engine hoist slings are covered in Chapter 71.

AIRPLANE HOISTING - MAINTENANCE PRACTICES

1. General

The procedures for hoisting the complete airplane, complete wing, and the stabilizer panels are shown in figure 7-10-201. The procedures for hoisting single wing panels and fuselage are covered in Chapter 57. Engine hoist procedure is covered in Chapter 71.

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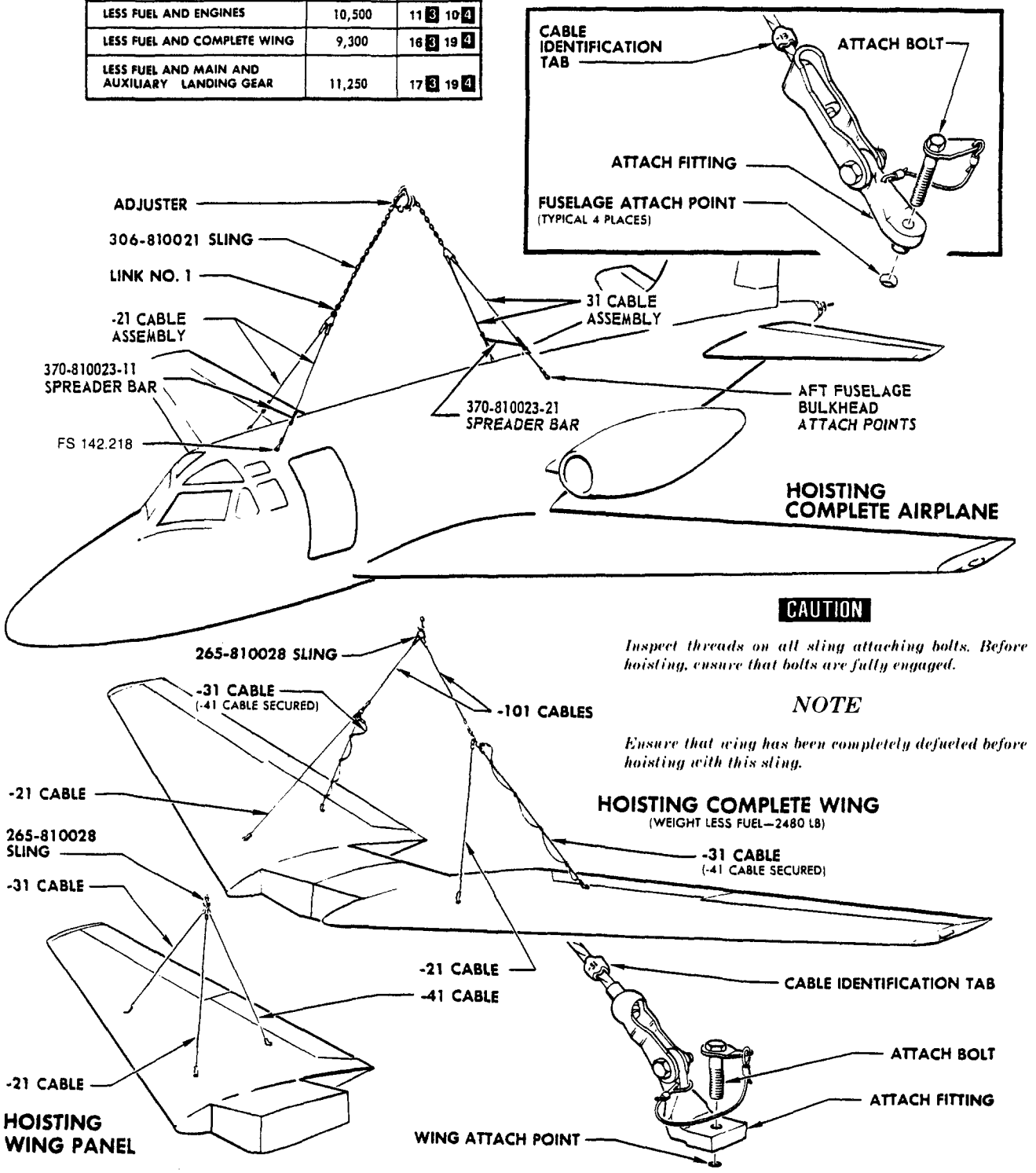
ADJUSTING SLING TO HOISTING CONDITIONS ON COMPLETE AIRPLANE		
HOISTING CONDITIONS	WEIGHT 1	LINK NO 2
MAXIMUM LOAD	21,700 3 23,500 4	20 3 21 4
EMPTY	11,850	17 3 19 4
LESS FUEL AND ENGINES	10,500	11 3 10 4
LESS FUEL AND COMPLETE WING	9,300	16 3 19 4
LESS FUEL AND MAIN AND AUXILIARY LANDING GEAR	11,250	17 3 19 4

1 LBS — APPROX.

2 PLACE DESIGNATED CHAINLINK IN CENTER OF ADJUSTER

3 S/N 370-2 THRU -6, -8 AND -9

4 S/N 370-1 AND -7 ALSO S/N 380-1 AND SUBS



CAUTION

Inspect threads on all sling attaching bolts. Before hoisting, ensure that bolts are fully engaged.

NOTE

Ensure that wing has been completely defueled before hoisting with this sling.

Hoisting Sling
Figure 7-10-201

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AIRPLANE JACKING - DESCRIPTION

1. General (See figure 7-20-201.)

All jack points on the fuselage and wings of this airplane require the use of jacking adapters. Adapters are not required at landing gear strut jack points.

Jack points are on each wing at station 94.187. A fuselage nose jack point is aft of the nose gear at station 116.772. Main and nose gear jack points are on the bottom of each landing gear strut. A jack point at the center wing skid is used to support the wing when it is removed from the airplane. A jack point in each engine pylon is used to support the airplane when the wing is removed.

CAUTION: JACK POINTS IN THE CENTER WING SECTION AND ENGINE PYLONS SHOULD NOT BE USED TO JACK THE AIRPLANE WHEN THE WING IS INSTALLED, AS STRUCTURAL DAMAGE WILL RESULT.

2. Jacking Weight Limitations

The maximum allowable weight for jacking the complete airplane is 21,700 pounds for NA-370-2 through -6, -8 and -9, and 23,500 pounds for NA-370-1, -7, NA-380-1 and later airplanes.

3. Jacking Nose Gear

Jack points used for nose gear jacking are the nose landing gear strut jack point between the two nose wheels, or the fuselage jack point. The nose landing gear strut jack point is not to be used for jacking the complete airplane. On NA-370-2 through -6, -8 and -9 airplanes, the nose gear strut may be jacked, at the nose gear strut jacking point for servicing the nose gear wheels, with the airplane fully fueled and without occupants. On NA-370-1 and -7, NA-380-1 and later airplanes, the nose gear strut jacking point or the fuselage forward jacking point should not be used as a single jacking point to raise the nose gear. When performing any maintenance or servicing procedures which require that the nose gear wheels be clear of the ground, the fuselage forward jacking point and the two wing jacking points should be used.

4. Jacking Main Gear

The main landing gear struts may be jacked individually with the airplane fully fueled and without occupants.

AIRPLANE JACKING - MAINTENANCE PRACTICES

1. Jacking Airplane (See figure 7-20-201.)

A. Jacking Procedures and Precautions

(1) Operate all jacks evenly, so that airplane remains as nearly level as possible.

(2) Set locking devices on jack stands to prevent accidental lowering of airplane.

CAUTION: WALKING ON THE EMPENNAGE OR OUTER WING WHILE THE AIRPLANE IS ON JACKS MAY CAUSE MOVEMENT OF THE AIRPLANE ON THE JACKS.

(3) Head airplane into wind if it is to be jacked out of doors.

(4) When necessary to jack from a single point, keep lift to an absolute minimum and always within safe limits of jack.

(5) Use 5-ton (minimum) capacity jacks at all points when jacking airplane.

B. Jacking From Wing/Nose Jack Points

(1) Install jacking adapters in holes provided in each wing.

(2) Place a 5-ton capacity tripod jack in position and run up to contact jacking adapters on each wing.

NOTE: The airplane should not be jacked using only one wing jack point. Jacks should be used under each wing whenever it is necessary to jack the airplane from the wing jack points.

(3) Remove access cover at nose jack point, and install jacking adapter.

CAUTION: BEFORE USING WING JACKS, BE SURE THAT FLAPPER VALVE ACCESS DOOR IN WING REAR SPAR (FLAP STATION 86.081) IS INSTALLED.

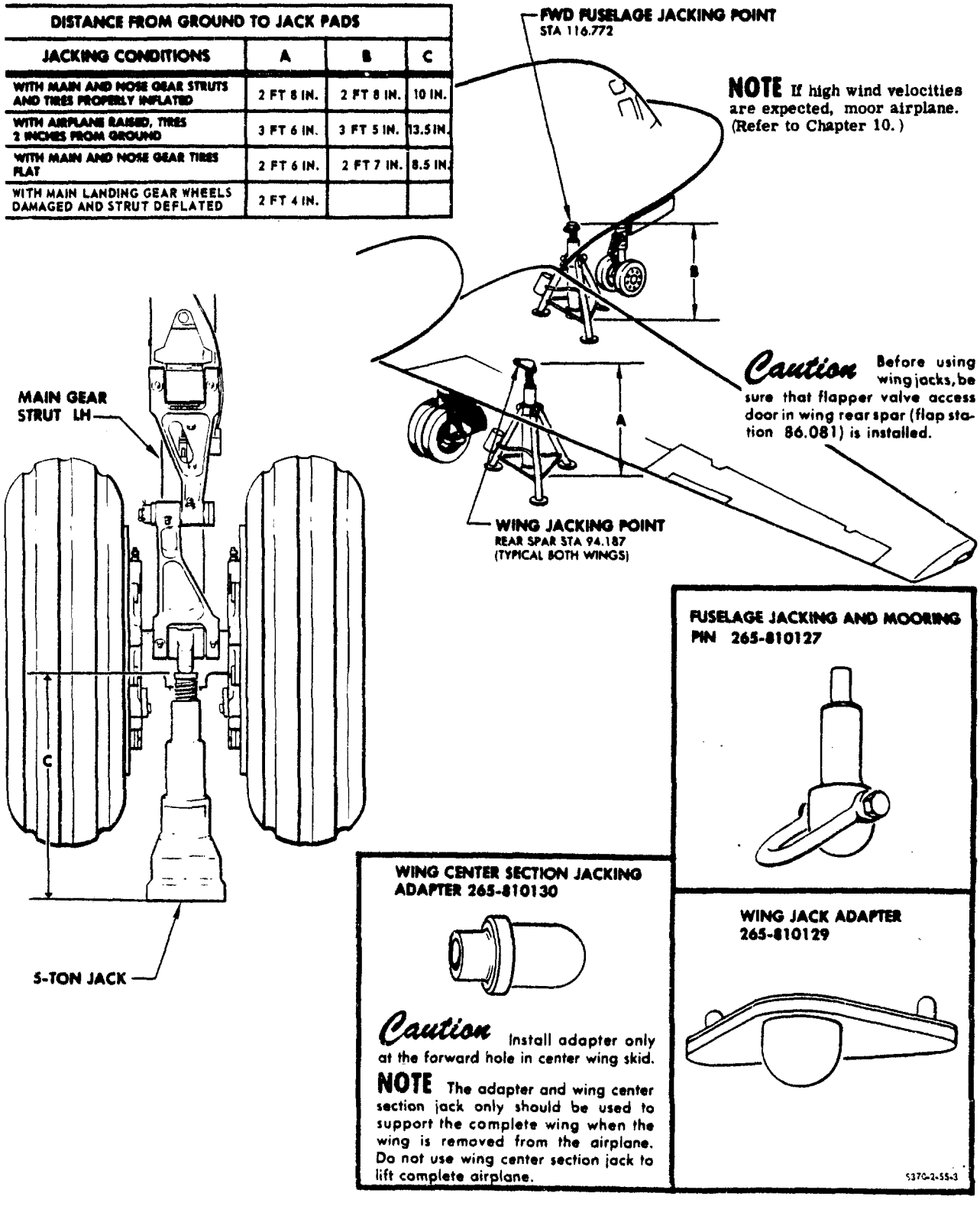
(4) Place a 5-ton capacity tripod jack in position and run up to contact nose jack pad.

C. Jacking With Engines Removed

(1) There are no limitations on jacking of the airplane with the engines removed; therefore, normal jacking procedures may be used.

Caution Do not use engine pylon jack points to lift airplane when wing is installed. Structural damage to fuselage will result.

DISTANCE FROM GROUND TO JACK PADS			
JACKING CONDITIONS	A	B	C
WITH MAIN AND NOSE GEAR STRUTS AND TIRES PROPERLY INFLATED	2 FT 8 IN.	2 FT 8 IN.	10 IN.
WITH AIRPLANE RAISED, TIRES 2 INCHES FROM GROUND	3 FT 6 IN.	3 FT 5 IN.	13.5 IN.
WITH MAIN AND NOSE GEAR TIRES FLAT	2 FT 6 IN.	2 FT 7 IN.	8.5 IN.
WITH MAIN LANDING GEAR WHEELS DAMAGED AND STRUT DEFLATED	2 FT 4 IN.		


FWD FUSELAGE JACKING POINT
STA 116.772
NOTE If high wind velocities are expected, moor airplane. (Refer to Chapter 10.)
B
A
Caution Before using wing jacks, be sure that flapper valve access door in wing rear spar (flap station 86.081) is installed.
WING JACKING POINT
REAR SPAR STA 94.187
(TYPICAL BOTH WINGS)
MAIN GEAR STRUT LH
C
5-TON JACK
FUSELAGE JACKING AND MOORING PIN 265-810127
WING CENTER SECTION JACKING ADAPTER 265-810130
Caution Install adapter only at the forward hole in center wing skid.
NOTE The adapter and wing center section jack only should be used to support the complete wing when the wing is removed from the airplane. Do not use wing center section jack to lift complete airplane.
WING JACK ADAPTER 265-810129
5370-2-55-3

Jacking Provisions
Figure 7-20-201

CHAPTER

08

LEVELING AND WEIGHING

CHAPTER 8

LEVELING AND WEIGHING

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8		1	Mar 1/70
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		2	Mar 1/70
8-20-0		1*	Jul 6/73
8-20-1		201*	Jul 6/73

*the asterisk indicates pages revised, added, or deleted by the current revision.

LEVELING - DESCRIPTION

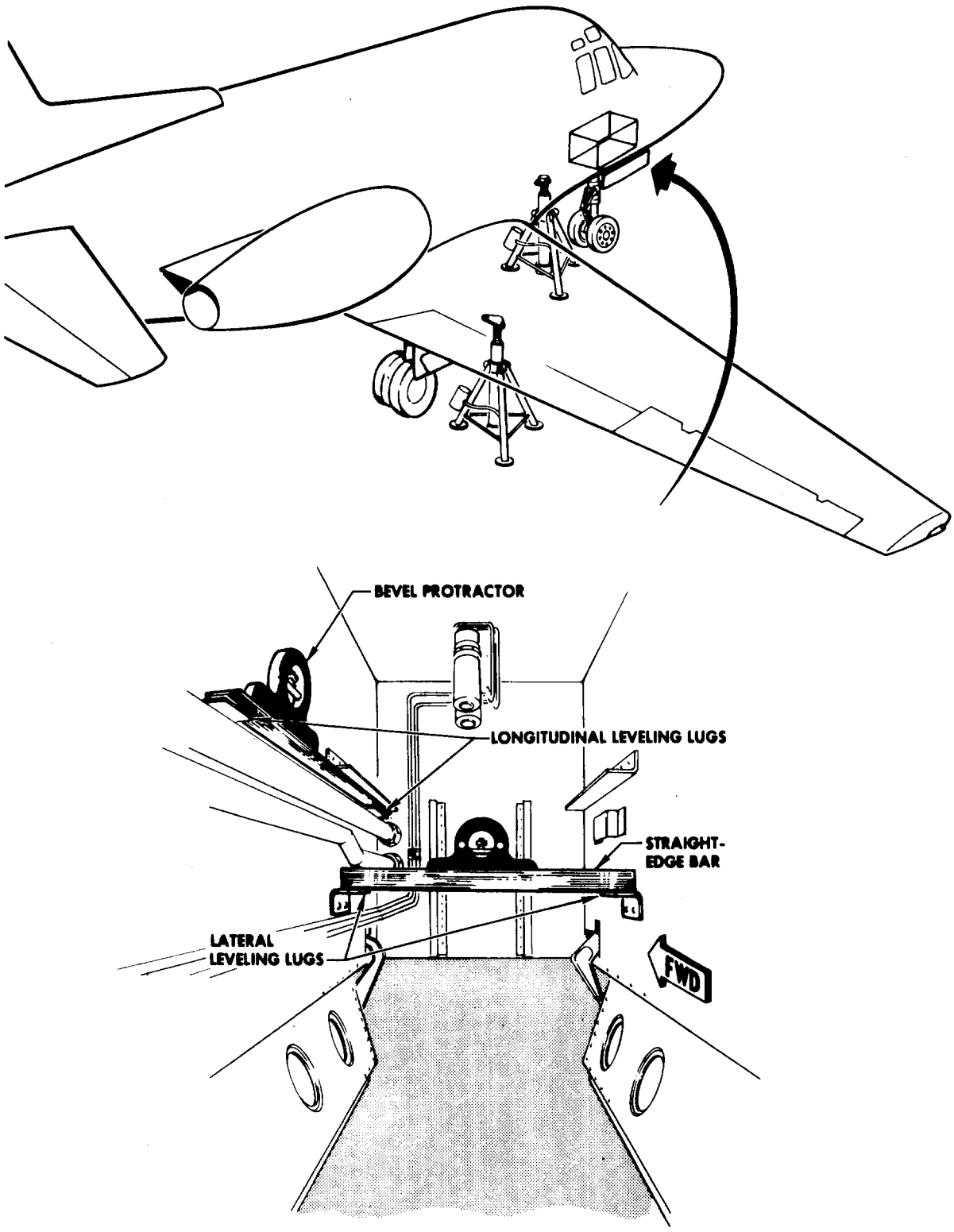
1. General (See figure 8-10-1.)

Determination of level position is done by the use of a bevel protractor and leveling brackets in the nose wheel well. The longitudinal leveling brackets are along the left side of the nose wheel well. The lateral leveling brackets are at the forward end of the wheel well.

A straightedge bar 17-3/4 inches long may be used, with the bevel protractor, on both leveling lugs.

To simulate flight attitude, adjust airplane with nose jack to a 3-degree nose-up attitude.

Leveling of the airplane is done by jacking the airplane and raising or lowering the jacks as required. Leveling can also be done by inflating the landing gear struts to the fully extended position. The nose gear strut can then be bled and lowered approximately 1.5 inches to level the airplane.



Leveling Airplane
Figure 8-10-1

WEIGHING - DESCRIPTION

1. General

Weighing of the airplane is done by two methods: Weighing on wheels using platform scales, or weighing on fuselage and wing jacks using electronic scales.

2. Airplane Condition and Definitions

A. Empty-weight Condition

The empty-weight condition is established with the wheels down, the wing flaps and slats retracted, and all control surfaces in the neutral position. This includes the following fluids:

(1) Fuel hydraulic fluid

(2) Unusable fuel (includes undrainable fuel)

(3) Full engine oil

B. Fuel

Unusable fuel is the quantity of fuel unavailable to the fuel pumps. This level corresponds to the zero point on the fuel gage. Undrainable fuel is the fuel trapped in the fuel cells and lines which cannot be drained.

The weight and center of gravity of unusable and undrainable fuel can be found in the applicable Master Equipment List, Report NA-68-865 for NA-370 airplanes or Report NA-72-929 for NA-380 airplanes.

When weighing the airplane on wheels with platform scales, the landing gear struts should be in the fully extended position. The nose gear strut can then be bled and lowered approximately 1.5 inches to level the airplane.

WEIGHING - MAINTENANCE PRACTICES

1. Weighing Airplane

A. Preparation for Weighing

(1) Conduct a complete inventory check of airplane to determine that all required equipment for empty-weight condition is installed or properly accounted for.

NOTE: It is recommended that a check list be used as an aid to consistent and thorough check-out. The quipment list section of the weight report furnished with the airplane may be used for this purpose if it is supplemented by the operator to reflect changes made to the airplane after delivery.

(2) The following fluid tanks and systems must be filled to normal operating level:

(a) Engine oil

(b) Hydraulic fluid

(3) The following fluids and/or items should be drained or removed:

(a) Fuel (defueled, then drained to the undrainable-fuel level with the airplane level)

(b) Wash water

(c) Toilet

(d) Galley and/or bar items not on equipment list (liquids, food, etc.)

(4) Remove work equipment, protective material, etc., from interior.

(5) Exterior must be dry and clean, and with protective covers, etc., removed.

(6) Close entrance door.

B. Weighing Procedure

(1) For detailed weighing instructions, refer to the applicable Load and Balance Schedule, Report NA-68-866 for NA-370 airplanes, or Report SR-73-6 for NA-380 airplanes.

CHAPTER

09

TOWING AND TAXIING

CHAPTER 9

TOWING AND TAXIING

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	2	Mar 1/70
9-10-1	201*	Aug 1/77
	202*	Aug 1/77
9-20-0	1*	Aug 1/77

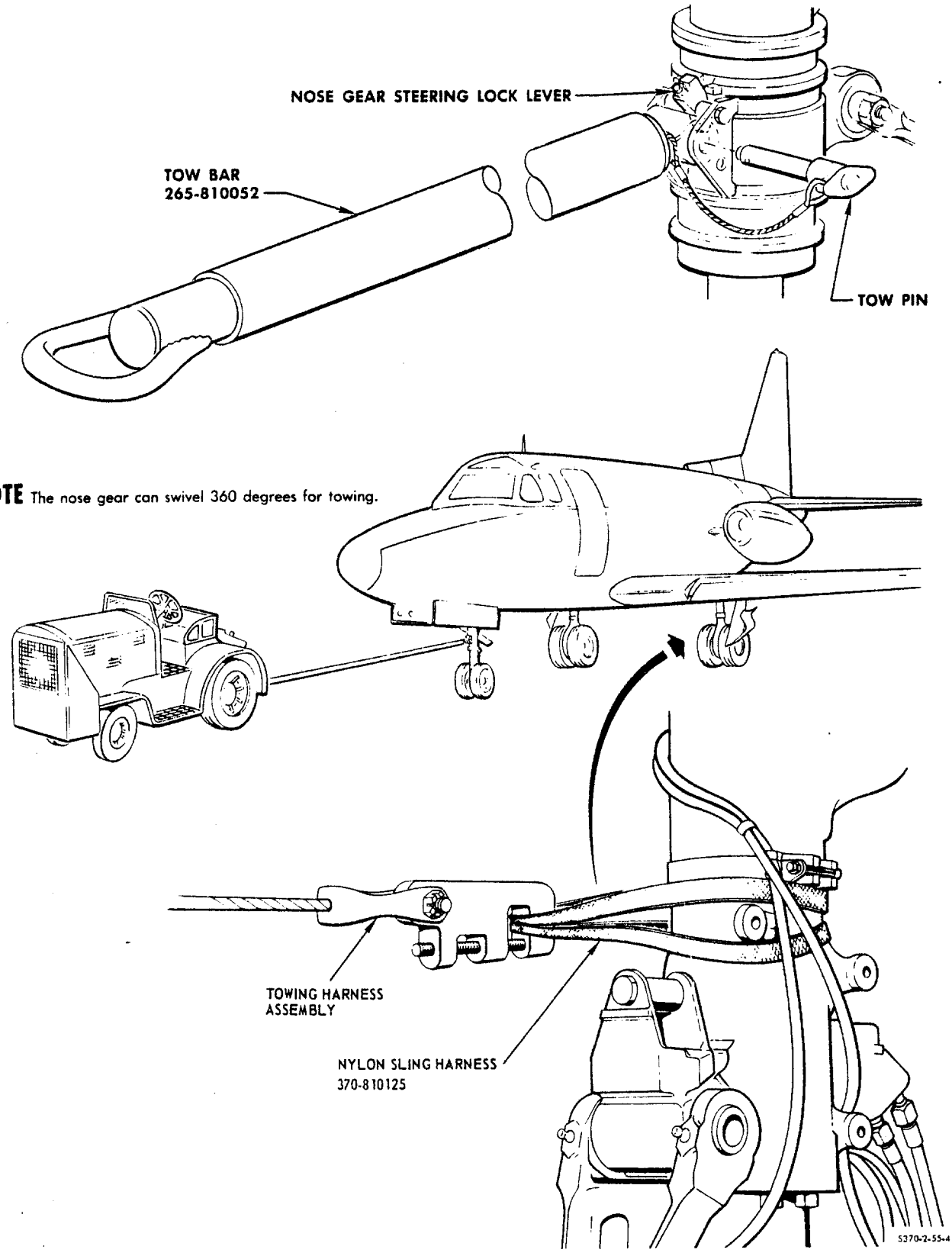
*The asterisk indicates pages revised, added, or deleted by the current revision.

TOWING - DESCRIPTION

1. General (See figure 9-10-1.)

Provision is made on the nose gear for installation of a tow bar for towing and positioning the airplane. A nylon harness may be looped around each main gear for cable attachment when rough- or muddy-surface towing is required.

NOTE: It is recommended that only towing equipment designed or approved by Rockwell International Corp., be used for towing the airplane.



NOTE The nose gear can swivel 360 degrees for towing.

Towing Provisions
Figure 9-10-1

TOWING - MAINTENANCE PRACTICES

1. Towing Airplane (See figure 9-10-1.)

A. Connecting Tow Bar

(1) Connect tow bar to nose gear, using tow pin provided. When the North-American-designed tow bar (265-810052) is used, the nose gear steering lock lever is released when the tow pin is installed. If other than the North-American-designed tow bar (265-810052) is connected to the airplane, check to ensure that lock lever is disconnected from nose gear strut collar before attempting to tow airplane.

CAUTION: IF THE LOCK LEVER IS NOT DISCONNECTED FROM THE NOSE GEAR STRUT COLLAR, DAMAGE TO THE COLLAR WILL OCCUR DURING TOWING.

- ON NA-380-1 AND LATER AIRPLANES, WHEN MOVING THE AIRPLANE IN REVERSE, EXTREME CAUTION SHOULD BE USED TO PREVENT THE NOSE WHEELS FROM RAISING OFF THE GROUND. THE AIRPLANE IS MORE SUSCEPTIBLE TO THIS CONDITION WITH THE BATTERIES REMOVED AND/OR WITH PERSONNEL IN THE CABIN AREA. IF THE BATTERIES ARE REMOVED AND PRIOR TO MOVING THE AIRCRAFT, AN EQUIVALENT AMOUNT OF BALLAST SHOULD BE INSTALLED.

(2) Disconnect the nose gear steering electrical plug on nose gear.

B. Towing Procedure and Precautions

NOTE: Maximum towing weight is equal to the maximum take-off weight configuration.

(1) Install nose and main landing gear maintenance safety locks as follows:

(a) Insert nose gear lockpin in hole in right-hand drag strut.

(b) Install main gear locks on actuator pistons and install retaining pins.

(2) Close entrance door to prevent bouncing of door against stops.

(3) During all towing operations, have a man in cockpit to operate brakes when necessary. To operate brakes without hydraulic power, pull emergency brake system control handle out on center pedestal and operate brakes in normal manner.

NOTE: There are no towing rings on the main landing gear struts. If it becomes necessary to tow the airplane by the main gears, a nylon sling (370-810201) should be wrapped around the strut, and the towing cable should be attached to the sling. This prevents the cable from damaging the strut.

(4) If airplane must be towed over rough or muddy terrain, a cable assembly can be connected to each main landing gear with a nylon sling. When airplane is towed by main landing gear, hook cable assembly directly to tug. An assistant can then guide the airplane with the tow bar installed on the nose gear.

CAUTION: BEFORE TOWING OVER ROUGH SURFACES, THE SPEED BRAKE MUST BE RETRACTED TO PREVENT IT FROM HITTING THE GROUND.

ENSURE THAT ELECTRICAL POWER IS AVAILABLE, AND THAT THE GEAR POSITION INDICATORS DEPICT A GEAR SAFE POSITION.

(5) Make sure that landing gear is free of obstructions, and that brakes are released.

NOTE: Do not tow airplane if nose gear strut is completely deflated or overextended.

(6) After airplane has been towed, check nose wheel steering quick-disconnect plug to ensure that it is connected and undamaged.

(7) After disconnecting tow bar, check that lock lever is free to reconnect with nose gear strut collar. Install and tighten nose gear steering electrical plug.

TAXIING - DESCRIPTION

1. GENERAL

The airplane is provided with an electrically controlled, hydraulically operated nose wheel steering system. Operation of the nose steering system is accomplished by depressing and releasing the nose wheel steering button on either control wheel along with the necessary movement of the rudder pedals. The gust lock must be disengaged, prior to taxing the airplane.

CHAPTER 10

PARKING, MOORING, STORAGE, RTN TO SVC.

CHAPTER 10

PARKING AND MOORING

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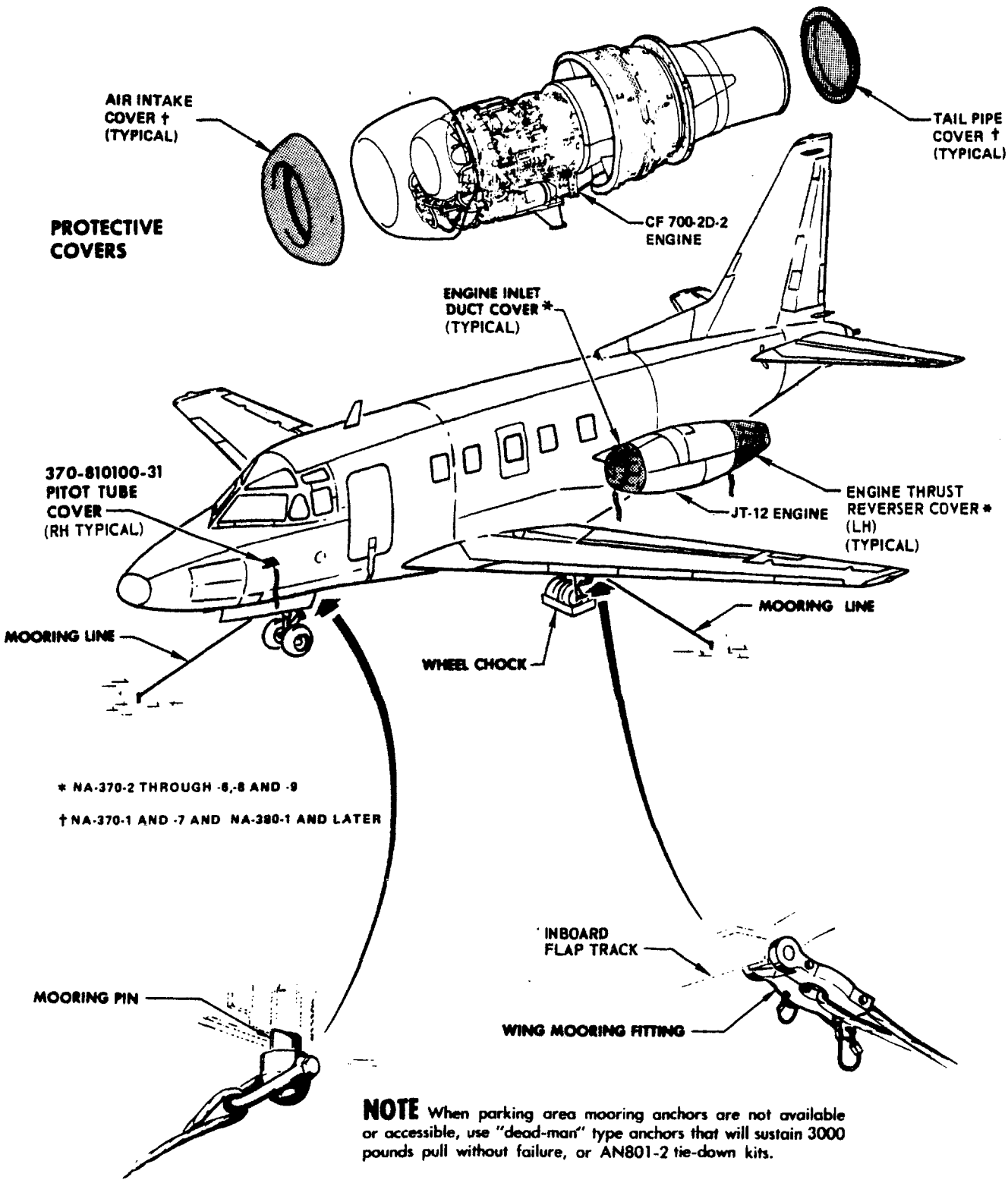
		2*	Jul 6/73
10-00-1		201	Mar 1/70

*The asterisk indicates pages revised, added, or deleted by the current revision.

PARKING AND MOORING - DESCRIPTION

1. General (See figure 10-00-1.)

Provisions are incorporated on the fuselage forward section, on each engine pylon, and on each wing for mooring the airplane. The fuselage mooring point is the same point used for jacking, and the jacking adapter serves as the mooring adapter. A complete mooring kit is provided with the airplane.


AIR INTAKE COVER † (TYPICAL)
TAIL PIPE COVER † (TYPICAL)
PROTECTIVE COVERS
CF 700-2D-2 ENGINE
ENGINE INLET DUCT COVER * (TYPICAL)
370-810100-31 PITOT TUBE COVER (RH TYPICAL)
JT-12 ENGINE
ENGINE THRUST REVERSER COVER * (LH) (TYPICAL)
MOORING LINE
MOORING LINE
WHEEL CHOCK
* NA-370-2 THROUGH -6,-8 AND -9
† NA-370-1 AND -7 AND NA-380-1 AND LATER
INBOARD FLAP TRACK
MOORING PIN
WING MOORING FITTING
NOTE When parking area mooring anchors are not available or accessible, use "dead-man" type anchors that will sustain 3000 pounds pull without failure, or AN801-2 tie-down kits.
5370-2 56-58

Tie-down and Mooring Provisions
Figure 10-00-1

PARKING AND MOORING - MAINTENANCE PRACTICES

1. Parking/Mooring Procedures

A. Parking Airplane

(1) Set parking brakes and engage gust lock.

(2) Chock main landing gear fore and aft. Do not chock nose wheels.

(3) If conditions are windy or dusty, it is advisable to install engine covers.

NOTE: Engine covers should not be installed while engines are hot.

(4) Install pitot covers.

(5) Moor airplane if winds of 40 knots or above are expected.

(6) Check that pilot's sliding window is closed.

B. Mooring Airplane (See figure 10-00-1.)

(1) When a storm or severe wind condition is expected, space airplanes a little more than a wingspan apart.

(2) Head airplane into wind and follow parking-airplane procedures.

(3) Install mooring fittings and attach tie-down lines.

NOTE: Chain or cable that will withstand a 3000-pound pull test should be used. Use 3/4-inch Manila rope if chain or cable is not available. Rope is least desirable, because of its tendency to shrink when wet.

(4) Deflate nose gear strut to prevent bouncing of airplane.

(5) Retract speed brake.

(6) If snow is expected, it is advisable to cover cockpit, wings, and horizontal stabilizer. (See figure 10-00-1.)

CHAPTER

11

PLACARDS AND MARKINGS

CHAPTER 11

REQUIRED PLACARDS

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RECEIVED

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		2	Mar 1/70
11-50-0	Deleted	1*	May 15/81
	Deleted	2*	May 15/81

* The asterisk indicates pages revised, added, or deleted by the current revision.

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REQUIRED PLACARDS - DESCRIPTION

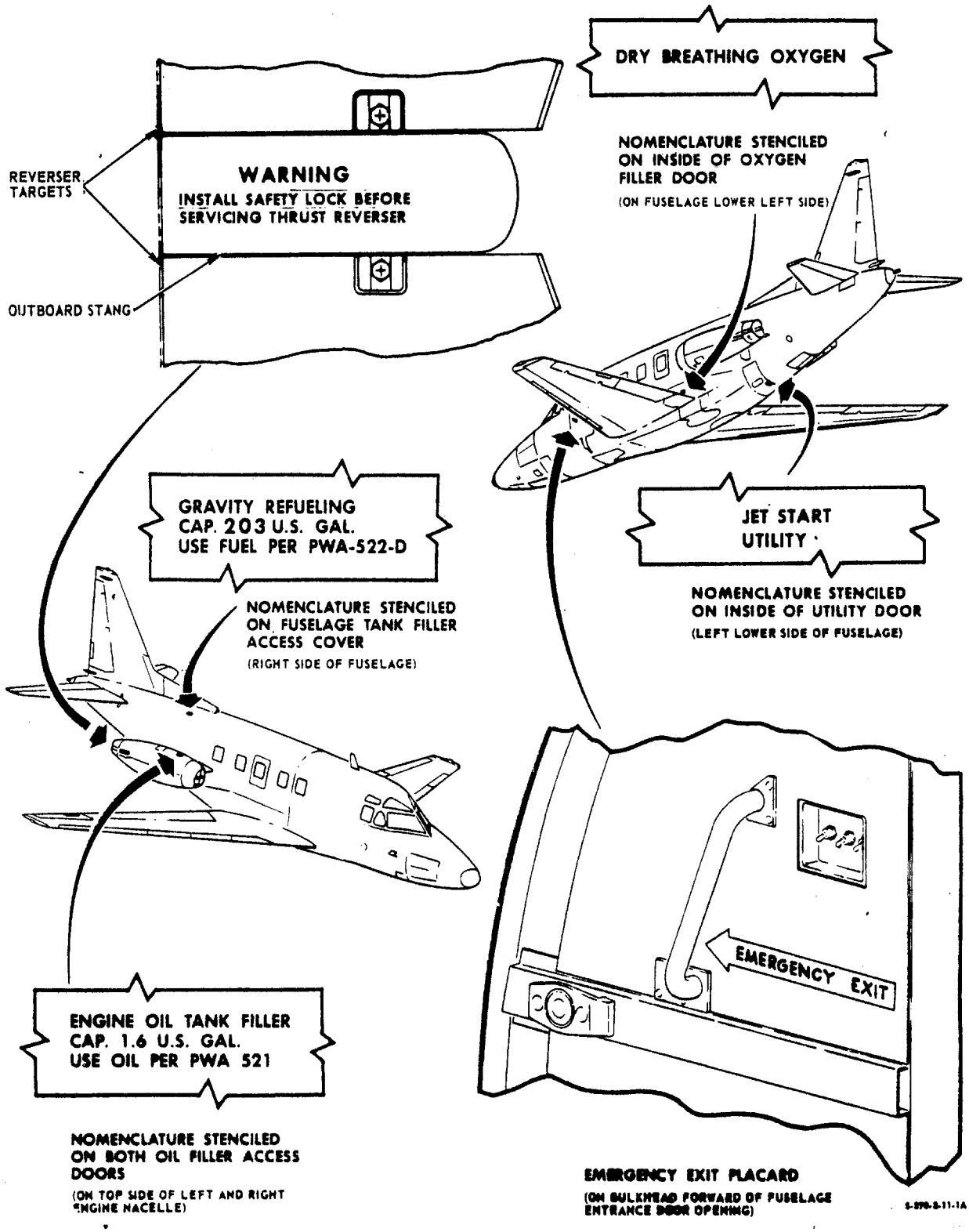
1. General

Required placards, name plates, and stencil markings which give operating or installation instructions; directional movements; servicing instructions; part or position identification; and safety precautions, both internal and external, are illustrated with locations and nomenclature shown.

FUSELAGE AND NACELLE - DESCRIPTION AND OPERATION

1. General

Figure 11-10-1 shows placards, decals, and markings for the fuselage and nacelle.

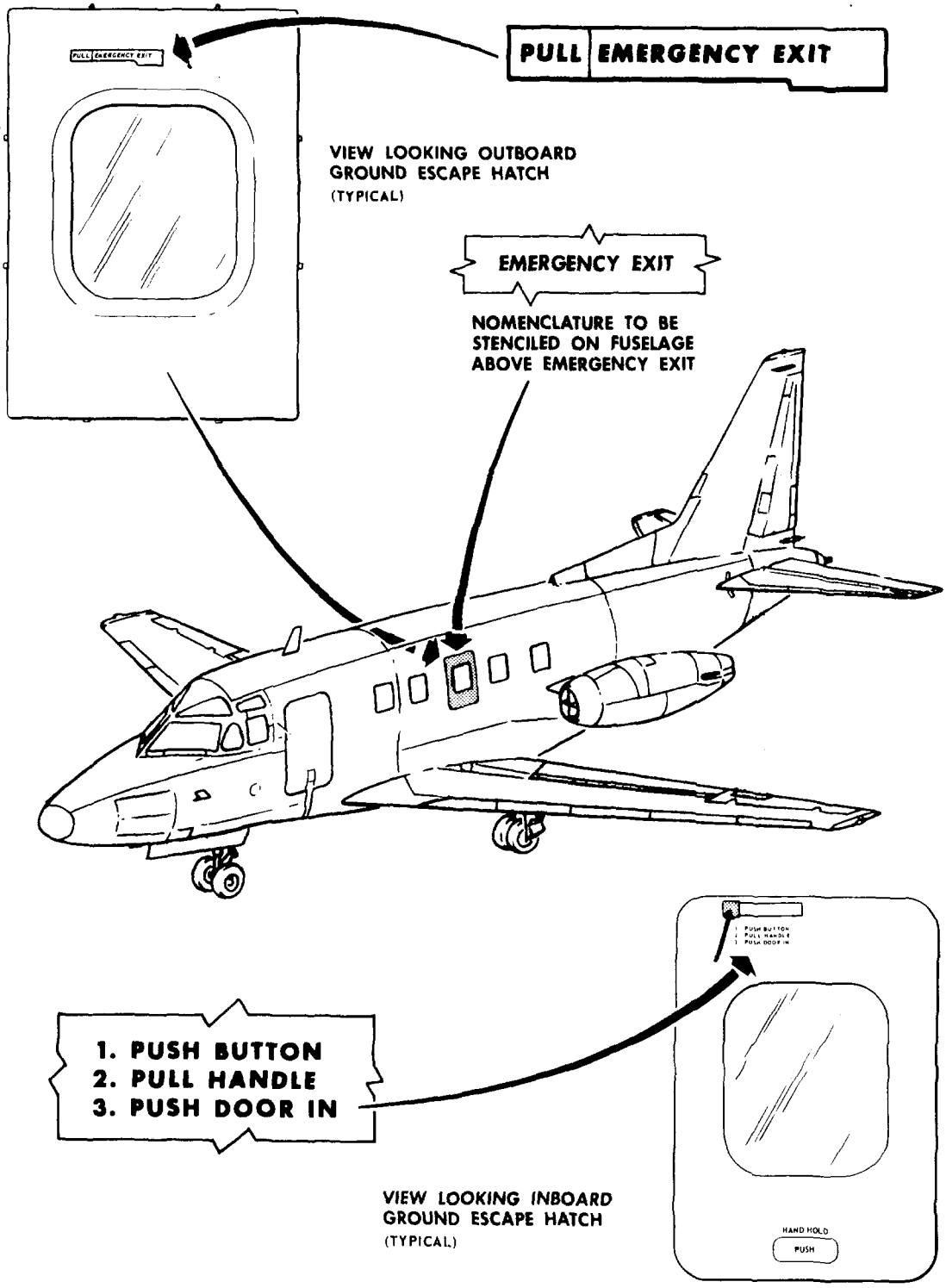

DRY BREATHING OXYGEN
NOMENCLATURE STENCILED ON INSIDE OF OXYGEN FILLER DOOR
(ON FUSELAGE LOWER LEFT SIDE)
REVERSER TARGETS
WARNING
INSTALL SAFETY LOCK BEFORE SERVICING THRUST REVERSER
OUTBOARD STANG
GRAVITY REFUELING
CAP. 203 U.S. GAL.
USE FUEL PER PWA-522-D
NOMENCLATURE STENCILED ON FUSELAGE TANK FILLER ACCESS COVER
(RIGHT SIDE OF FUSELAGE)
JET START UTILITY
NOMENCLATURE STENCILED ON INSIDE OF UTILITY DOOR
(LEFT LOWER SIDE OF FUSELAGE)
EMERGENCY EXIT
ENGINE OIL TANK FILLER
CAP. 1.6 U.S. GAL.
USE OIL PER PWA 521
NOMENCLATURE STENCILED ON BOTH OIL FILLER ACCESS DOORS
(ON TOP SIDE OF LEFT AND RIGHT ENGINE NACELLE)
EMERGENCY EXIT PLACARD
(ON BULKHEAD FORWARD OF FUSELAGE ENTRANCE DOOR OPENING)

Fuselage and Nacelle Markings
Figure 11-10-1

GROUND ESCAPE HATCH - DESCRIPTION AND OPERATION

1. General

Figure 11-20-1 shows placards, decals, and markings for the ground escape hatch.

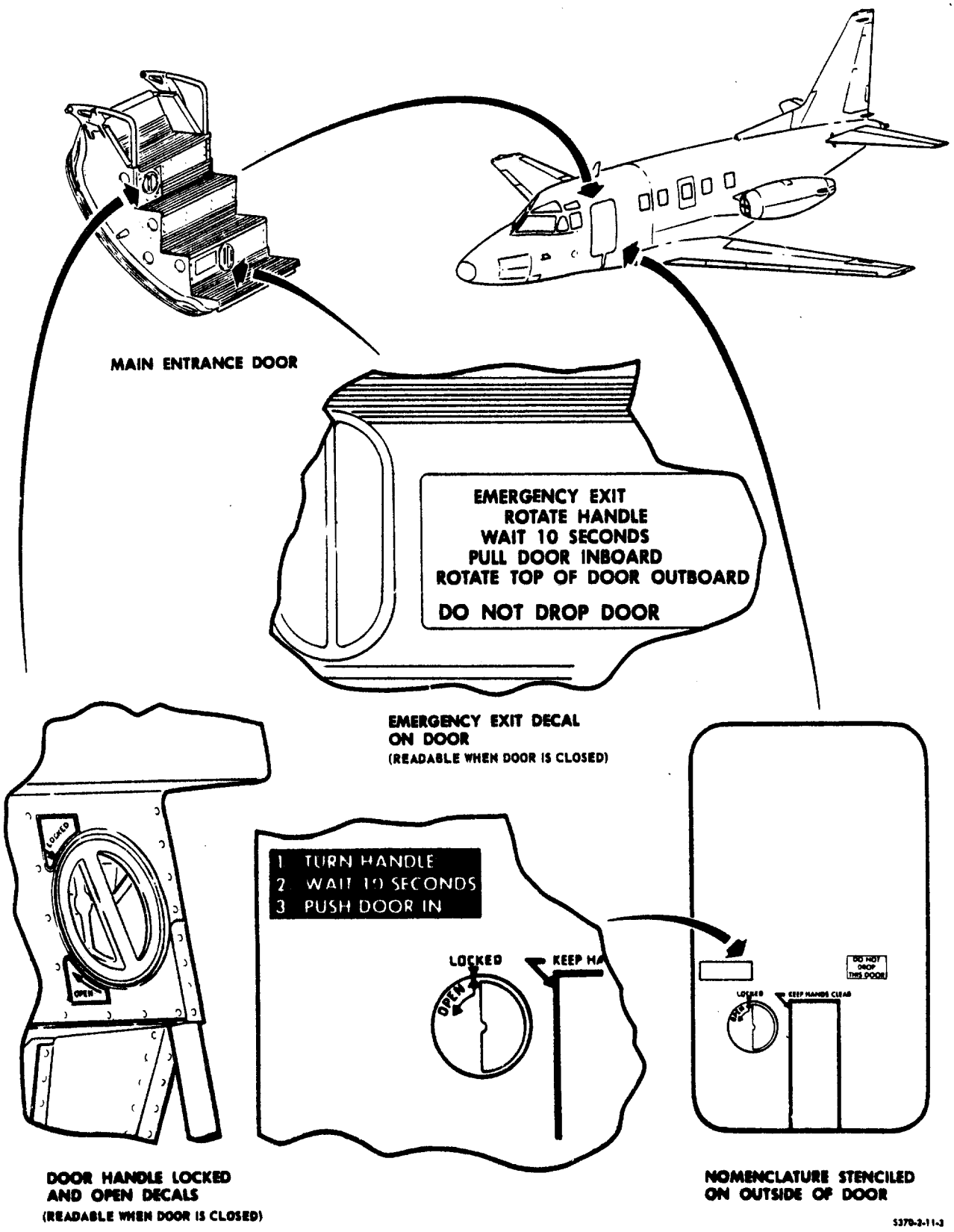


Ground Emergency Escape Hatch Markings
Figure 11-20-1

MAIN ENTRANCE DOOR - DESCRIPTION AND OPERATION

1. General

Figure 11-30-1 shows placards, decals, and markings for the main entrance door.

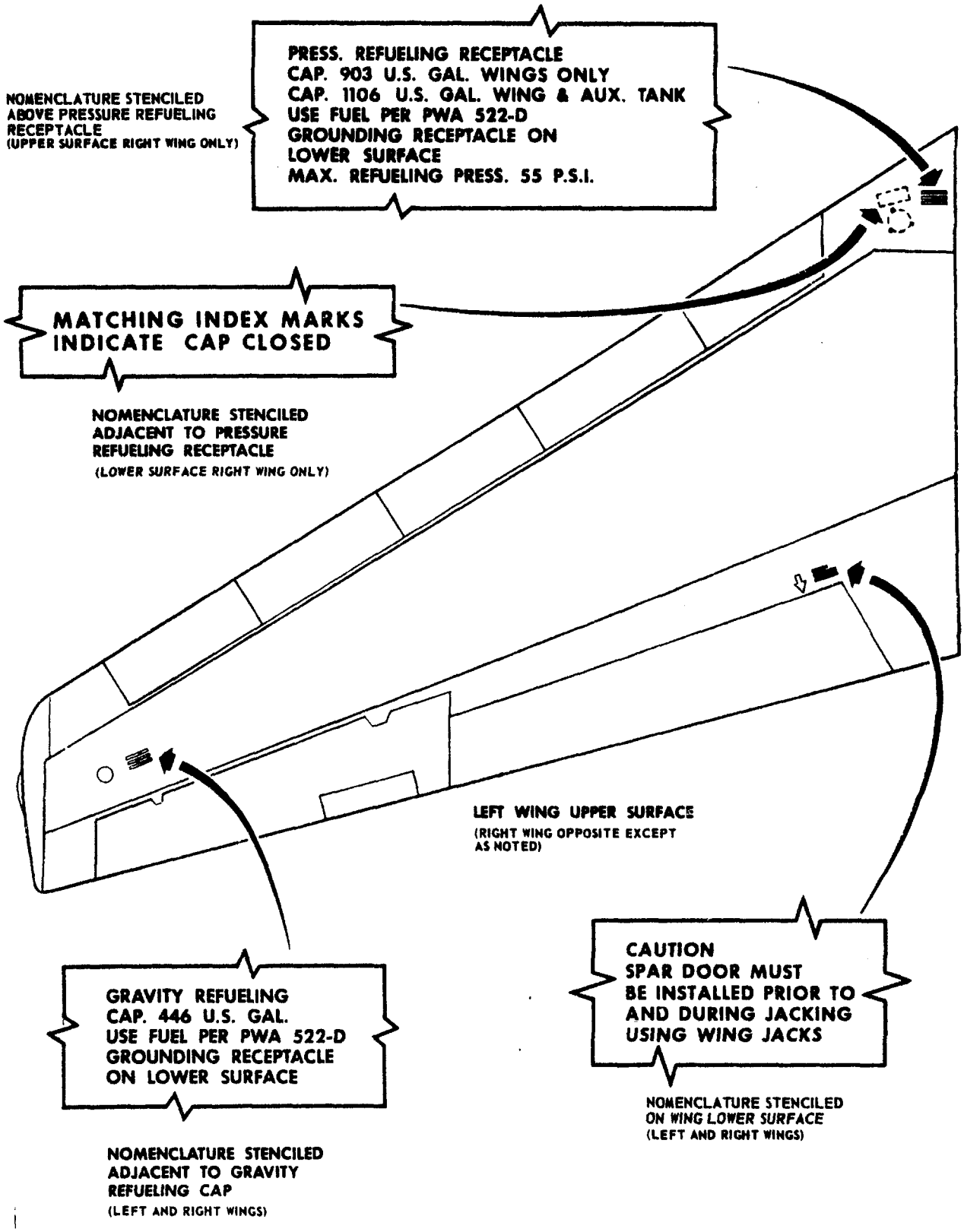


Main Entrance Door Markings
Figure 11-30-1

WINGS - DESCRIPTION AND OPERATION

1. General

Figure 11-40-1 shows placards, decals, and markings on the upper and lower sides of the left and right wings.



Wing Markings
Figure 11-40-1

CHAPTER

12

SERVICING

CHAPTER 12

SERVICING

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*The asterisk indicates pages revised, added, or deleted by the current revision.

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SERVICING - DESCRIPTION

1. GENERAL

A. Servicing/Replenishing Fluid Systems

Access provisions are shown in Chapter 6, figure 6-00-3 or 6-00-4. When servicing and replenishing the various systems, pay particular attention to the following:

(1) Drip pans under accessories

(2) Fire extinguisher discharge indicators

(3) Tire condition

(4) Tire slip marks

(5) Oxygen pressure indicator

(6) Hydraulic lines (landing gear and speed brake)

(7) Access doors closed and secure

(8) Fuel stains caused by leaks from fuel tanks, plumbing, or components.

(9) All visible plumbing for leaks.

B. Replenishing Data (See figure 12-00-1 for location of servicing point).

UNITS & SPEC.		SERVICING CAPACITIES*			FOR SERVICING REFER TO CHAPTER
		US. GAL.	IMP. GAL.	LITERS	
FUEL TANKS PWA-522 AS REVISED	WING EACH WING TOTAL FUSELAGE TOTAL	451.5(446) 903 (892) 203 1106	376(371) 752(742) 164 922	1709(1688) 3418(3376) 768 4181	12-13-0
ENGINE OIL	EACH ENGINE	1.6† 1.0‡	1.3† 0.8‡	6.05† 3.7 ‡	12-16-0
HYDRAULIC RESERVOIR MIL-H-5606A(RED)		2	1.66	7.57	12-15-0

*Capacities shown are for servicing only and do not necessarily represent usable or total quantity in system.
†JT-12A Engines
‡CF700-2D-2 Engines
Fuel servicing capacities in parentheses are for gravity refueling, the other fuel capacities are for single-point pressure refueling.

		PSI	ATM	FOR SERVICING REFER TO CHAP.
TIRES NITROGEN BB-N-411 OR DRY CLEAN AIR	NOSE GEAR (2 req'd) MAIN GEAR (4 req'd)	75(+5-0) †165(+5,-0) ‡180(+5,-0)	6.8 9.57	12-10-0 12-10-0
HYDRAULIC ACCUMULATORS NITROGEN BB-N-411 OR DRY CLEAN AIR	NORMAL AND AUX THRUST REVERSER†	1500±50 900±50	102±3.4 61.25±3.4	12-15-0 12-15-0
HYDRAULIC RESERVOIR AIR TANK		90 MAX	6.1	12-15-0
OXYGEN CYLINDER(S) OXYGEN MIL-O-27210 AVIATOR BREATHING		1800	122.5	12-17-0

LANDING GEAR SHOCK STRUTS	HYD. FLUID MIL-H-5606A	FILL TO LEVEL OF FILLER HOLE IN STRUT	12-11-0
	NIT. BB-N-411 OR CLEAN DRY AIR	REFER TO STRUT INSTRUCTION PLATE (MAIN GEAR ONLY)	12-11-0
REFRIGERATION UNIT COOLING TURBINE	*	CAPACITY IS 100 CUBIC CENTIMETERS(3.4 FLUID OZ.)	21-50-2
DEHYDRATOR CAPSULES (3 REQ)	SILICA GEL MIL-D-3716 GRADE H TYPE IV		56-00-1
ENGINE FIRE EXTINGUISHER (2 REQ)	MONOBROMOTRIFLUOROMETHANE (CF_3Br)		26-20-1

*Refer to: "Table of Approved Refrigeration Unit Cooling Turbine Oils."
†NA-370-2 Through -6, -8 and -9
‡NA-370-1 and -7, NA-380 and Later

C. Table of Acceptable Fuels - NA-370-2 Through -6, -8 and -9

The fuels in the following list are applicable to aircraft NA-370-2 through -6, -8 and -9, and are known to conform to Pratt & Whitney Aircraft Specification PWA 522. This is not intended as a complete listing, but is provided as examples of those fuels that do comply with the specification.

(1) American Jet Type A

(2) American Jet Type A-1

(3) Caltex Universal Aviation Turbine Fuel

(4) Caltex Jet Fuel Type JP-4

(5) Conoco Jet 50

(6) Conoco Jet 60

(7) Conoco JP-4

(8) Esso Turbo Fuel 1-A

(9) Esso Turbo Fuel A4

(10) Mobil Jet 140 (Mobil Jet-A)

(11) Mobil Jet 158 (Mobil Jet A-1)

(12) Mobil Jet 476 (Mobil Jet B)

(13) Phillips Kerosene Grade TF

(14) Richfield Kerosene Type A

(15) Richfield Kerosene Type A-1

(16) Richfield Kerosene Type B

(17) Aero Shell Turbine Fuel 640

(18) Aero Shell Turbine Fuel 650

(19) Aero Shell Turbine Fuel JP-4

(20) Sinclair Superjet Type A

(21) Chevron Aviation Turbine Fuel No. 1

(22) Standard Oil of Ohio's Type A Turbine Kerosene

(23) Esso Turbine Fuel 1B (Standard Oil of Kentucky)

(24) AV Jet K-40 (Texaco)

(25) AV Jet K-58 (Texaco)

(26) 76 Turbine Fuel (Union Oil Co.)

(27) Military JP-4

(28) Gulflite Turbine Fuel "A"

C1. Table of Acceptable Fuels - NA-370-1 and -7, NA-380-1 and Later

The fuels in the following list are applicable to aircraft NA-370-1 and -7, NA-380-1 and Later, and are known to conform to General Electric Co., Specification D50TF2. Mixing of fuel types is acceptable, providing the density control on the engine fuel control is properly set. The following list is not intended as a complete listing but is provided as examples of fuels complying with General Electric Co., Specification D50TF2.

(1) American Jet Fuel Type A

(2) American Jet Fuel Type A-1

(3) Arcojet A

(4) Arcojet A-1

(5) Arcojet B

(6) BPA T.K.

(7) BPA T.G.

(8) BPAVCAT 48

(9) Caltex Jet A1

(10) Caltex Jet B

(11) Turbine Type A

(12) Conoco Jet 40

(13) Conoco Jet 50

(14) Conoco Jet 60

(15) Conoco JP-4

(16) Esso Turbo Fuel A

(17) Esso Turbo Fuel A1

(18) Esso Turbo Fuel B

(19) Esso Turbo Fuel 5

(20) Enco Turbo Fuel A

(21) Enco Turbo Fuel A-1

(22) Enco Turbo Fuel B

(23) Enco Turbo Fuel 5

(24) Gulf Jet A

(25) Gulf Jet A-1

(26) Mobil Jet A

(27) Mobil Jet A-1

(28) Mobil Jet B

(29) Mobil Jet 4

(30) Mobil Jet 5

(31) Philjet A-50

(32) Philjet JP-4

(33) Purejet Turbine Fuel Type A

(34) Purejet Turbine Fuel Type A-1

(35) Aeroshell Turbine Fuel JP-4

(36) Aeroshell Turbine Fuel 640

(37) Aeroshell Turbine Fuel 650

(38) Sinclair Superjet Fuel

(39) Chevron Jet Fuel A-1

(40) Chevron Turbine Fuel B

(41) Standard JF A

(42) Standard JF A-1

(43) Murphy Jet A

(44) Jet A Kerosene

(45) Jet A-1 Kerosene

(46) Standard Turbine Fuel A-1

(47) Standard Turbine Fuel B

(48) Texaco Avjet A

(49) Texaco Avjet A-1

(50) Texaco Avjet B

(51) 76 Turbine Fuel

(52) Union JP-4

The following engine fuels conforming to the following Industry/Government Specifications are also acceptable fuels:

(1) Air Total Turbine Fuel, 1 and 1A

(2) ASTM Jet A Aircraft Turbine Fuel

(3) ASTM Jet B Aircraft Turbine Fuel

(4) ASTM Jet A-1

(5) British Fuel D Eng. R.D. 2482, AVTUR 40

(6) British Fuel D Eng. R.D. 2486, AVTAG

(7) British Fuel D Eng. R.D. 2494, AVTUR 50

(8) British Fuel D Eng. R.D. 2498 AVCAT 48

(9) British Fuel D Eng. R.D. 2488 AVCAT

(10) Canadian Fuel 3-GP-22

(11) Canadian Fuel 3-GP-23

(12) MIL-T-5624G JP-4

(13) MIL-T-5624G JP-5

(14) NATO F-30 (Jet A)

SERVICING (Continued)

Table of Acceptable Fuels — S/N 370-1 and -7, 380-1 and Subs (Cont'd)

(15) NATO F-34 (Jet A-1)
(16) NATO F-35 (Jet A-1)
(17) NATO F-40 (JP-4)
(18) NATO F-42 (JP-5)
(19) NATO F-44 (JP-5)
(20) NATO F-45 (JP-4)

D. Table of Approved Engine Oils — S/N 370-2 thru -6, -8, and -9

The engine lubricating oils in the following list comply with Specification PWA 521 for use in Pratt & Whitney Aircraft commercially operated turbojet engines.

	(1)	Castrol 3C Turb. Lubricant	1 (6)	Mobil Jet Oil 254
1	(2)	Esso Turbo Oil 2380/Exxon Turb. Oil 2380	(7)	BP Aero Turbine Oil 15
1	(3)	Esso Turbo Oil 25/Exxon Turbo Oil 25	1 (8)	Aero Shell Turbine Oil 500/Royco 500
1	(4)	Castrol 5000	1 (9)	Aero Shell Turbine Oil 555/Royco 555
1	(5)	Mobil Jet II	1 (10)	Aeroshell Turbine Oil 560/Royco 560

D1. Table 6. Approved Engine Oils — S/N 370-1 and -7, 380-1 and Subs

The engine lubricating oils in the following lists are applicable to S/N 370-1 and -7, 380-1 and subs and are known to conform to the requirements of General Electric Co., Specification D50TF1. Both Type I and Type II oils are listed. Type I oils are recommended for use in climates where the temperature is expected to be below -18°C or in instances where the Type II oils are not available. Type II oils have the capability of withstanding higher operating temperatures and also have improved anti-coking characteristics.

Table 6.

	NAME	SUPPLIER
	(1) 2389 Turbo Oil	Humble Oil and Refining Co.
	(2) RM 184 A	Mobil Oil Company
	(3) Shell Aircraft Oil 307	Shell Oil Company
1	(4) Aeroshell Turbine Oil 500	Shell Oil Company
1	(5) Aeroshell Turbine Oil 555	Shell Oil Company
1	(6) Aeroshell Turbine Oil 560	Shell Oil Company
1	(7) AVTUR Oil Synthetic	Dillons Chemical Company
1	(8) Castrol 205	Stauffer Chemical Company, Castrol, Ltd.
1	(9) Caltex RPM Jet Engine Oil 5	Caltex Petroleum Corporation
1	(10) Caltex 7388	Caltex Petroleum Corporation
1	(11) Chevron Jet Engine Oil 5	Chevron International Oil Company
1	(12) Enco Turbo Oil 2380	Humble Oil & Refining Company

(Table 6 continued on page 4D)

1 Type II oils (All others listed are Type I oils)

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SERVICING (Continued)

Table 6. (Cont'd)

	NAME	SUPPLIER
1	(13) Esso Turbo Oil 2380	Humble Oil & Refining Company
1	(14) Mobil Jet Engine Oil II	Mobil Oil Company
1	(15) Mobil Oil 254	Mobil Oil Company
1	(16) Royco Turbine Oil 560	Royal Lubricants
1	(17) Sinclair Turbo-S Type 2	Sinclair Refining Company
1	(18) Stauffer Jet II	Stauffer Chemical Company
1	(19) Texaco SATO 7388	Texaco, Incorporated
1	(20) Texaco Starjet 5	Texaco, Incorporated
1	(21) Caltex Starjet 5	Caltex Petroleum Corporation
1	(22) Exxon Turbo Oil 2380	Exxon Company

NOTES

- ***Stauffer Jet II, AVTUR Oil Synthetic, and Castrol 205 are identical oils.***
- ***Texaco SATO 7388 and Caltex 7388 are identical oils.***
- ***Chevron Jet Engine Oil 5 and Caltex RPM Jet Engine Oil 5 are identical oils.***
- ***Enco Turbo Oil 2380, Esso Turbo Oil 2380 and Exxon Turbo Oil 2380 are identical oils.***
- ***Texaco Starjet 5 and Caltex Starjet 5 are identical oils.***
- ***Aeroshell Turbine Oil 560 and Royco Turbine 560 are identical oils.***

D2. **Table 7. Approved Lubricating Oils — GAS Turbine Auxiliary Power Unit**

The following list of lubricating oils complies with the auxiliary power unit manufacturer's requirements and specifications.

CAUTION

The engine will operate satisfactorily with any of the oils listed; however, different types of oils should not be mixed. If the engine oil supply is low and the type of oil in use is not available, drain the sump and replenish with another qualified oil. The oil filter element should be changed if a new element is available.

2	MIL-L-7808 or commercial equivalent	Sinclair Turbo Oil 35
	MIL-L-23699 or commercial equivalent	Texaco Sato 35
	Esso Turbo Oil 35	Caltex Sato 35
	Esso Extra Turbo Oil 274	Regent Sato 35
	Enco Turbo Oil 35	Castrol 98 UK
	Enco Turbo Oil 2380	B.P. Aero Turbine Oil 35
	Enco Extra Turbo Oil 274	PWA 521
	Aeroshell Turbine Oil 750	Mobil Jet II

1 Type II oils (All others listed are Type I oils)

2 This oil is approved for a minimum oil temperature of -65°F; all other oils are approved to -40°F.

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SERVICING (Continued)

E. **Table 8. Approved Refrigeration Unit Cooling Turbine Oils for S/N 370-1 thru 370-9 380-1 and Subs**

(1) Aero Shell Turbine Oil 500	(9) Mobil Jet Oil II
(2) Brayco 899S	(10) Signal Astrol Turbine 25
(3) Caltex RPM Jet Engine Oil 5	(11) Stauffer Jet II
(4) Castrol 205	(12) Texaco or Caltex Sato 5180
(5) Chevron Jet Oil No. 5	(13) Texaco or Caltex Sato 7388
(6) Exxon Turbo Oil 2380	(14) Texaco or Caltex Sato 7730
(7) Exxon Turbo Oil 5251	(15) Texaco Synthetic Lube Oil 4508
(8) Hatcol 3211	

E1. **Table 9. Approved MIL-H-5606A Hydraulic Fluids**

NAME	SUPPLIER
Chevron Aviation Hydraulic Fluid A	Chevron International Oil Co.
Univis J-12	Exxon Oil Co.
Aero Hydraulic Fluid HFA	Mobil Oil Co.
Aero Shell 4	Shell Oil Company
Aircraft Hydraulic Oil 15 (Reference No. 1537)	Texaco Oil Co.

F. **Table of Lubrication Requirements Cross Reference**

MILITARY SPECIFICATION	COMMERCIAL EQUIVALENT
MIL-G-21164 — Grease, Molybdenum Disulphide (for Low and High Temperatures)	Castrolease MSA(C) Electro-Moly/11 Everlube 211-G Royco 64C
MIL-G-23827 A — Grease, Aircraft and Instrument, Gear and Actuator Screw	BP Aero Grease 31B Braycote 627S Supermil No. A72832 Esso No. 5114EP
MIL-G-81827 — Grease, Aircraft, High Load Capacity, Light Temperature Range	Mobil Grease 29 Aero Shell 23 Royco 22MS
MIL-G-81322D — Grease, Aircraft, General Purpose Wide Temperature Range	Aero Shell 22 Mobilgrease 28
MIL-L-7870 Oil, General Purpose Low Temperature	Brayco 363 Caltex Low Temp. 'PQ' Rust Preventive No. 107 Winsor Lube L-1018
MIL-L-23398 — Lubricant, Solid Film, Air Drying (Resin, Moly Disulphide, Graphite)	Lubri-Bond A

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SERVICING (Continued)

G. **Lubrication Data**

See Figures 12-00-2 thru 12-00-11.

Zerk fittings not specifically referenced in these figures are for sealed bearings using lifetime lubricants. Lubrication is required only at time of component overhaul. Only the points indicated on the charts require periodic lubrication.

H. **Table of Approved Thrust Reverser Lubricants S/N 370-2 thru -6, -8, and -9**

SUPPLIER	MANUFACTURE DESIGNATION
Product Techniques 1153 North Stanford Los Angeles, CA 90059	PT-605 (Preferred)
Fel Pro Inc. Skokie, IL 60076	C-100 (First Alternate)
Electro-Film Corp. 7116 -21 Laurel Canyon Blvd. North Hollywood, CA 91605	Lubri-Bond A (Second Alternate)

I. **Table of Approved Thrust Reverser Lubricants — S/N 370-1, -7 and 380-1 and Subs**

SUPPLIER	MANUFACTURE DESIGNATION
Ram Chemicals 210 E. Alondra Blvd. Gardena, CA	Ram GS-3

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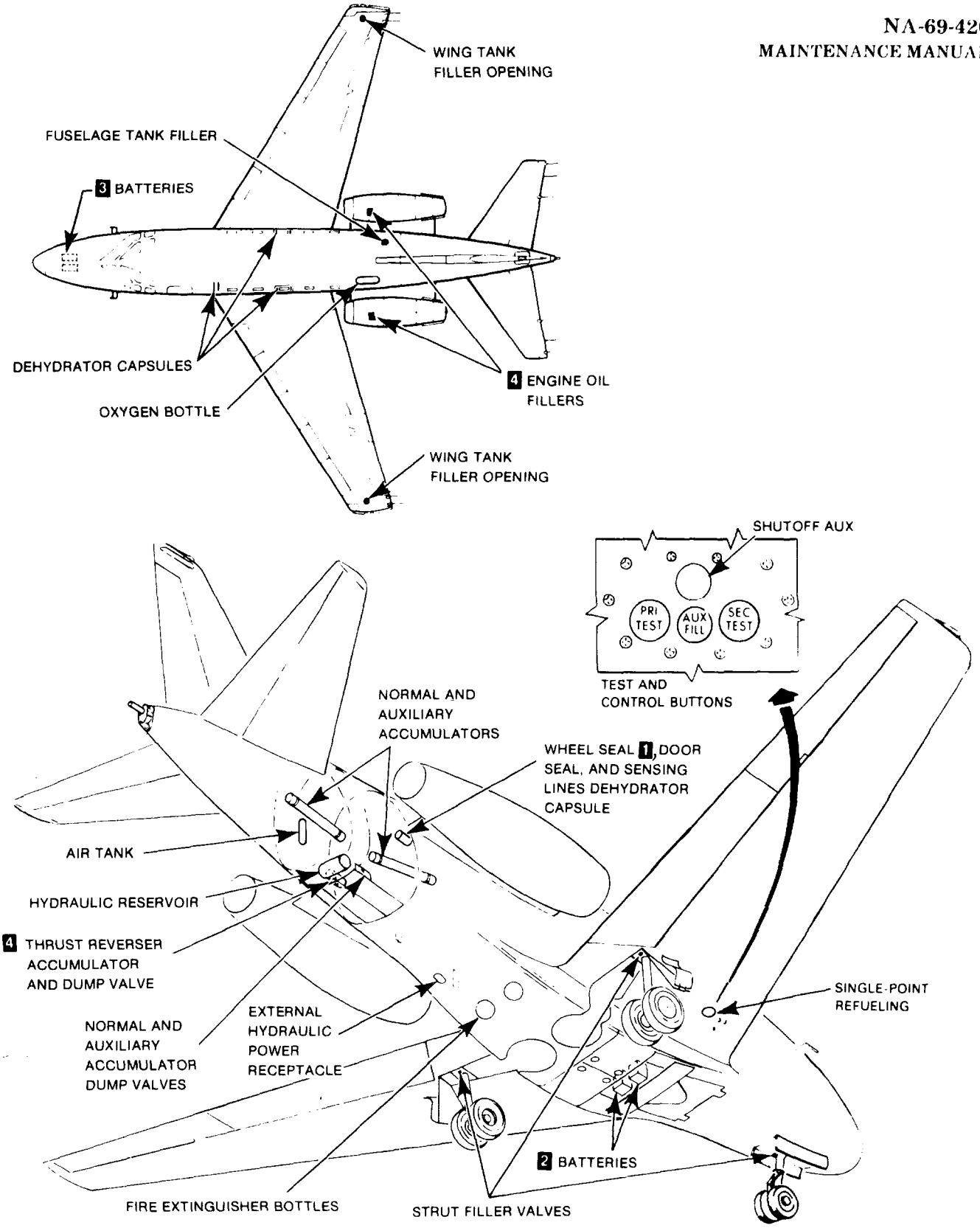

NOTES

1. *NA 265-70 Airplanes before S/B 72-8*
2. *S/N 370-1 thru -9*
3. *S/N 380-1 and subs*
4. *S/N 370-2 thru -6, -8, and -9*

Figure 12-00-1 — Servicing Diagram

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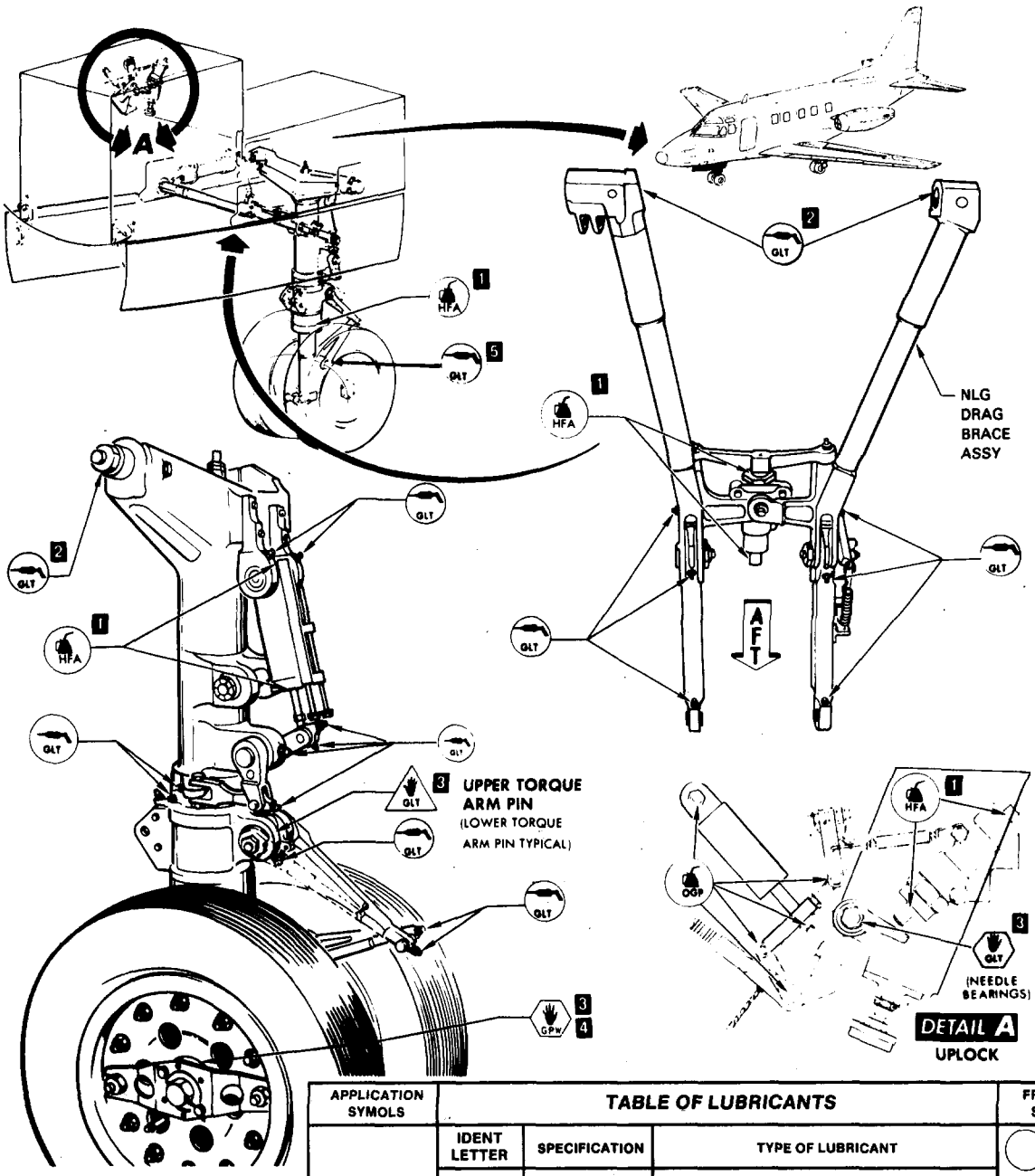

APPLICATION SYMOLS	TABLE OF LUBRICANTS			FREQUENCY SYMBOLS
	IDENT LETTER	SPECIFICATION	TYPE OF LUBRICANT	150 HOURS
GREASE GUN	GLT	MIL-G-81827	GREASE, GEAR AND ACTUATOR SCREW	
	OGP	MIL-L-7870	OIL, GENERAL PURPOSE LOW TEMP LUB	300 HOURS
	HFA	MIL-H-5606A	HYDRAULIC FLUID, AIRCRAFT, PETROLEUM BASE	
HAND				
OIL CAN	GPW	MIL-G-81322A	GREASE, GENERAL PURPOSE WIDE TEMP RANGE	SPECIAL

1 FELT WIPERS

2 GREASE FITTINGS ON STRUCTURE, RH AND LH SIDE

3 DISASSEMBLE, CLEAN AND REPACK AT 600 HOURS IF MIL-G-23827 GREASE IS USED AND NO GREASE FITTING IS INSTALLED. AIRPLANES WITH GREASE FITTING INSTALLED IN TORQUE ARM PINS AND USING MIL-G-81827 GREASE MAY PRESSURE LUBE AT 300 HOURS AND INSPECT AT STRUT RESEAL.

4 LUBRICATE WHEEL BEARINGS AT TIRE CHANGE

5 TORQUE LINK PIN 5010737-101

Figure 12-00-2 — Lubrication of Nose Gear (Sheet 1 of 2)

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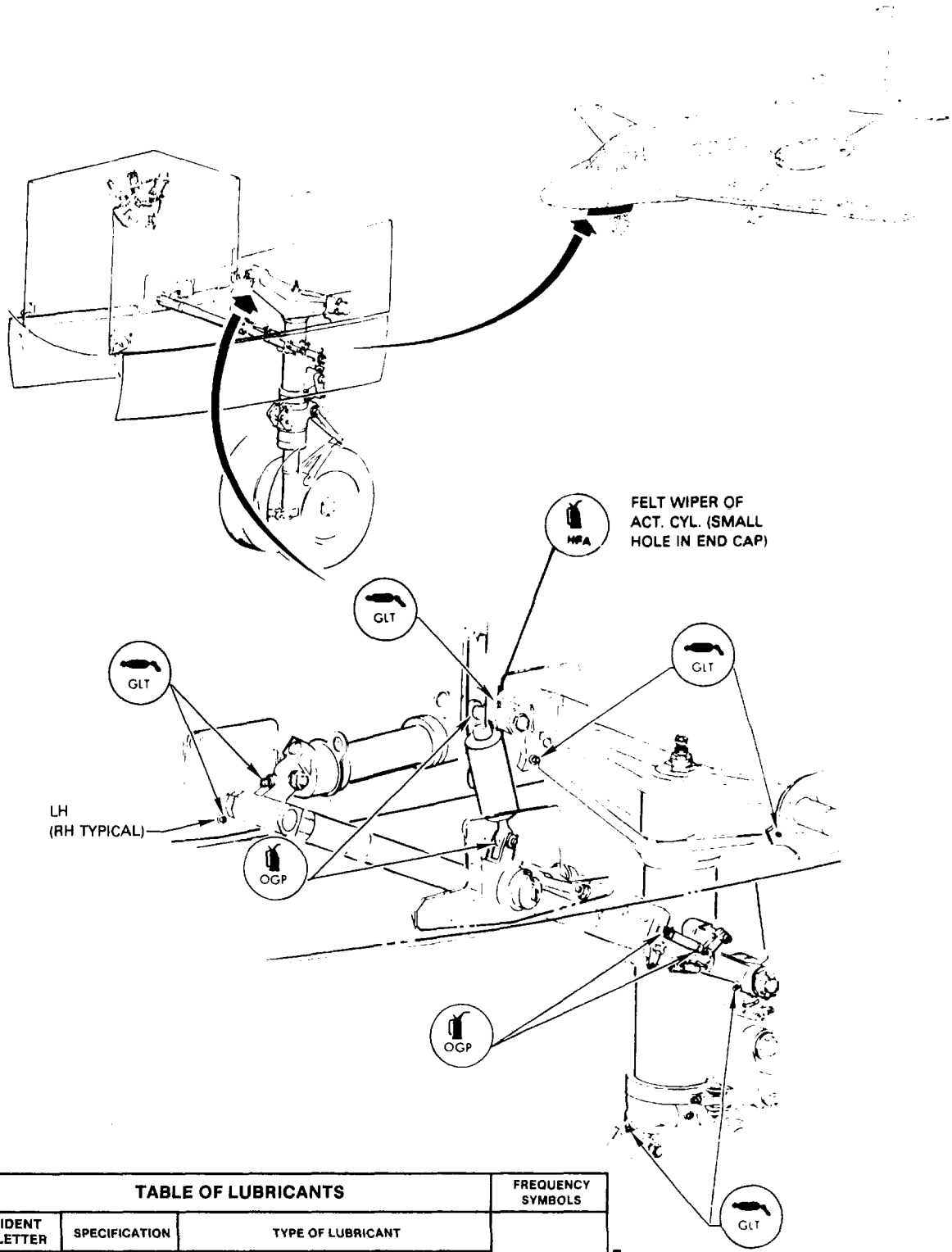

APPLICATION SYMOLS	TABLE OF LUBRICANTS			FREQUENCY SYMBOLS
	IDENT LETTER	SPECIFICATION	TYPE OF LUBRICANT	
GREASE GUN	GLT	MIL-G-81827	GREASE, GEAR AND ACTUATOR SCREW	
	OGP	MIL-L-7870	OIL, GENERAL PURPOSE LOW TEMP LUB	150 HOURS
OIL CAN	HFA	MIL-H-5606A	HYDRAULIC FLUID, AIRCRAFT, PETROLEUM BASE	

Figure 12-00-2 — Lubrication of Nose Gear (Sheet 2 of 2)

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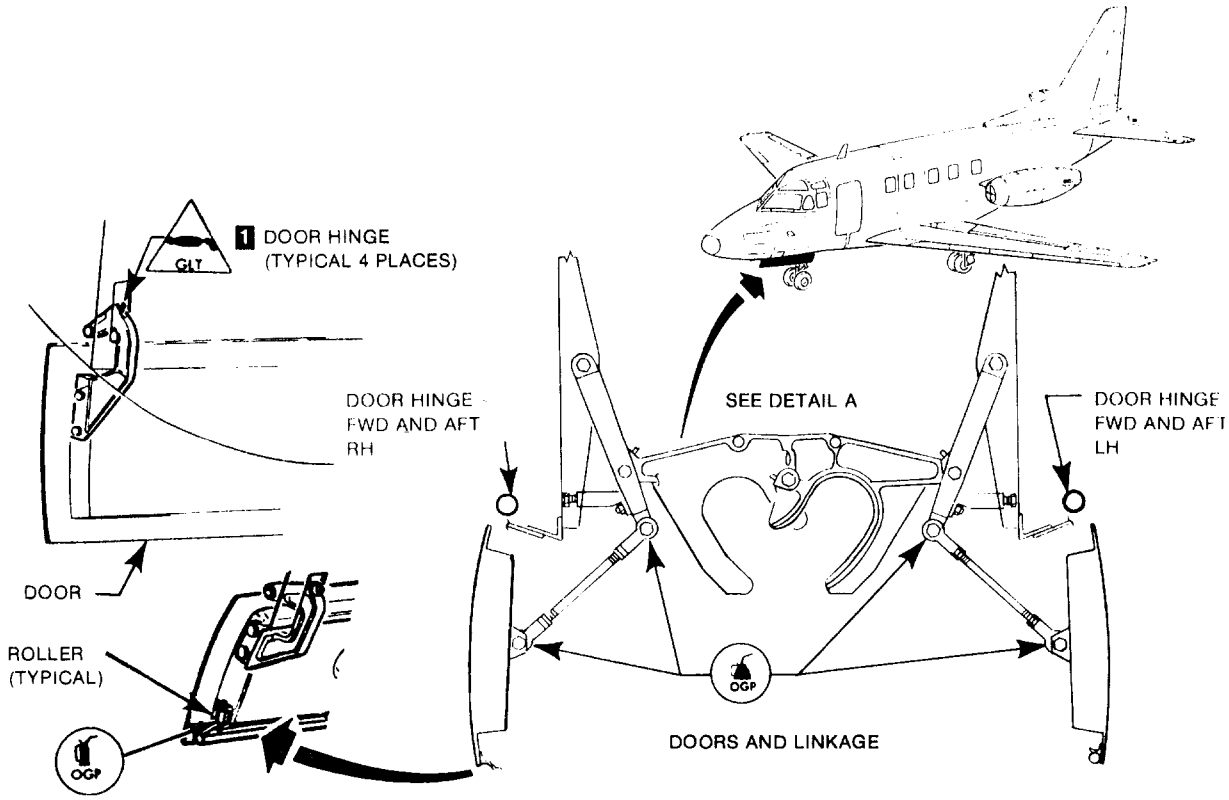

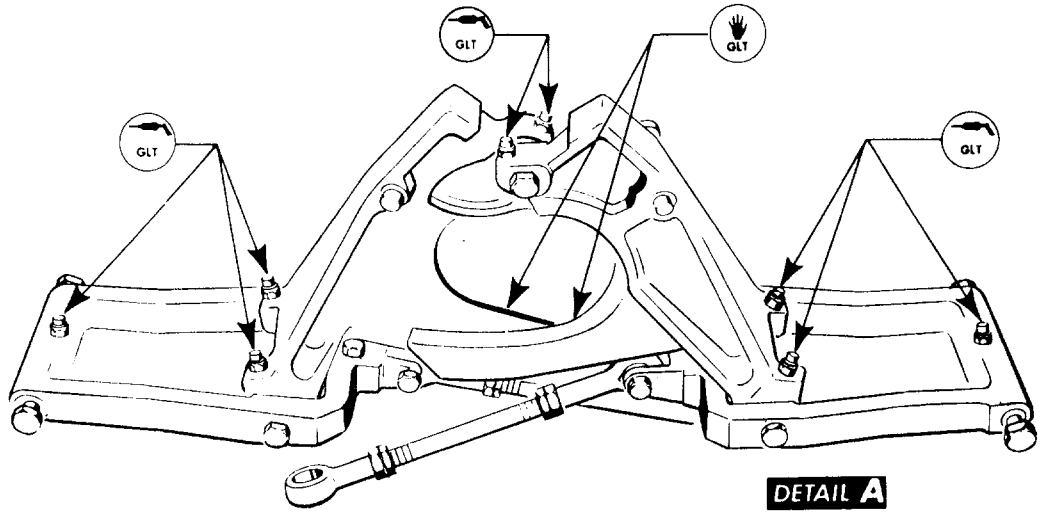

DETAIL A
DOOR ACTUATING LINKAGE
(SHOWN WITH GEAR UP AND DOOR CLOSED FOR CLARITY)

APPLICATION SYMOLS	TABLE OF LUBRICANTS			FREQUENCY SYMBOLS
	IDENT LETTER	SPECIFICATION	TYPE OF LUBRICANT	150 HOURS
GREASE GUN	GLT	MIL-G-81827	GREASE, GEAR AND ACTUATOR SCREW	
	OGP	MIL-L-7870	OIL, GENERAL PURPOSE LOW TEMP LUB	300 HOURS
HAND				
OIL CAN				

1 THE NOSE GEAR MUST BE PARTIALLY RETRACTED TO LUBRICATE THE RH AFT DOOR HINGE FITTING.

Figure 12-00-3 — Lubrication of Nose Gear Doors

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EQUIPMENT/TOOLS

- 28 VDC Electrical Power Source
- Suitable Cleaning Container
- MLG Actuator and Center Gear Door Actuator Safety Locks
- Set of Jacks (5-ton minimum)

CONSUMABLES

- Stoddard Solvent P-D-680 or Equivalent
- Grease, Gear and Actuator Screw — MIL-G 23827-GLT

PROCEDURE

NOTE

Procedure is similar for LH and RH assemblies.

1. Place airplane on jacks and raise enough to permit gear retraction.
2. Apply 28 VDC electrical power to airplane.
3. Install landing gear locks on MLG retract cylinders.
4. Remove two nuts retaining spring assembly.
5. Remove spring assembly and disassemble.
6. Clean parts with solvent and dry.
7. Inspect parts for defects, replace as required.
8. Lubricate parts with grease and assemble.
9. Install spring assembly with bolts and nuts and torque 20-25 in.-lbs.
10. Remove landing gear safety locks previously installed.
11. Perform a landing gear retraction test to verify correct operation of landing gear.
12. Turn off and disconnect 28 VDC electrical power to airplane.
13. Secure airplane as necessary.

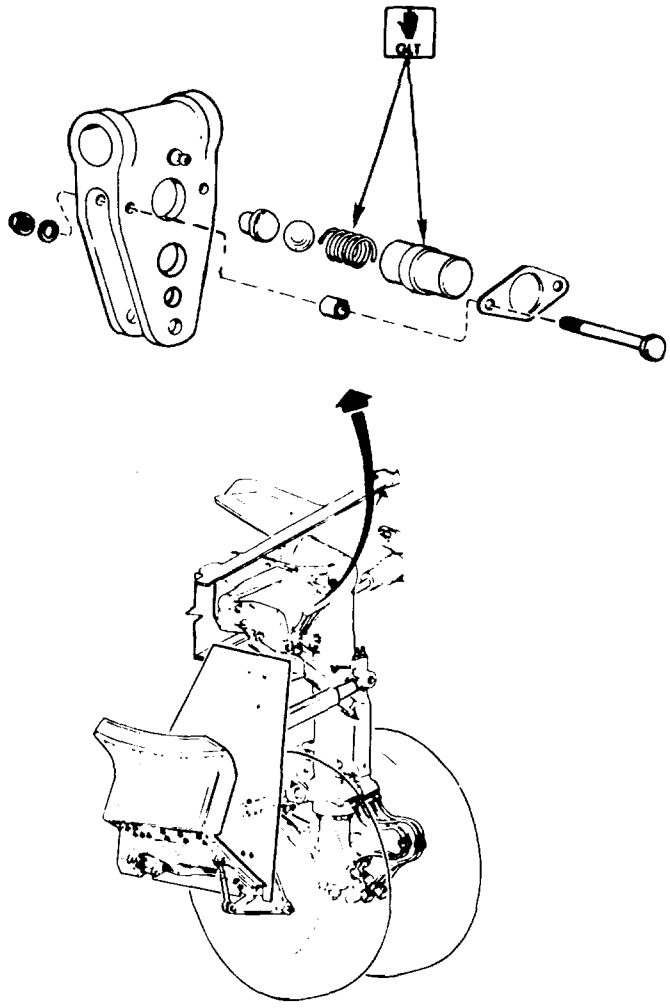

Figure 12-00-3A — Lubrication of Side Brace Idler Link Spring Assembly

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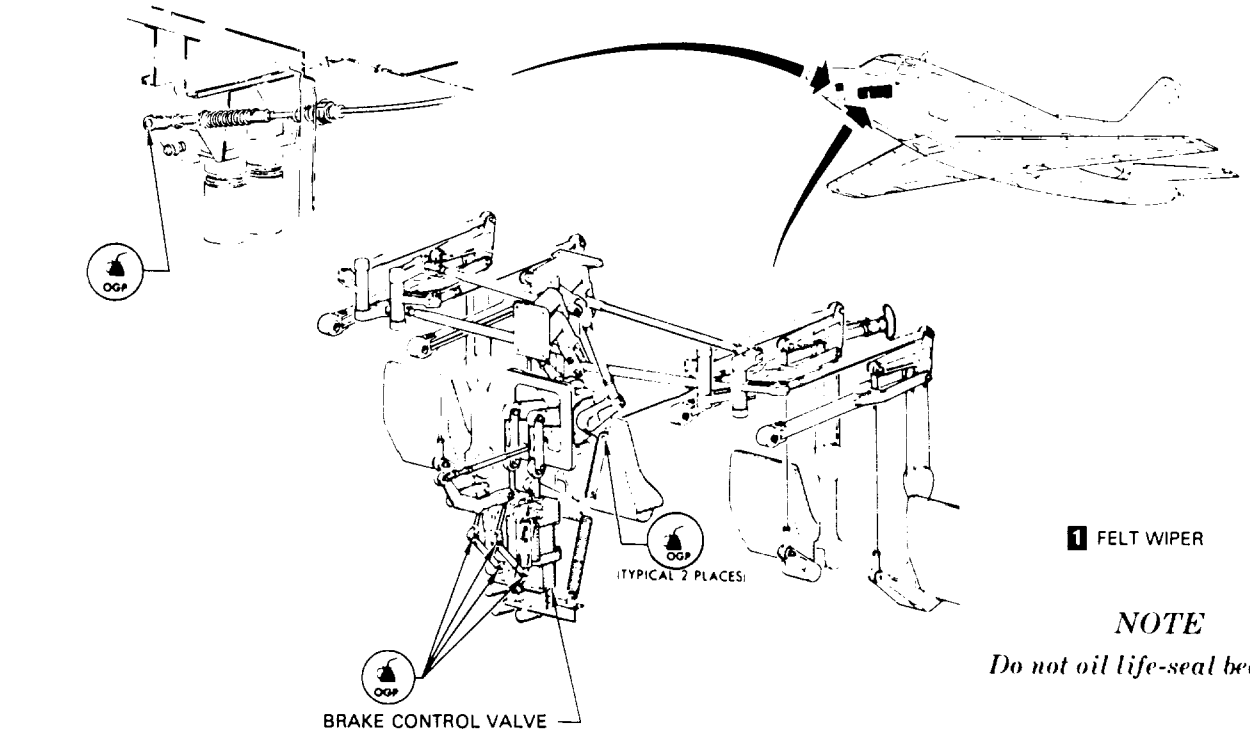

APPLICATION SYMOLS	TABLE OF LUBRICANTS			FREQUENCY SYMBOLS
	IDENT LETTER	SPECIFICATION	TYPE OF LUBRICANT	
GREASE GUN	GLT	MIL-G-81827	GREASE, GEAR AND ACTUATOR SCREW	150 HOURS
OIL CAN	OGP	MIL-L-7870	OIL, GENERAL PURPOSE LOW TEMP LUBE	
	HFA	MIL-H-5606A	HYDRAULIC FLUID, AIRCRAFT, PETROLEUM BASE	

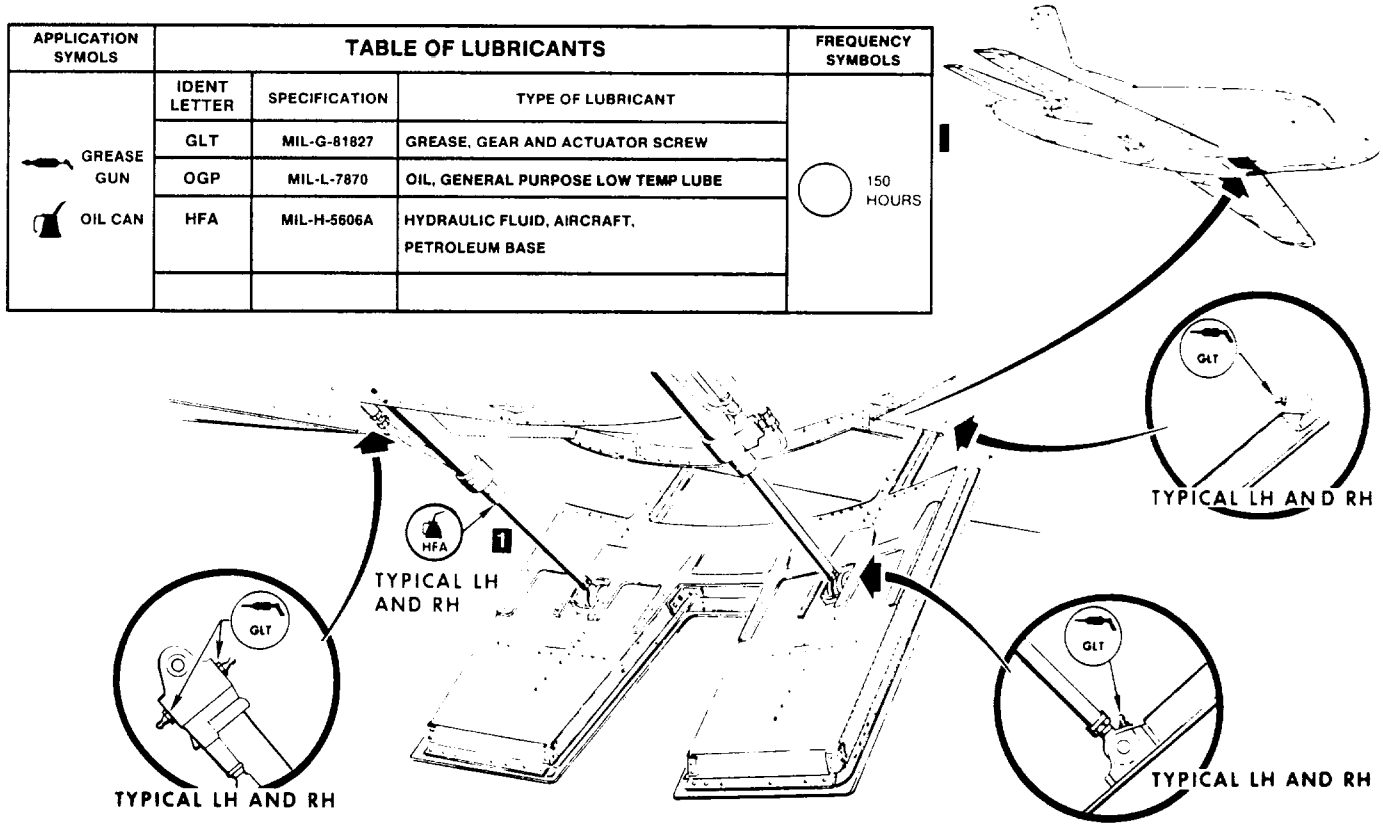

Figure 12-00-4 — Lubrication of Brake Control Valve and Speed Brake

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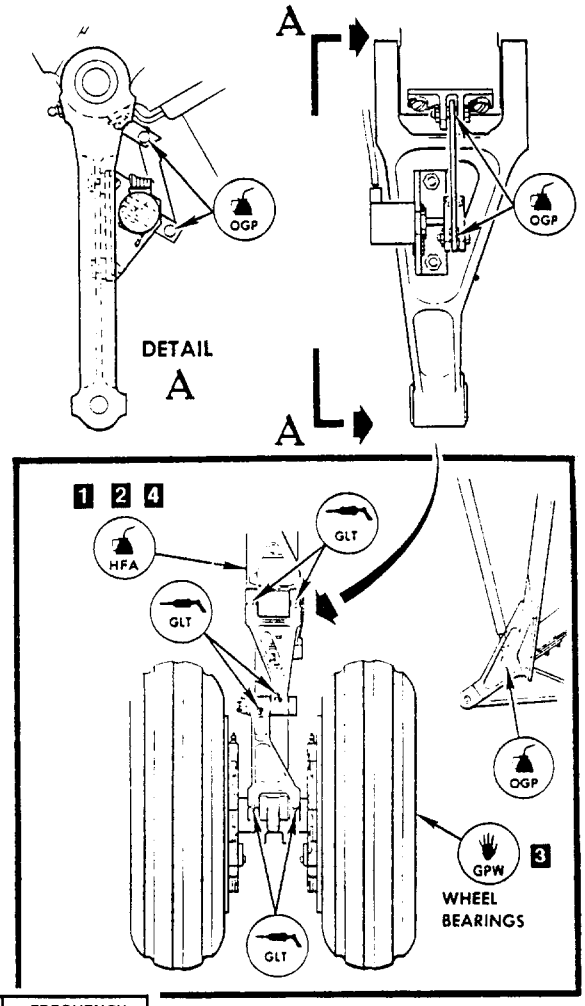
CAUTION

Movement of landing gear by means other than hydraulic pressure may result in partial evacuation of hydraulic fluid from the actuator. Movement by other means may result in damage to the actuator during subsequent landing gear operation.



APPLICATION SYMBOLS	TABLE OF LUBRICANTS			FREQUENCY SYMBOLS
	IDENT LETTER	SPECIFICATION	TYPE OF LUBRICANT	○ 150 HOURS
GREASE GUN	GLT	MIL-G-81827	GREASE, GEAR AND ACTUATOR SCREW	
	CPC	MIL-C-16173D (GRADE 1)	CORROSION PREVENTIVE COMPOUND	
HAND	OGP	MIL-L-7870	OIL, GENERAL PURPOSE LOW TEMP LUBE	△ 300 HOURS
	HFA	MIL-H-5606A	HYDRAULIC FLUID, AIRCRAFT, PETROLEUM BASE	
OIL CAN	GPW	MIL-G-81322A	GREASE, GENERAL PURPOSE WIDE TEMP RANGE	□ 600 HOURS

1. FELT WIPER
2. NAS516-1 FLUSH FITTING, USE STEWART-WARNER NOZZLE ADAPTER, P/N 314150 OR EQUIVALENT.
3. EACH TIME WHEEL IS REMOVED
4. LUBRICATE FLUSH FITTING (AFT INBD SIDE OF STRUT)
5. DISASSEMBLE AND LUBRICATE
6. PARTIALLY RETRACT AND SUPPORT MAIN GEAR; THEN REMOVE HYDRAULIC PRESSURE TO GAIN ACCESS TO FLUSH FITTING ON AFT SIDE OF DOWNLOCK PIN. LUBRICATE FITTING UNTIL GREASE IS OBSERVED BETWEEN PIN AND HOUSING.
7. SOME MLG CYLINDERS DO NOT HAVE GREASE ZERK PROVISIONS. IF GREASE PROVISIONS ARE NOT PROVIDED, THERE ARE NO LUBRICATION REQUIREMENTS.

Figure 12-00-5 — Lubrication of Main Landing Gear (Sheet 1 of 2)

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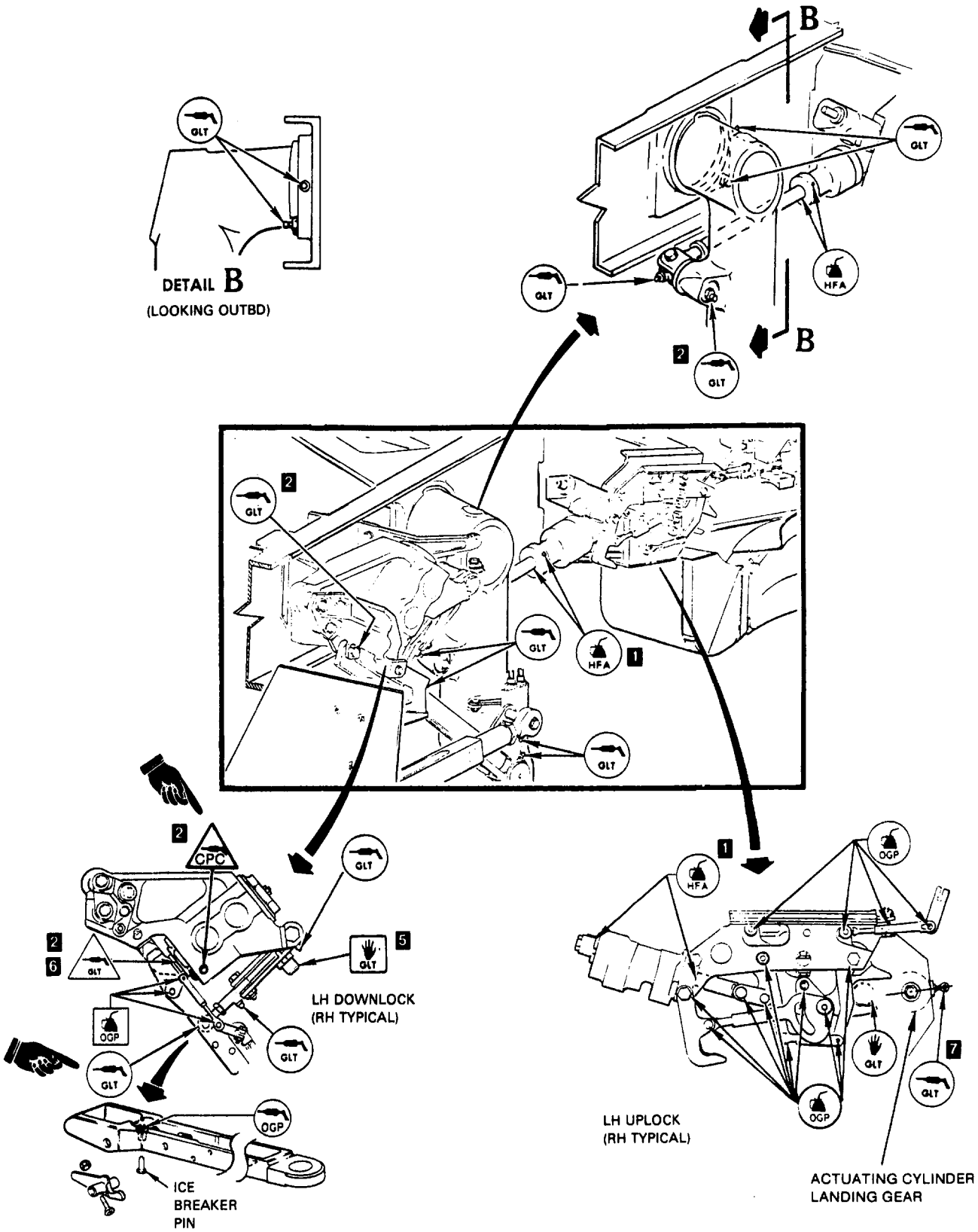

Figure 12-00-5 — Lubrication of Main Landing Gear (Sheet 2 of 2)

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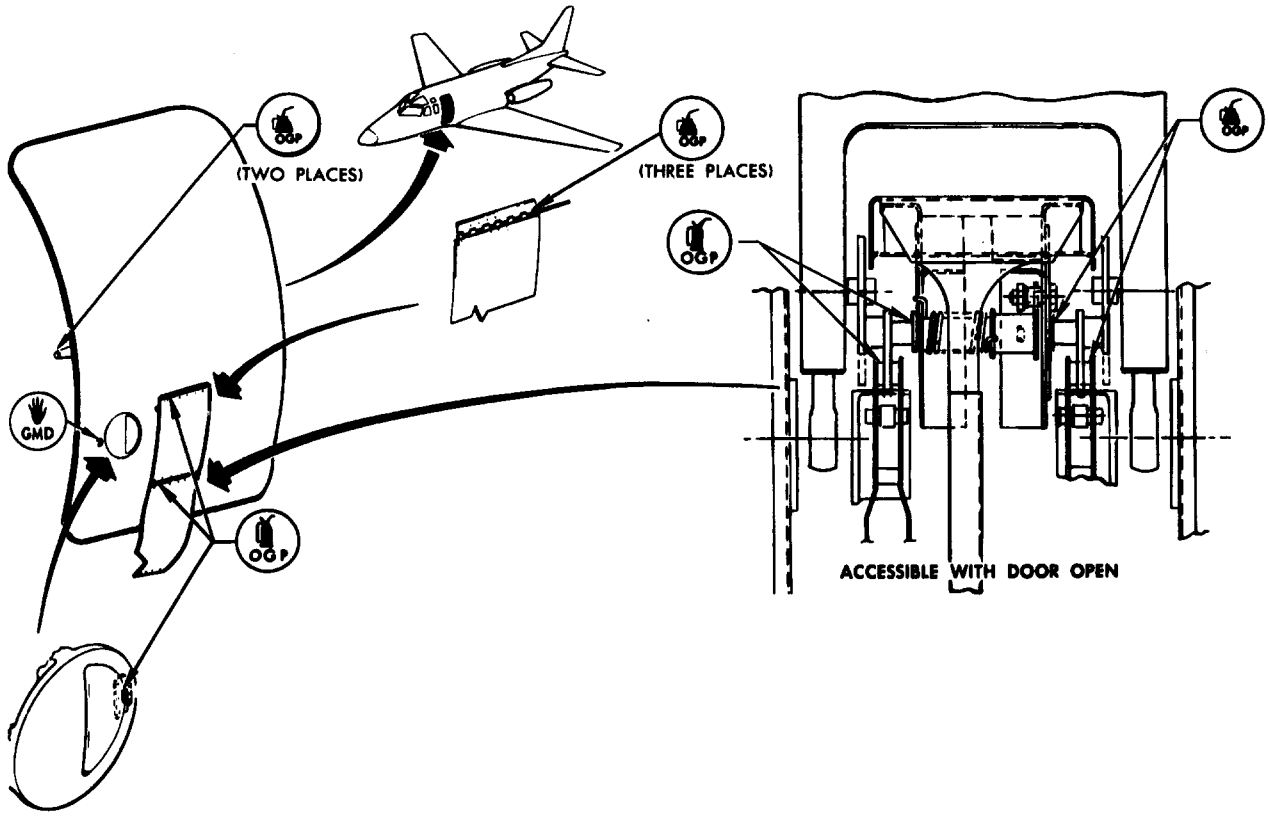

APPLICATION SYMBOLS	TABLE OF LUBRICANTS			FREQUENCY SYMBOLS
	IDENT LETTER	SPECIFICATION	TYPE OF LUBRICANT	
OIL CAN	OGP	MIL-L-7870	OIL, GENERAL PURPOSE LOW TEMP LUBE	150 HOURS
HAND	GMD	MIL-G-21164	GREASE, MOLYBDENUM DISULPHIDE (HIGH & LOW TEMP)	

Figure 12-00-6 — Lubrication of Main Entrance Door

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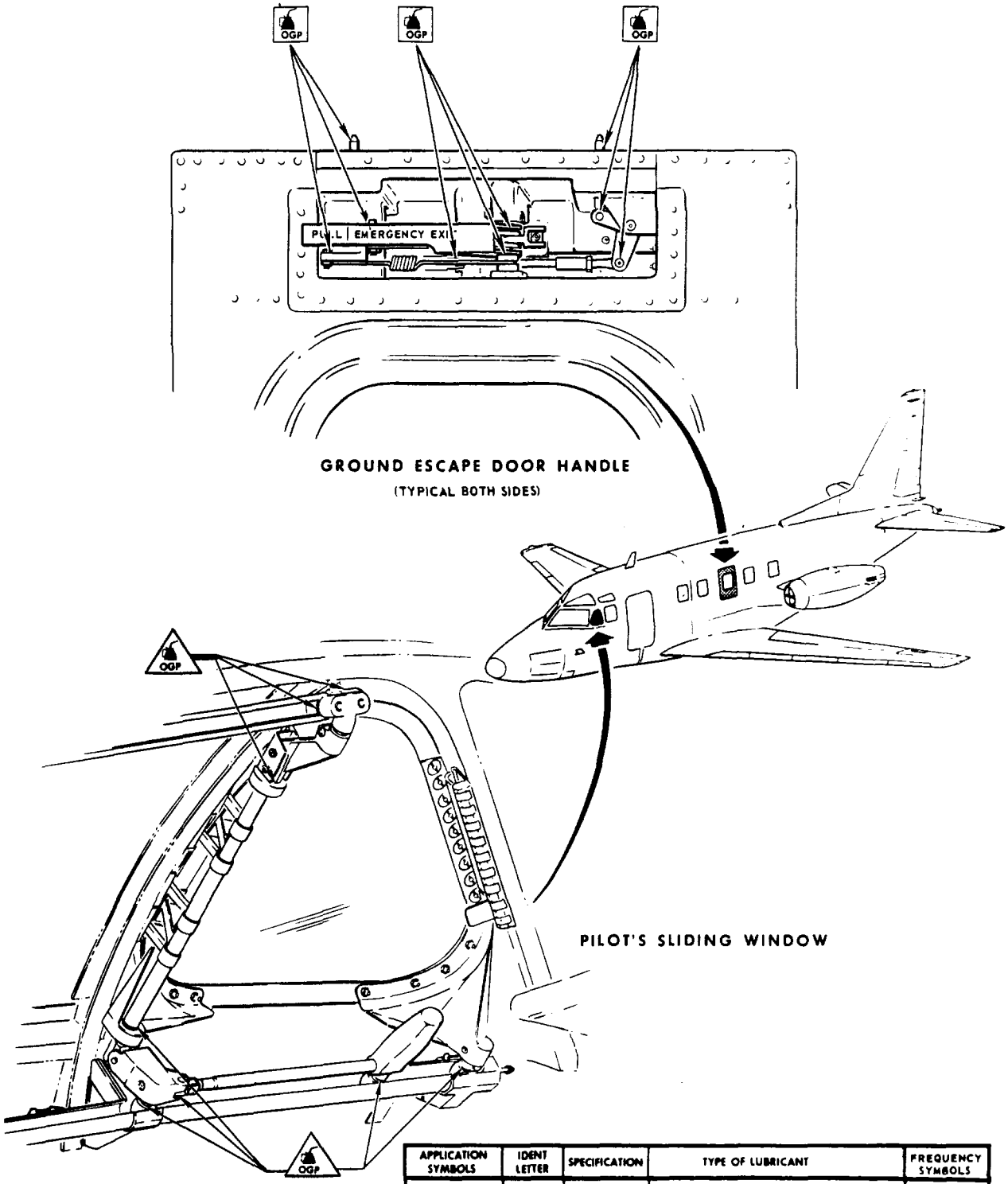

APPLICATION SYMBOLS	IDENT LETTER	SPECIFICATION	TYPE OF LUBRICANT	FREQUENCY SYMBOLS
OIL CAN	OGP	MIL-L-7870	OIL, GENERAL PURPOSE LOW TEMP LUB	□ 12 MONTHS △ 300 HOURS

Lubrication of Pilot's Sliding Window
and Ground Escape Hatch
Figure 12-00-7

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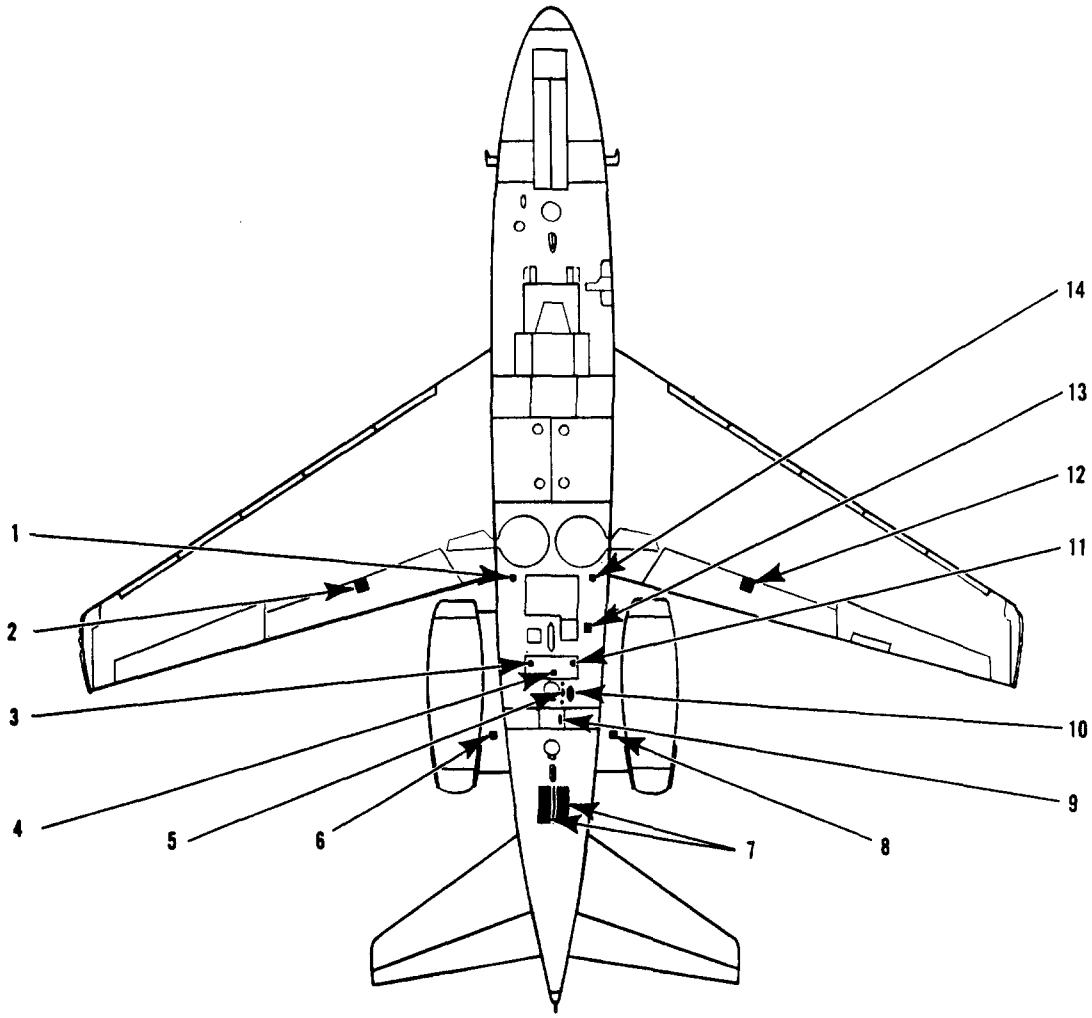

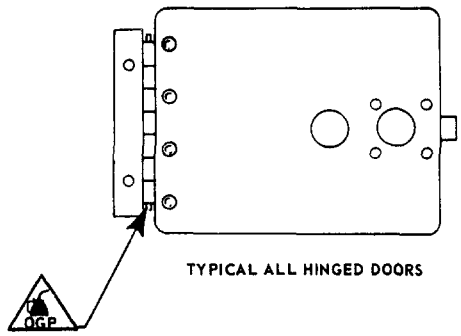

TYPICAL ALL HINGED DOORS

APPLICATION SYMBOLS	IDENT LETTER	SPEC	TYPE OF LUBRICANT	FREQUENCY SYMBOLS
OIL CAN	OGP	MIL-L-7870	OIL GENERAL PURPOSE LOW TEMP LUB	△ 300 HOURS

Lubrication of Access Doors
Figure 12-00-8

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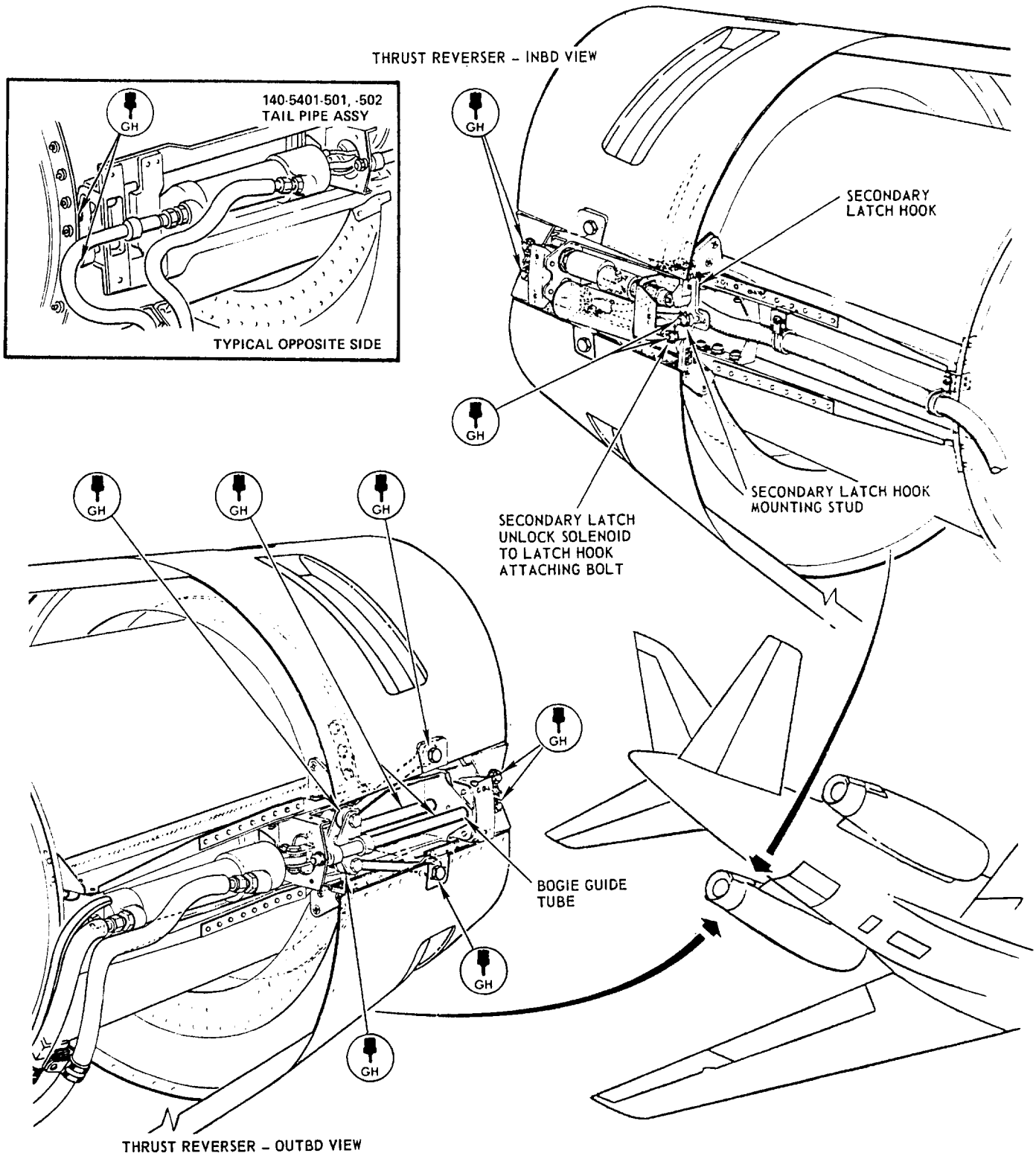

APPLICATION SYMBOLS	IDENT LETTER	SPECIFICATION	TYPE OF LUBRICANT	FREQUENCY SYMBOLS
BRUSH	GH	PRODUCTS TECHNIQUE PT605	GREASE, HIGH TEMP	150 HOURS

Figure 12-00-9 — Lubrication of Thrust Reverser S/N 370-2 thru -6, -8, and -9

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SERVICING (Continued)

Consumables

- Electrical Contact Cleaner — available through local electronics supplier
- Dry film lubricant — RAM GS-3 or equivalent (RAM Chemicals, 210 East Alondra Blvd., Gardena, CA)
- Methylethylketone (MEK)

To lubricate thrust reverser, proceed as follows:

1. Stow thrust reversers, and remove the two upper access doors on aft fairing.

CAUTION

On RH thrust reverser, pneumatic latches are oriented such that any cleaning solvent spilled on or applied to upper end of the pneumatic latches is trapped. Vent holes in the upper surface allow the cleaning solvent to enter the pneumatic latches, possibly damaging the pneumatic latches and infiltrating into other parts of the pneumatic system, damaging other components as well. Positively avoid getting cleaning solvent into the cupped area on top of the RH thrust reverser pneumatic latches.

2. Using a soft bristle brush, clean latch pivot point and roller with methylethylketone (MEK) or a suitable electrical contact cleaner.

CAUTION

Do not use RAM GS-3 on pneumatic latch stem, seals, or near air inlet port. RAM GS-3 is incompatible with lubricant applied to stem, seal, and internal parts during manufacture. When the two lubricants are mixed, a sticky gummy residue is formed that binds pneumatic latch piston and stem and damages seals.

The proper remedy for a binding pneumatic latch is to replace the faulty pneumatic latch with a serviceable unit. As a temporary measure until a serviceable unit is obtained, a sticking pneumatic latch can be cleaned then lubricated by spraying Permatex No. 116DA lubricant (Woodhill Chemical Co., 18731 Greenwood Parkway, Cleveland OH 41128) into the air inlet port and around the stem of the piston.

3. Allow latch and roller to dry, then apply an adequate amount of dry film lubricant (RAM GS-3 or equivalent) to latch pivot point and roller.
4. Deploy thrust reversers.
5. Lubricate rack and beam assemblies of the translating structure with dry film lubricant (RAM GS-3 or equivalent).
6. Stow thrust reverser and install access doors.

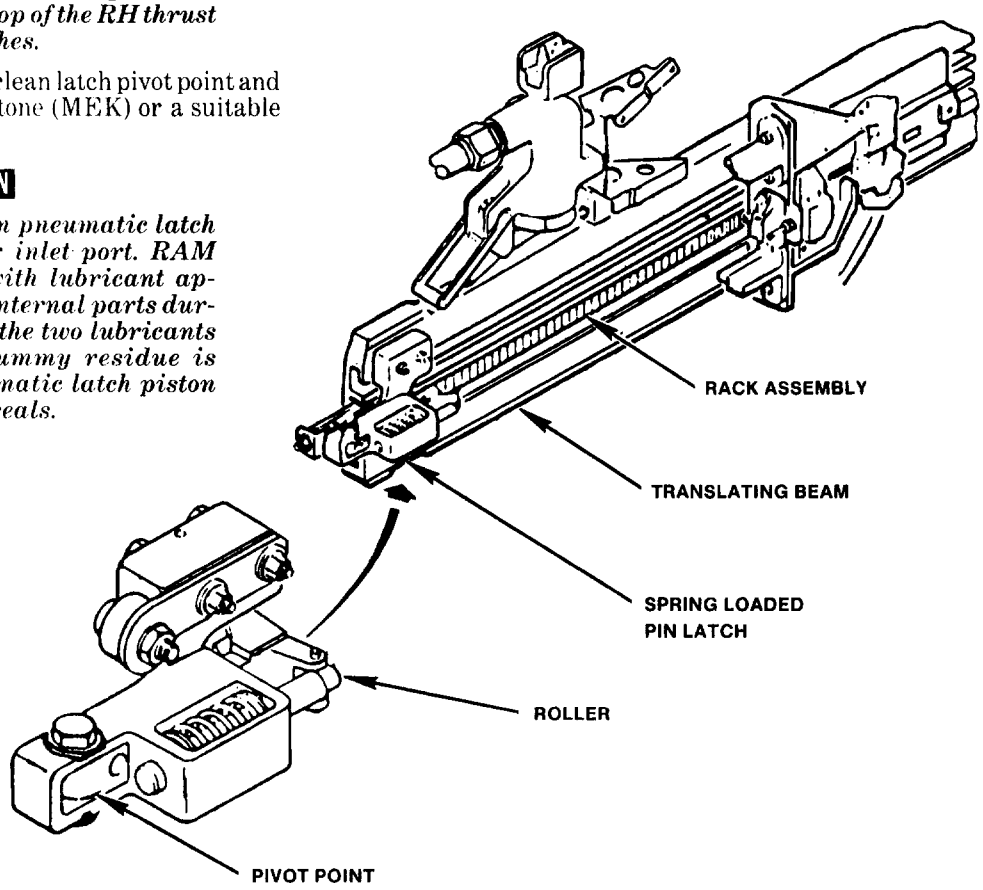

Figure 12-00-10 — Lubrication of Thrust Reverser S/N 370-1 and -7 and 380-1 and Subs

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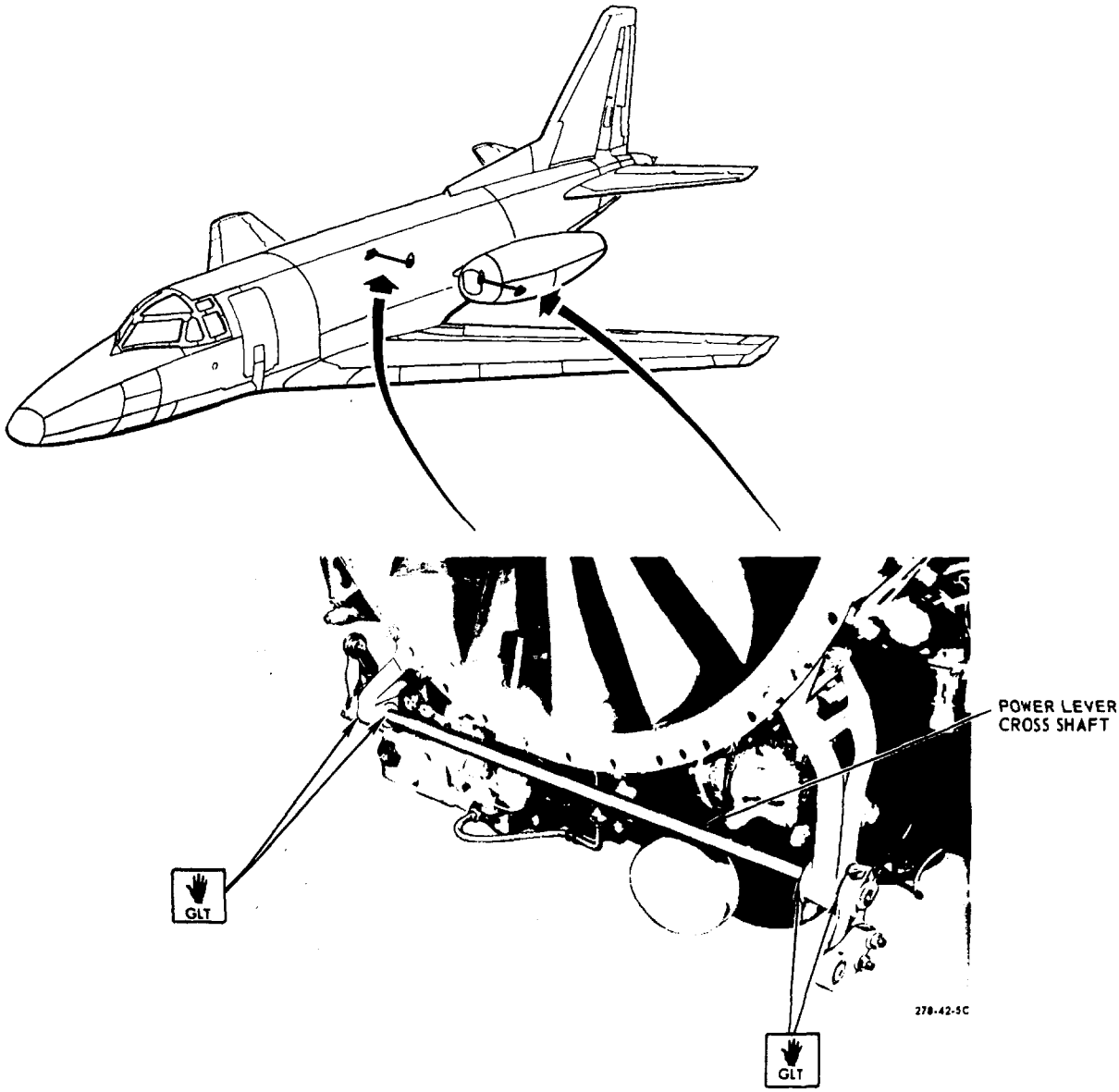

APPLICATION SYMBOLS	TABLE OF LUBRICANTS			FREQUENCY SYMBOLS
	IDENT LETTER	SPECIFICATION	TYPE OF LUBRICANT	
(HAND)	GLT	MIL-G-81827	GREASE, GEAR AND ACTUATOR SCREW	60 HOURS

Figure 12-00-11 — Lubrication of Power Cross Shaft S/N 370-2 thru -6, -8, and -9

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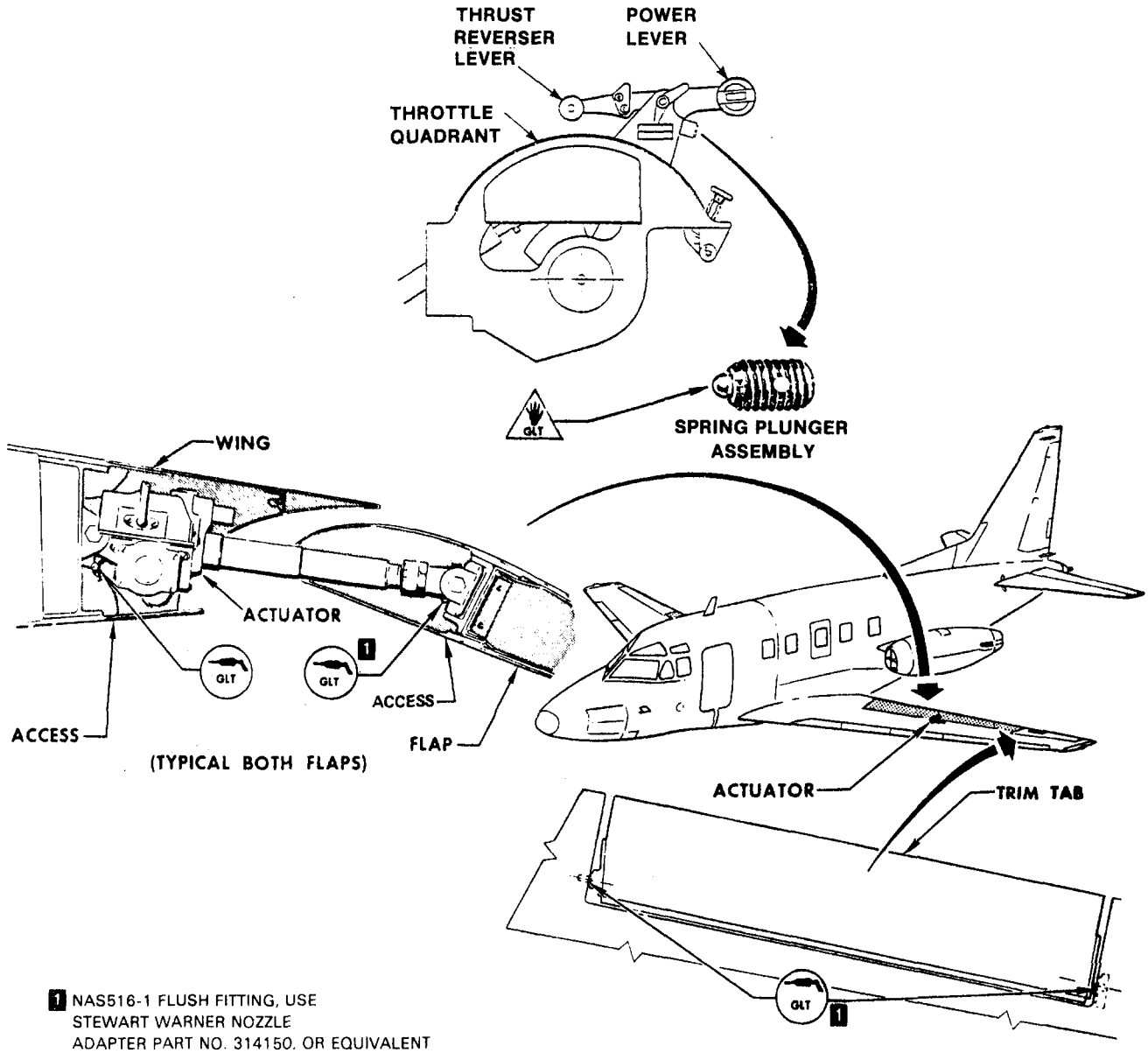

1 NAS516-1 FLUSH FITTING, USE STEWART WARNER NOZZLE ADAPTER PART NO. 314150, OR EQUIVALENT

APPLICATION SYMOLS	TABLE OF LUBRICANTS			FREQUENCY SYMBOLS
	IDENT LETTER	SPECIFICATION	TYPE OF LUBRICANT	150 HOURS
GREASE GUN HAND	GLT	MIL-G-81827	GREASE, GEAR AND ACTUATOR SCREW	
				300 HOURS

Figure 12-00-12 — Lubrication of Throttle Quadrant, Flaps, and Aileron Trim Tabs
S/N 370-7 and 380-1 and Subs

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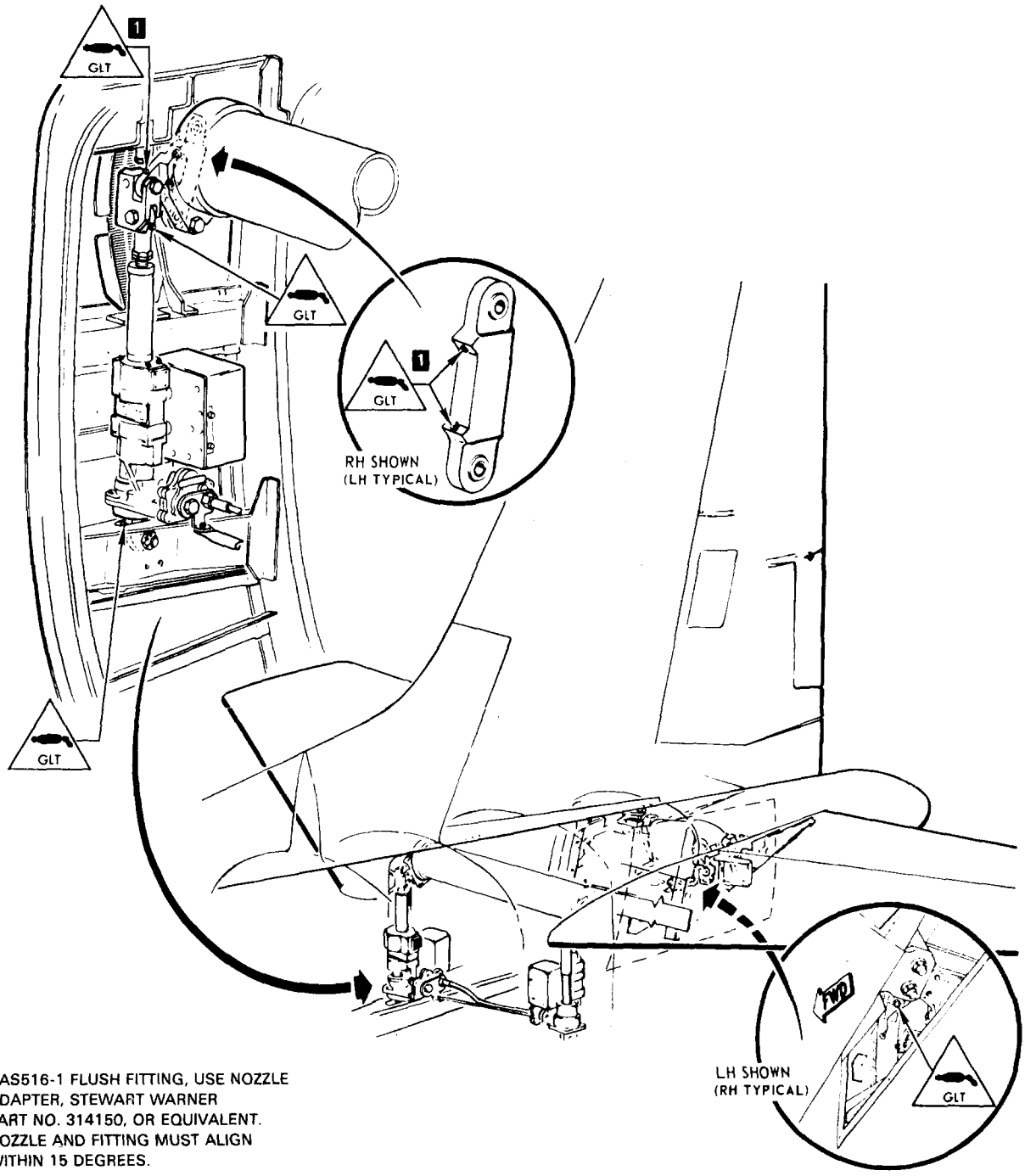

1 NAS516-1 FLUSH FITTING, USE NOZZLE ADAPTER, STEWART WARNER PART NO. 314150, OR EQUIVALENT. NOZZLE AND FITTING MUST ALIGN WITHIN 15 DEGREES.

APPLICATION SYMOLS	TABLE OF LUBRICANTS			FREQUENCY SYMBOLS
	IDENT LETTER	SPECIFICATION	TYPE OF LUBRICANT	
GREASE GUN	GLT	MIL-G-81827	GREASE, GEAR AND ACTUATOR SCREW	△ 300 HOURS

Figure 12-00-13 — Lubrication of Horizontal Stabilizer

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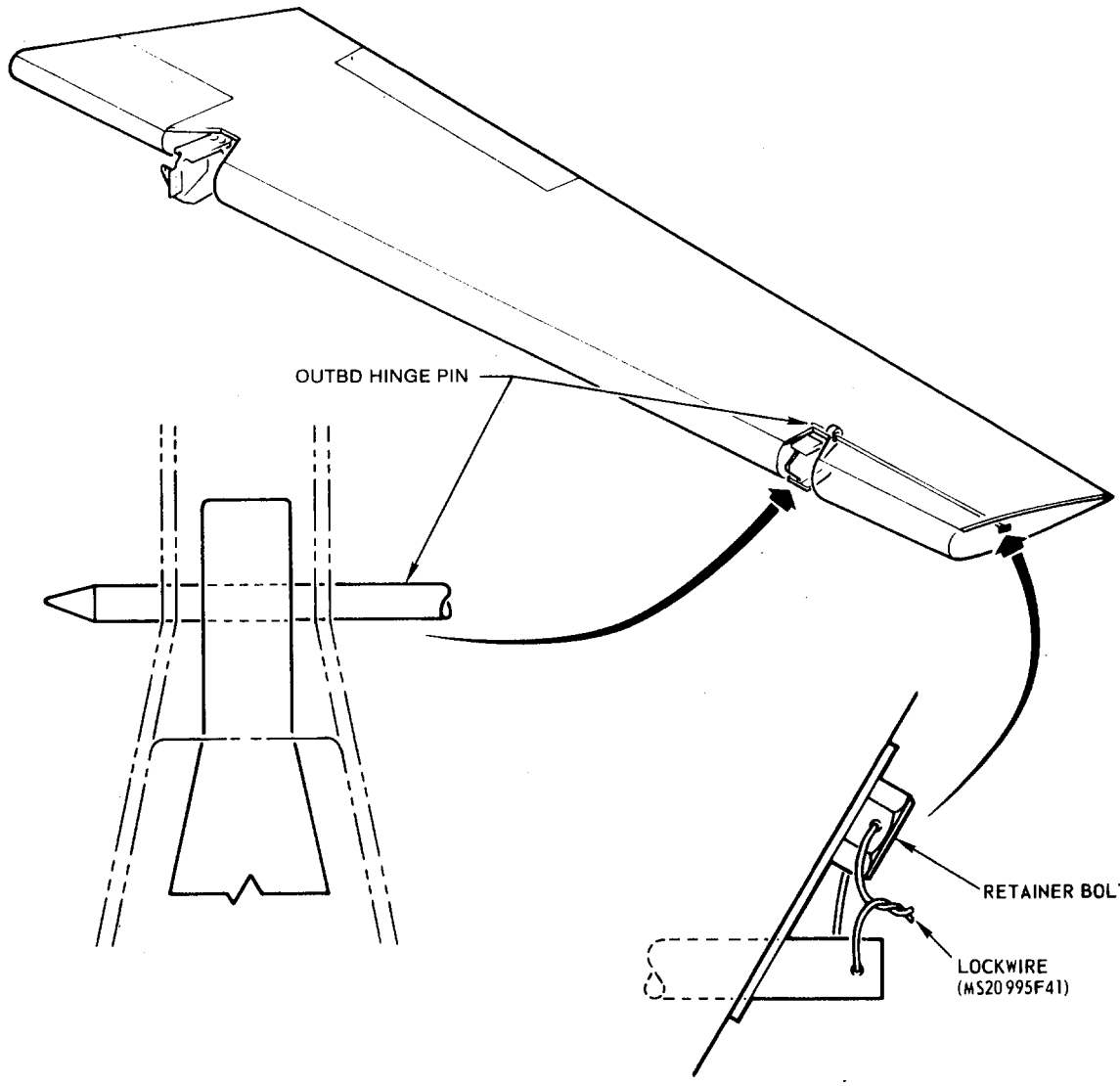

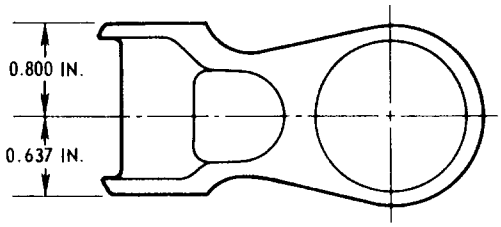

OUTBD AILERON HINGE LINK

NOTES

- ***During installation of the outboard aileron hinge link, ensure that link is not installed upside down. Misalignment of the aileron to the wing mold line will result if link is improperly installed.***
- ***Inspect hinge pin for corrosion, cracks, wear, and damage before installation. Lubricate with MIL-G-81322 grease. Install hinge pin.***

Figure 12-00-14 — Lubrication of Aileron Hinge Pin

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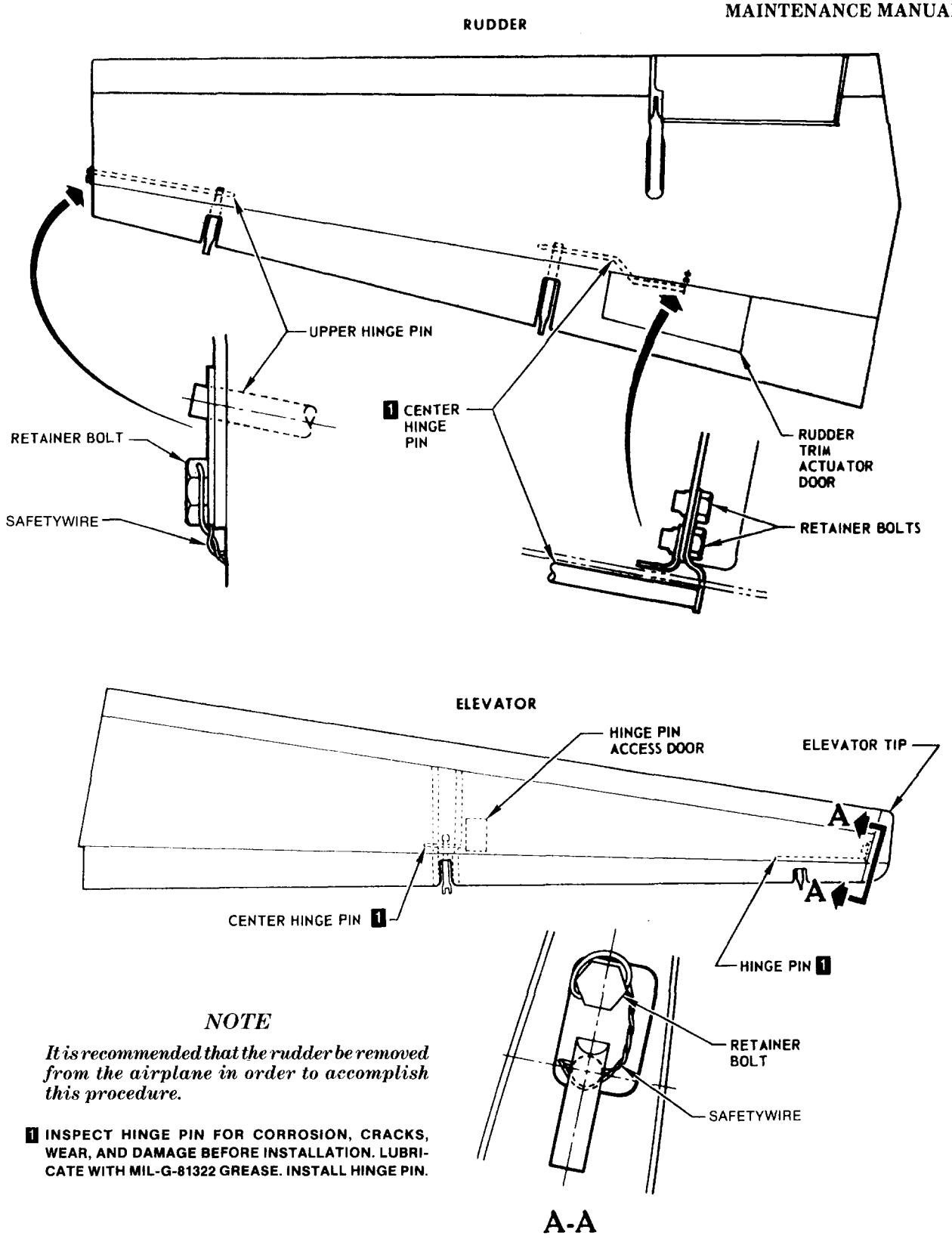

NOTE

It is recommended that the rudder be removed from the airplane in order to accomplish this procedure.

1 INSPECT HINGE PIN FOR CORROSION, CRACKS, WEAR, AND DAMAGE BEFORE INSTALLATION. LUBRICATE WITH MIL-G-81322 GREASE. INSTALL HINGE PIN.

Figure 12-00-15 — Lubrication of Rudder and Elevator Hinge Pin

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AILERON HINGE BEARING INSPECTION

Consumables

Grease, MIL-G-81332A

Procedure

NOTE

Inspection procedures for LH and RH aileron hinge bearings are identical.

1 Remove aileron and hinge links. (See chapter 27.)

2 Inspect inbd and outbd bearings for the following and replace as required:

- Excessive roughness
- Security
- Excessive tightness
- Dislodged or missing seals/retainers

NOTE

Self aligning bearings, when rotated in a misaligned position, may exhibit a tightness and/or roughness in excess of that exhibited when bearing is rotated in an aligned position. This additional tightness and/or roughness is not cause for bearing replacement.

3 Lightly coat inner bore of bearings with grease.

4 Install aileron and hinge links. (See chapter 27.)

COMPONENT LOCATION

Index No.	Bearing Nomenclature
1	Aileron, inbd hinges
2	Aileron, outbd hinges
3	Rudder, top support hinge
4	Rudder, center support hinge
5	Rudder, top of torque tube
6	Rudder, bottom of torque tube
7	Elevator, outbd hinges
8	Elevator, center hinges
9	Elevator, inbd hinges
10	Elevator, torque tube assy, fwd
11	Elevator, torque tube assy, aft
12	Elevator, torque tube assy, bottom
13	Elevator, torque tube assy, top

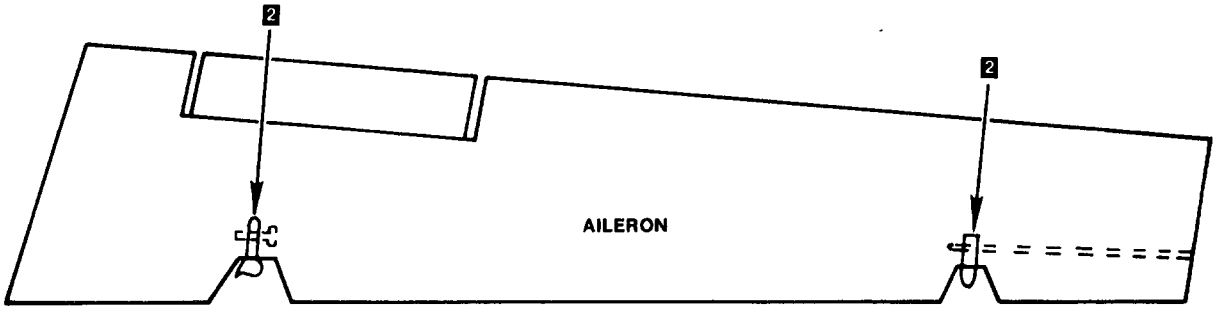

Figure 12-00-16 — Inspection of Flight Control Bearings (Sheet 1 of 3)

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RUDDER BEARINGS INSPECTION

Consumables

Grease, MIL-G-81332A

Procedure

1 Remove rudder and support hinges. (See chapter 27.)

2 Inspect bearings for the following and replace as required:

- Excessive roughness
- Security
- Excessive tightness
- Dislodged or missing seals/retainers

NOTE

Self aligning bearings, when rotated in a misaligned position, may exhibit a tightness and/or roughness in excess of that exhibited when bearing is rotated in an aligned position. This additional tightness and/or roughness is not cause for bearing replacement.

3 Lightly coat inner bore of bearings with grease.

4 Install rudder and support hinges. (See chapter 27.)

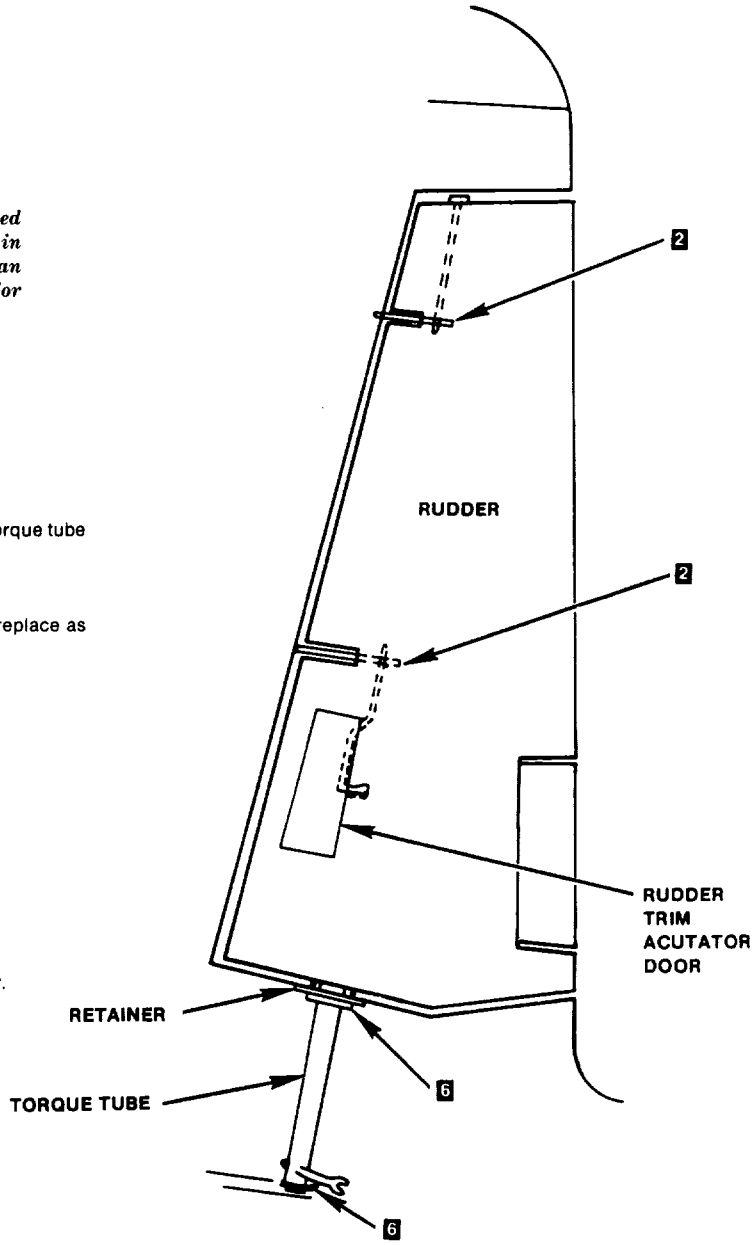
5 Remove fuselage access doors, remove retainer and lift torque tube enough to inspect bearings.

6 Inspect top and bottom bearings for the following and replace as required:

- Excessive roughness
- Security
- Excessive tightness
- Dislodged or missing seals/retainers

7 Lightly coat inner bore of bearings with grease.

8 Reinstall torque tube, retainer, and fuselage access door.



MAJOR CHANGE

Inspection of Flight Control Bearings
Figure 12-00-16 (Sheet 2 of 3)

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ELEVATOR BEARINGS INSPECTION

Consumables

Grease, MIL-G-81332A

Procedure

NOTE

Inspection for LH and RH elevator bearings are the same.

1 Remove elevator and hinge links. (See chapter 27.)

2 Inspect outbd and center bearings for the following and replace as required:

- Excessive roughness
- Security
- Excessive tightness
- Dislodged or missing seals/retainers

NOTE

Self aligning bearings, when rotated in a misaligned position, may exhibit a tightness and/or roughness in excess of that exhibited when bearing is rotated in an aligned position. This additional tightness and/or roughness is not cause for bearing replacement.

3 Lightly coat inner bore of bearings with grease.

4 Install elevator and hinge links. (See chapter 27.)

5 Remove fuselage access doors, vertical bolt and bushing from elevator torque tube assy, and through bolt. Separate hinge support section from tube assy and inbd hinge support.

6 Inspect inbd hinge bearing and four elevator torque tube assy bearings for the following and replace as required.

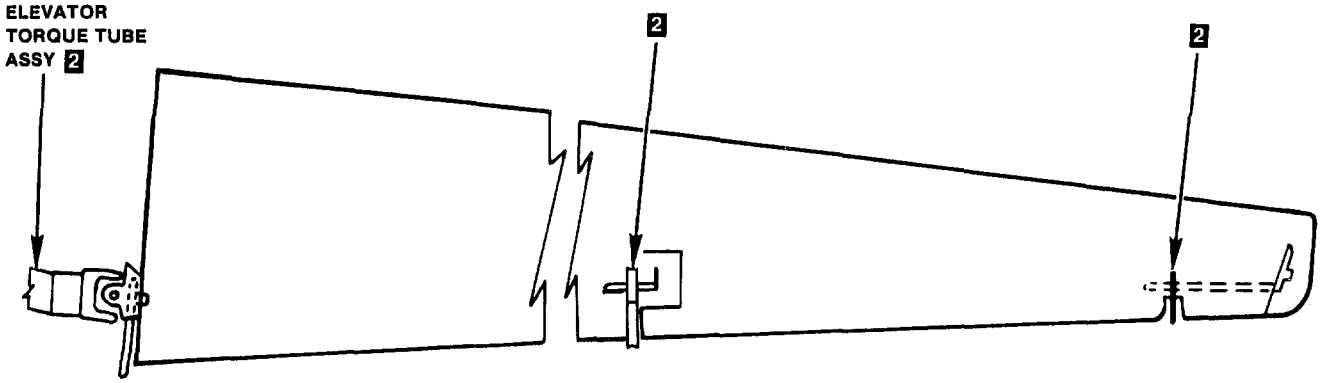
- Excessive roughness
- Security
- Excessive tightness
- Dislodged or missing seals/retainer

NOTE

Self aligning bearings, when rotated in a misaligned position, may exhibit a tightness and/or roughness in excess of that exhibited when bearing is rotated in an aligned position. This additional tightness and/or roughness is not cause for bearing replacement.

7 Lightly coat inner boare of bearings with grease.

8 Reinstall hinge support section to elevator torque tube assy and fuselage access doors. (See chapter 27.)



MAJOR CHANGE

Inspection of Flight Control Bearings
Figure 12-00-16 (Sheet 3 of 3)

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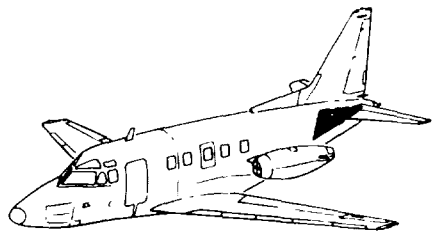


LUBRICATE — ELEVATOR PUSHRODS

Equipment/Tools

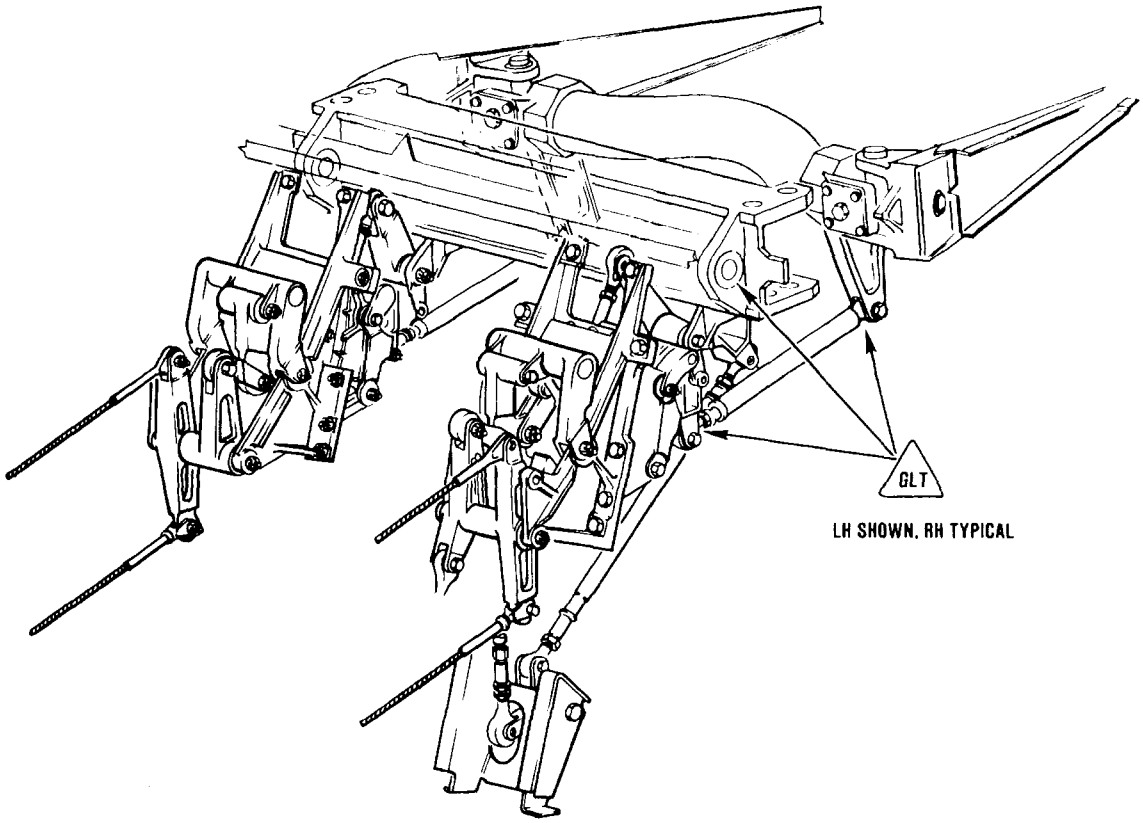
- Grease, Gun — with high-pressure fitting

Consumables

- Grease — Gear And Actuator Screw — MIL-G-81827 — GLT

PROCEDURE

Lubricate as illustrated.



APPLICATION SYMBOLS	TABLE OF LUBRICANTS			FREQUENCY SYMBOLS
	IDENT LETTER	SPECIFICATION	TYPE OF LUBRICANT	
GREASE GUN	GLT	MIL-G-81827	GREASE, GEAR AND ACTUATOR SCREW	△ 300 HOURS

Figure 12-00-17 — Lubricate Elevator Pushrods

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LUBRICATE — AILERON SECTOR PUSHRODS

Equipment/Tools

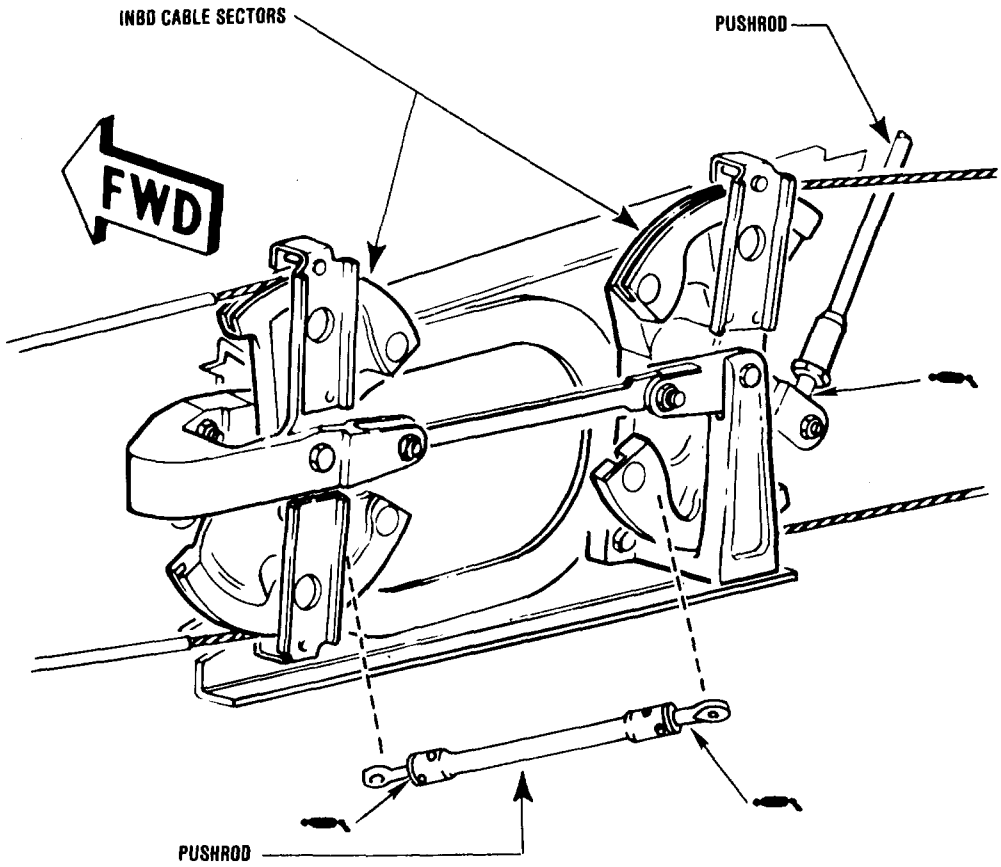
- Grease, Gun — with high-pressure needle fitting

Consumables

- Grease — Gear And Actuator Screw — MIL-G-81827 — GLT

PROCEDURE

Lubricate as illustrated.



APPLICATION SYMBOLS	TABLE OF LUBRICANTS		
GREASE GUN	IDENT LETTER	SPECIFICATION	TYPE OF LUBRICANT
	GLT	MIL-G-81827	GREASE, GEAR AND ACTUATOR SCREW

NEW

Figure 12-00-18 — Lubricate Aileron Sector Pushrods

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SERVICING TIRES

Servicing Nose Gear Tires

NOTES

- ***It is recommended that only nitrogen be used for tire inflation to preclude possible condensation build-up with air inflation; nitrogen will not sustain combustion and will reduce degradation of innerliner material due to oxidation.***
- ***It is recommended that dry talc or a water-free lubricant be used as an aid for installation of tire during wheel assembly.***

Inflation pressure for nose gear tires is 75 (+5, -0) psi.

Servicing Main Gear Tires

Inflation pressure for main gear tires is 165 (+5, -0) psi for S/N 370-2 thru -6, -8 and -9 and 180 (+5, -0) psi for 370-1 and -7, 380-1 and subs.

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COMBINED AIR AND FLUID SERVICING - MAINTENANCE PRACTICES

1. Servicing Main and Nose Landing Gear Shock Struts

A. Cleaning Landing Gear Shock Struts

(1) Clean shock strut piston polished surfaces and landing gear actuating cylinders exposed piston rod surfaces by wiping with a soft clean cloth saturated with hydraulic fluid, Specification MIL-H-5606A.

B. Main Gear Shock Strut Filling Procedure

(1) Jack airplane, using outer wing and fuselage jack points. (Refer to Chapter 7.)

(2) Remove air filler valve as follows:

(a) Slowly loosen swivel nut until air is released at desired rate.

(b) Cut lockwire and remove complete air filler valve assembly.

(3) Using a 5-ton axle jack at main gear jack point, fully compress strut and fill with hydraulic fluid (Specification MIL-H-5606A) to level of filler hole.

(4) Operate axle jack to fully extend and compress strut three times to bleed off any entrapped air. Check fluid level and refill if necessary.

(5) Replace air filler valve assembly and torque to 100-110 inch-pounds and safety with lockwire.

(6) Inflate strut to correct dimension. (Refer to procedure for inflating shock struts.)

C. Nose Gear Shock Strut Filling Procedure

(1) Jack airplane. (Refer to Chapter 7.)

(2) Remove air filler valve. (Refer to main gear shock strut filling procedure.)

(3) Fully compress strut and fill to level of filler hole with Specification MIL-H-5606A hydraulic fluid.

(4) Extend and compress strut three or four times to bleed off any entrapped air. Check fluid level and refill if necessary.

(5) Replace air filler valve and torque to 100-110 inch-pounds. Safety valve assembly with lockwire.

(6) Inflate strut to correct dimension. (Refer to procedure for inflating shock struts.)

D. Main Gear and Nose Gear Shock Strut Inflating Procedure

WARNING: DO NOT INFLATE SHOCK STRUTS WITH OXYGEN, AS AN EXPLOSION MAY RESULT WHICH COULD RESULT IN SERIOUS INJURY OR DEATH TO PERSONNEL.

(1) Remove air filler valve cap and attach air filler chuck.

(2) Loosen swivel nut on filler valve one to two turns.

(3) If airplane is on jacks, service nose and main gear shock struts in accordance with step (a). If airplane is resting on landing gear, determine airplane fuel load, and then service main gear shock strut as indicated by step (b) or (c), and service nose gear shock strut in accordance with pressure versus gap inflation curve as shown in figure 12-11-1.

(a) With nose wheel clear of ground and strut fully extended, inflate strut to 180 psi. With main gear wheels clear of ground and struts fully extended, inflate struts to 459 (±5) psi.

NOTE: This procedure provides optimum gear shock strut inflation and should be followed whenever the shock struts are refilled. Steps (b) and (c) and figure 12-11-1 are provided for convenience to determine the acceptable shock strut inflation. Because of temperature and airplane weight-empty CG variations, bottoming or overextension of the strut may occur when steps (b) and (c) are met. If this occurs, the struts should be inflated in accordance with step (a).

(b) With airplane fully fueled (no crew or baggage), inflate main gear shock strut until upper and lower torque link attach bolts are 5-3/4 inches apart, measured from center of each attach bolt as shown on strut instruction plate. Inflate nose gear shock strut until gap and pressure correspond to curve as shown in figure 12-11-1 or on instruction plate on strut.

(c) With wing tanks full (no auxiliary fuel and no crew or baggage), inflate nose gear shock strut until gland nut and shoulder on inner cylinder are 5 inches apart.

(4) With wing tanks full (no auxiliary fuel and no crew or baggage), inflate main gear shock struts until upper and lower torque link attach bolts are 6 inches apart, measured from center of each attach bolt. Inflate nose gear shock strut until gap and pressure correspond to curve as shown in figure 12-11-1 or on instruction plate on strut.

NOTE: If struts are serviced with the airplane resting on the landing gear, it is suggested that the wings be rocked up and down during inflation. This will partially overcome

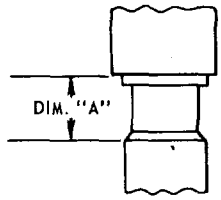
the binding tendency of the strut, and will aid in obtaining uniform strut inflation.

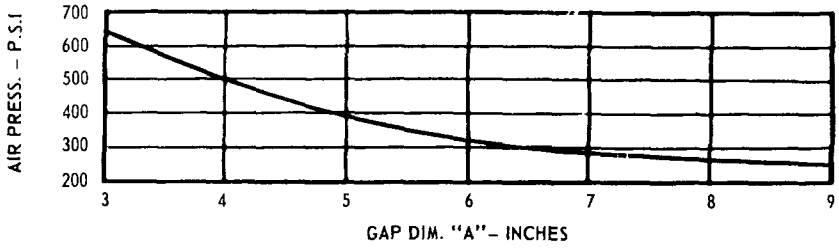
(5) Tighten swivel nut to 50-70 inch-pounds. Shut off air supply to air filler valve and remove air filler chuck.

(6) Check valve for leaks and replace valve cap.

AIR PRESS. VS GAP DIMEN.

1. MEASURE DIM. "A" ON STRUT.
2. ATTACH GAGE TO AIR VALVE AND READ STRUT PRESSURE.
3. MEASURED VALUES MUST CORRESPOND TO CHART.
4. ADJUST AIR PRESSURE AS REQUIRED (GAP WILL CHANGE WITH CHANGE IN PRESSURE) UNTIL A GAP DIMENSION AND PRESSURE CORRESPONDING TO CHART VALUE (WITHIN ± 20 P.S.I.) IS OBTAINED.





5370-2-00-19

Nose Gear Shock Strut Servicing Instructions
Figure 12-11-1

FIRE EXTINGUISHER AGENT SERVICING - MAINTENANCE PRACTICES

1. Servicing Fire Extinguisher System

A. Check Container Pressure

A fire extinguisher container pressure gage is on each fire extinguisher container. The gages are visible on the aft bulkhead of each wheel well. Two discharge indicators are flush-mounted on the left side of the fuselage, inboard of the left engine nacelle. One indicates thermal discharge of the fire extinguisher system; the other indicates manual discharge.

(1) Check both fire extinguisher agent containers for proper pressure. If pressure does not fall with limits as shown in the following table, remove and replace container.

W. KIDDE FIRE EXTINGUISHER AGENT CONTAINERS P/N 892034 and 895724		HTL INDUSTRIES FIRE EXTINGUISHER AGENT CONTAINERS		
TEMPERATURE DEGREES F.	MINIMUM ALLOWABLE PRESSURE	TEMP (°F)	CONTAINER PRESSURE - (PSI) MIN	MAX
-40	100	-60	110	134
-20	120	-40	127	155
0	150	-20	148	180
20	180	0	174	212
40	230	20	207	251
60	280	40	249	299
80	350	60	304	354
100	430	70	334	385
120	520	80	367	417
140	630	100	442	492
		120	532	582

WARNING: MONOBROMOTRIFLUOROMETHANE (CF_3Br) IS NONTOXIC AND CLASSED AS NONPOISONOUS. HOWEVER, IT CAN BE HARMFUL AND SHOULD BE HANDLED WITH CARE. DO NOT BREATHE VAPOR OR PERMIT LI UID TO COME IN CONTACT WITH SKIN OR CLOTHING. IF LEAKAGE IN A CLOSED SPACE IS SUSPECTED, THIS SPACE SHOULD NOT BE ENTERED UNTIL ALL VAPOR HAS BEEN DISSIPATED AND THE AREA IS WELL VENTILATED. USE A HALIDE DETECTOR (PRES-O-LITE, OR EQUIVALENT) TO ENSURE THAT AREA IS SAFE.

(2) Check discharge indicators. If either disk is blown out, agent containers must be checked and the indicator replaced. The red indicator will be blown if either container has been discharged because of thermal expansion. The yellow indicator will be blown if either container has been manually discharged. A check of the agent container pressure gages will show whether one or both containers were discharged.

1. Servicing Fuel Tanks (See figures 12-00-1 and 12-13-201.)

A. Fuel Servicing Precautions

The following precautions should be observed during pressure or gravity refueling of the fuel tanks.

(1) Use two ground wires from different points on the airplane to prevent ungrounding because of accidental disconnection of one wire. The fueling nozzle and fueling unit should be grounded to the airplane during fueling operations.

(2) A fire extinguisher, holding at least 50 pounds of carbon dioxide or equivalent, should be located within 50 feet of refueling operations.

B. Single-point Pressure Replenishing of Fuel Tanks

The single-point refueling receptacle and refueling test buttons are on the lower surface of the right wing leading edge. The procedure for pressure replenishing of the fuel tanks is shown in figure 12-13-201.

NOTE: Ensure that fuel receptacle cap is installed so that the three alignment marks on caps are aligned with marks on wing skin.

C. Gravity Replenishing of Fuel Tanks

CAUTION: ON NA-370-2 THROUGH -6, -8 AND -9 AIRPLANES, ALL OF THE FUEL TANKS MAY BE FILLED THROUGH THE FUSELAGE TANK DURING GRAVITY REFUELING. ON NA-370-1 AND -7, NA-380-1 AND LATER AIRPLANES, THE WING FUEL TANKS MUST BE FILLED USING THE GRAVITY FUELING RECEPTACLES ON THE OUTBOARD ENDS OF EACH WING BEFORE FILLING THE FUSELAGE FUEL TANK. THE FUSELAGE FUEL TANK FILLER IS IN THE FUSELAGE, ABOVE THE RIGHT ENGINE PYLON. (SEE FIGURE 12-00-1.)

NOTE: When servicing fuel tanks, observe walkway markings. Walkways are designated on each wing near the fuselage and on each engine pylon.

Station a man at each wing to hold wing fill-cap spring-loaded doors open to allow wing tank vent. Just before fuel starts to overflow, the doors should be closed. The men on the wing should warn the man stationed at the hose nozzle of fuel rising on the filler necks if it is desired to fly with an empty fuselage tank.

WARNING: BE SURE THAT LEFT AND RIGHT FUEL TANKS ARE FILLED TO THE SAME LEVEL. ASYMMETRICAL FUEL LOADING WILL ADVERSELY AFFECT TRIM IN FLIGHT.

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2. Servicing Fuel System Under Icing Conditions

NOTE: Jet fuels have a stronger affinity for water than aviation gasoline. The settling time for water in jet fuel is five times as long as for water in gasoline. The following recommended procedures can reduce the suspended or free water to 30 parts per million. Suspended water in a quantity of 40 parts per million is sufficient to cause icing in a filter.

A. Corrective or Preventive Action - Ground Handling

The acceptable water content for jet fuel is 30 parts per million. Because fuel within this content is not available at all airports, the following procedures are recommended:

(1) Water-drain main tanks only after each flight and before refueling. This clears tanks of any water condensed in flight (See figure 12-13-202 for drain locations).

(2) Allow maximum possible time between refueling and take-off.

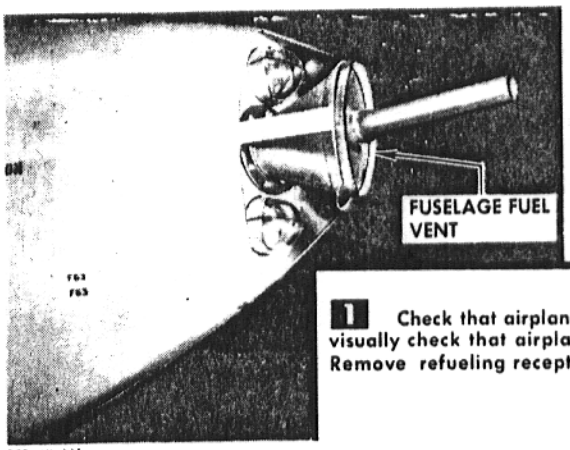
(3) Whenever possible, fuel at or below ambient temperature should be used.

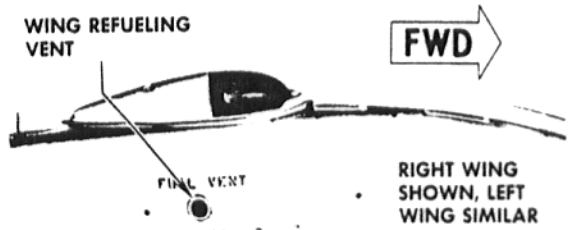
(4) Drain engine fuel filters daily.

(5) Water-drain fuel system just before each flight.

3. Defueling Fuel Tanks

For defueling procedure, refer to Chapter 28.

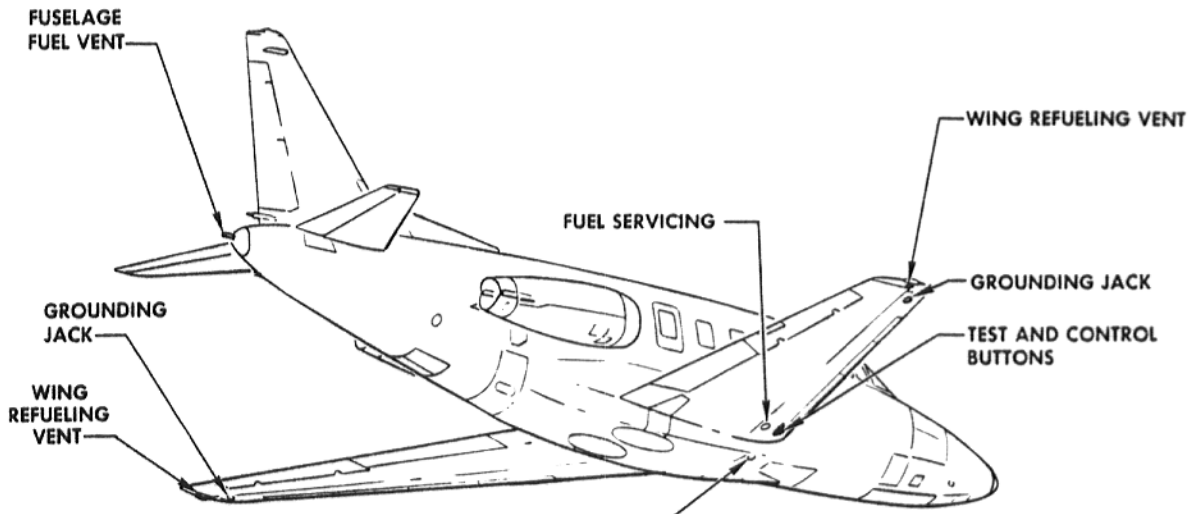




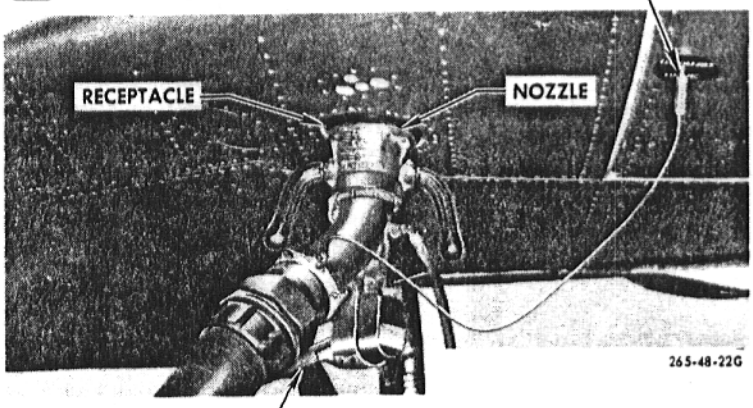
1 Check that airplane ground wires are attached, and visually check that airplane fuel vents are not obstructed. Remove refueling receptacle dust cover.

265-48-22J

265-48-2[illegible]A



2 Insert nozzle grounding plug into jack on airplane.



265-48-22G

3 Connect single-point refueling nozzle to receptacle and turn nozzle clockwise, to hold it in place. Actuate nozzle lever clockwise, to latch nozzle and open nozzle valve.

Caution Before turning on refueling pressure, be sure that refueling vents under wing tip and fuel vent outlet at fuselage are not covered or plugged. If this is not done, the airplane could be seriously damaged.

4 Start ground servicing equipment, and permit fuel to flow into airplane. Immediately check that there is escaping air flowing from wing tip and fuselage fuel vents.

NOTE The refueling pressure should not exceed 55 psi. Pressure lower than 45 psi will prolong refueling.

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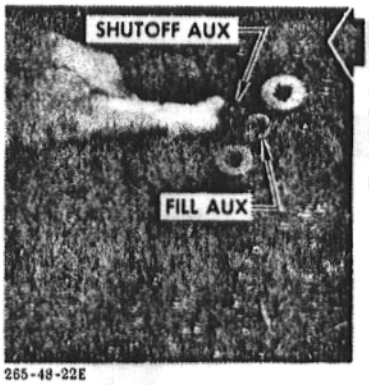
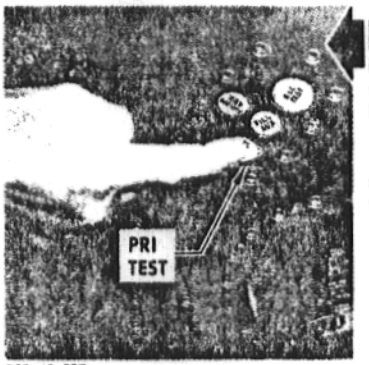


265-48-22E

5 Push "SHUTOFF AUX" button, to be sure that "FILL AUX" button is released. This will allow the primary and secondary diaphragms of the fuselage tank dual-level control valve to be tested when "PRI TEST" buttons are pressed.



265-48-22F

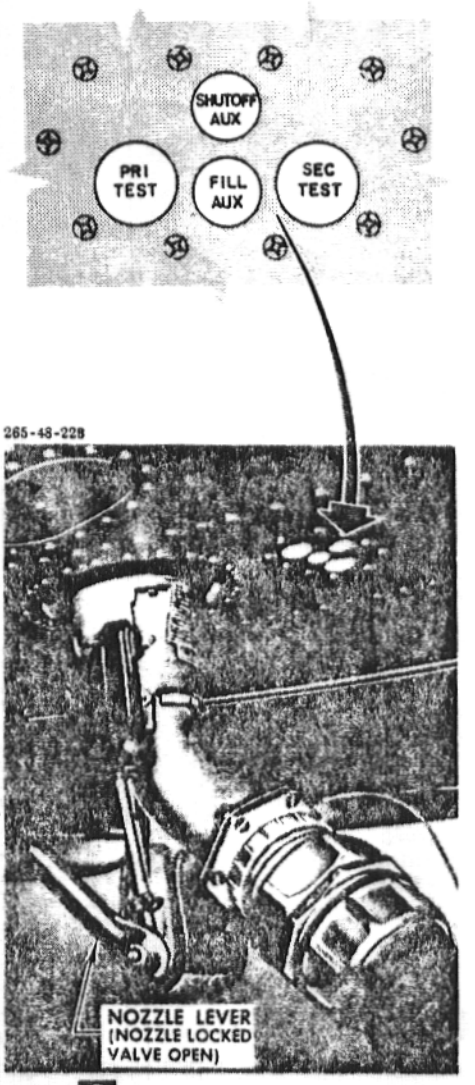
6 As soon as proper airplane venting is verified, check that the refueling shutoff valves will shut off fuel flow. Push and hold "PRI TEST" button. Vibration and noise at nozzle should stop; within 8 to 10 seconds, the refueling supply unit should indicate no more than 2 gpm flow.

265-48-22A

7 Release "PRI TEST" button and press "SEC TEST" button. Results should be the same as in step 6.

Caution Both the primary and secondary operation of the valves must be checked.

• Immediately stop refueling operation if fuel shutoff is not obtained when either test button is pressed; if this is not done, fuel may spill overboard through the fuselage fuel vent. Refueling may then be continued after the difficulty has been corrected.



265-48-22B

8 If fuselage tank is to be filled, after refueling shutoff valves have been tested, press "SHUTOFF AUX" button and hold button depressed while pressing "FILL AUX" button; then release "SHUTOFF AUX" button first. The "FILL AUX" button will be locked in, allowing the fuselage tank to fill.

265-48-22C

265-48-22H

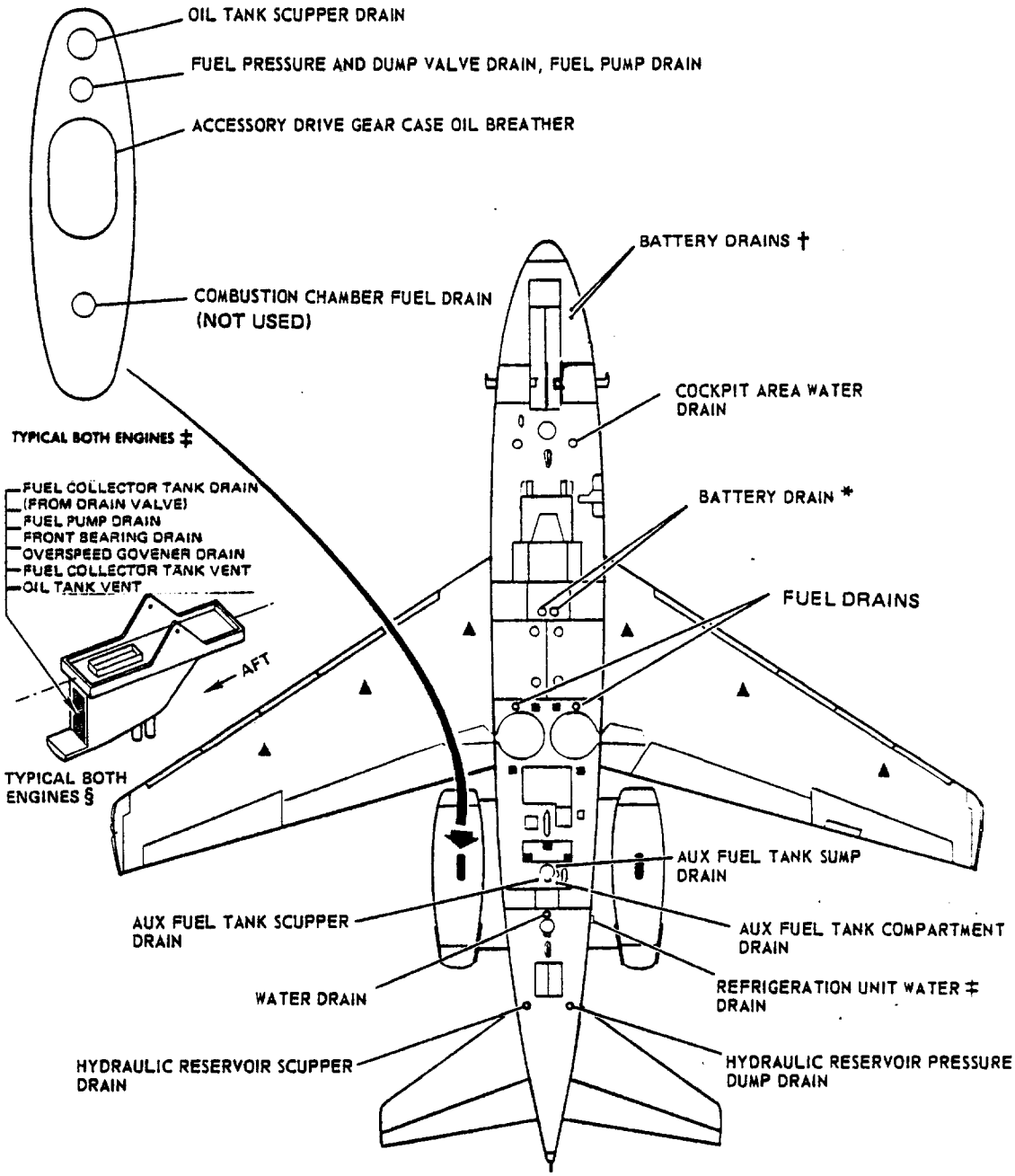
265-48-22H

9 When valves shut off refueling operation, prompty shut down refueling unit. Actuate nozzle lever counterclockwise, to close valve and unlock nozzle.

10 Rotate nozzle counterclockwise; nozzle will drop from receptacle adapter. Remove ground plug from jack, and stow single-point refueling nozzle.

11 Install refueling receptacle dust cover. If fuselage tank was filled, press "SHUTOFF AUX" button.

5370-2-00-21


OIL TANK SCUPPER DRAIN
FUEL PRESSURE AND DUMP VALVE DRAIN, FUEL PUMP DRAIN
ACCESSORY DRIVE GEAR CASE OIL BREATHER
COMBUSTION CHAMBER FUEL DRAIN (NOT USED)
TYPICAL BOTH ENGINES ‡
FUEL COLLECTOR TANK DRAIN (FROM DRAIN VALVE)
FUEL PUMP DRAIN
FRONT BEARING DRAIN
OVERSPEED GOVENER DRAIN
FUEL COLLECTOR TANK VENT
OIL TANK VENT
AFT
TYPICAL BOTH ENGINES §
BATTERY DRAINS †
COCKPIT AREA WATER DRAIN
BATTERY DRAIN *
FUEL DRAINS
AUX FUEL TANK SUMP DRAIN
AUX FUEL TANK SCUPPER DRAIN
AUX FUEL TANK COMPARTMENT DRAIN
WATER DRAIN
REFRIGERATION UNIT WATER ‡ DRAIN
HYDRAULIC RESERVOIR SCUPPER DRAIN
HYDRAULIC RESERVOIR PRESSURE DUMP DRAIN

▲ FUEL TANK SUMP DRAINS

■ FUEL LINE DRAINS

* NA-370 -1 THROUGH -9
† NA-380 -1 AND LATER
‡ NA-370 -2 THROUGH -6, -8, AND -9
§ NA-370 -1 AND -7, AND NA-380 -1 AND LATER

3370-2 00-22A

DEHYDRATOR SYSTEM SERVICING - MAINTENANCE PRACTICES

1. Servicing Cockpit and Cabin Window Dehydrator System

The dehydrating capsules contain a color agent to absorb any moisture in the air trapped between panels of the cockpit and cabin windows. As the color agent changes from deep blue (dry condition) to opaloid or bright pink (moisture-saturated), the absorbent qualitites of the capsules will diminish. Replacement of desiccant or capsules is required when the color agent indicates a high moisture content. The two capsules are in the cabin, on the left side, near the floor and just aft of the entrance door. Escape hatch window dehydrator capsules are replaced as an assembly. Refer to Chapter 56 for dehydrator removal, dehydrator servicing, and installation procedures.

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HYDRAULIC SERVICING - MAINTENANCE PRACTICES

1. Servicing Hydraulic System

A. Servicing Accumulators (See figure 12-00-1.)

The accumulators and servicing panel are in the aft equipment compartment. The accumulators are serviced through the hinged access doors on the underside of the fuselage rear section. Check that speed brakes are retracted.

(1) With reservoir filler cap closed and locking tab down, hold accumulator dump valves partially open, one at a time, and slowly bleed off all hydraulic pressure.

CAUTION: TOO RAPID DUMPING OF THE ACCUMULATORS MAY EXCEED THE RESERVOIR PRESSURE RELIEF VALVE RATE AND CAUSE POSSIBLE DAMAGE.

(2) Check normal and auxiliary accumulator air pressure gages. Pressure should be 1500 (±50) psi. On NA-370-2 through -6, -8 and -9, check thrust reverser accumulator air pressure gage for 900 (±50) psi.

(3) If air servicing is required, attach nitrogen or dry-air supply source chuck to filler valve stem and loosen small hexnut on filler valve of affected accumulator nut more than two turns.

(4) Precharge accumulators with dry air or nitrogen: normal and auxiliary accumulators, 1500 (±50) psi; thrust reverser accumulator on NA-370-2 through -6, -8 and -9 airplanes, 900 (±50) psi.

(5) Tighten filler valve hexnut, and open accumulator dump valve. Recheck precharge, and service as required.

(6) Check filler valve hexnut for tightness, and disconnect air supply source.

B. Checking Reservoir Fluid Level

NOTE: The fluid in the accumulators must be dumped into the reservoir in order to check proper fluid level.

(1) With reservoir filler cap closed and locking tab down, hold accumulator dump valves partially open, one at a time and slowly bleed off all hydraulic pressure.

CAUTION: TOO RAPID DUMPING OF THE ACCUMULATORS MAY EXCEED THE RESERVOIR PRESSURE RELIEF VALVE RATE AND CAUSE POSSIBLE DAMAGE.

(2) Check fluid level in reservoir sight gage. It must be maintained between the "FULL" and "REFILL" marks.

(3) Add fluid if necessary. (Refer to paragraph C.)

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C. Servicing Hydraulic Reservoir

(1) Dump reservoir air pressure as follows:

(a) Discharge air pressure from air storage tank (bottle) by operating drain valve under tank.

NOTE: If excessive hydraulic fluid is noticed coming from drain line, refer to Chapter 29 for corrective action.

(b) Discharge air pressure from reservoir by operating reservoir pressure dump valve until air discharge stops.

(2) Remove reservoir filler cap and pour hydraulic fluid from a clean container into reservoir. Fill reservoir to "FULL" mark on reservoir sight glass.

(3) Replace filler cap on reservoir and position locking tab down.

(4) Precharge of the hydraulic reservoir may be accomplished by using the ground servicing panel or by operating an engine. These procedures should be performed prior to operating hydraulic pump. Step (5) provides procedures to use the ground servicing panel and Step (6) utilizes an engine.

(5) To precharge the hydraulic reservoir with an external air source, proceed as follows:

(a) On NA380-1 thru -38 airplanes not changed by Optional Kit No. 7, remove APU sense line from "T" fitting on the back of the ground service panel; then cap "T" fitting.

(b) Connect a source of dry air or nitrogen (120 psi maximum) to "SEAL REG" fitting on air conditioning ground test panel.

(c) Apply air pressure until reservoir pressure gage indicates at least 35 psi.

(d) Shut off air supply and disconnect from panel and install cap. Remove cap from "T" fitting and install APU sense line. Tighten sense line "B" nut.

(6) To precharge the hydraulic reservoir using an engine, proceed as follows:

(a) Precharge reservoir by operating one engine at 70%*, 85 to 95%† rpm (cabin pressurization off) for 1/2 minute to supply reservoir air pressure.

NOTE: Use emergency brakes or adequately chock wheels for this procedure.

*NA370-2 thru -6, -8 and -9
†NA370-1, -7 and NA380 Airplanes

OIL SERVICING - MAINTENANCE PRACTICES

1. Servicing Oil System - Aircraft NA-370-2 Through -6, -8 and -9

A. Checking Engine Oil System

NOTE: The engine oil level must be checked within one hour after engine shutdown. The engine shutdown should include an oil scavenging run at 75% rpm for 30 seconds just before the power lever is moved to OFF. With the engine shutdown, the oil level in the tank will drop as oil seeps into the accessory gear case by gravity flow. When the engine is started, the scavenge pump will rapidly clear the gear case, returning the oil level to normal. If an oil level check cannot be made within an hour after shutdown, it is necessary to start the engine and make a scavenge run and normal shutdown before servicing the oil system.

It is undesirable to intermix different brands or types of oil when adding oil.

The oil used is a synthetic type and is harmful to some substances. Be careful not to spill oil on electrical wiring, painted surfaces, rubber parts, etc. Spilled oil must be wiped off immediately with a clean cloth saturated in aliphatic naphtha.

Prolonged contact with synthetic engine oil is very irritating to the skin and eyes. Wash hands with soap and water before touching any food after servicing oil system.

Containers and hoses used for petroleum-based oil must not be used for synthetic oils.

To ensure cleanliness, individual one-time-use quart cans are recommended, rather than bulk oil. If bulk is used, it should be filtered through a 325 mesh (44 micron) or smaller filter.

B. Oil Servicing Procedure (Each Engine)

(1) Open oil tank filler access door.

(2) Raise locking tab on filler cap and rotate counterclockwise until cap is released, then remove cap.

CAUTION: Make sure wind, prop wash, and/or jet blast does not blow dirt or other foreign matter into oil tank while filler cap is off.

(3) Remove dip stick, wipe it with clean cloth, and replace it in tank. Remove dip stick and check oil level. When it is necessary to add oil, do not overfill.

NOTE: Because about one quart of oil usually remains in the engine accessory gear case after shutdown, adding oil to bring level up to "FULL" line will cause overfilling.

If oil tank cannot be serviced (proper brand or type of oil is not available) a flight can be made if oil level is not more than two quarts below "FULL" line.

(4) Replace dip stick and filler cap. Rotate locking tab clockwise to lock cap, then press tab down to its locked position. Make sure cap is installed properly (cap flush with edge of recess in oil tank). Then close and fasten oil tank filler access door.

2. Servicing Oil System - Aircraft NA-370-1 and -7, NA-380-1 and Later

A. Checking Engine Oil System

NOTE: The engine oil level must be checked within 30 minutes after engine shutdown. If an oil level check cannot be made within 30 minutes after engine shutdown, it will be necessary to start the engine, operate for two minutes at idle speed, then perform a normal shutdown before servicing the oil system.

(1) Open engine lower access doors.

CAUTION: MAKE SURE WIND, PROP WASH AND/OR JET BLAST DOES NOT BLOW DIRT OR OTHER FOREIGN MATERIAL INTO OIL TANK WHILE FILLER CAP IS OFF.

(2) Raise locking tab on filler cap and rotate counterclockwise until cap is released, then remove cap.

(3) Remove dip stick, wipe it with a clean cloth and replace it in tank. Remove dip stick and check oil level.

(4) If oil level is within one pint of the FULL mark on the dip stick it will not be necessary to add oil. If the oil level is below one pint from the FULL mark, add enough engine oil to bring the oil level to the FULL mark on the dipstick.

NOTE: Always add the same brand and type of oil as the existing oil in the tank.

(5) Replace dipstick and filler cap. Rotate locking tab on cap clockwise to lock cap. Make certain cap is properly installed.

(6) Close engine lower access doors.

OXYGEN SERVICING - MAINTENANCE PRACTICES

1. Servicing Oxygen System

A. General

The oxygen system storage cylinder is mounted in the aft end of the passenger compartment, however some airplanes have two cylinders, which are plumbed in parallel. (The mounting location of the oxygen cylinders can vary as the result of various cabin interior and/or seating arrangements, but usually it is under the bench-type rear seat or behind the left rear side closeout panel.) The cylinder has a direct reading pressure gage and a manual shutoff valve which includes a frangible pressure relief valve. The relief valve ruptures if pressure in the cylinder exceeds 2700 to 2900 psi. This high-pressure oxygen discharge from the relief valve ruptures a green frangible disc indicator on the left side of the fuselage near the wing trailing edge and vents overboard through the disc. An intact disc indicates the relief valve has not blown. A walk around oxygen bottle is stowed in the coat compartment.

B. Servicing Oxygen System

Most airplanes have an external oxygen filler valve. On other airplanes the cylinder(s) may be serviced in the airplane if equipment is available, otherwise the cylinder(s) must be removed for servicing. During oxygen servicing, one man must remain stationed at the oxygen servicing unit to shut off the valves immediately upon command of person who is watching airplane oxygen cylinder pressure.

WARNING: THE OXYGEN SYSTEM SHOULD BE SERVICED ONLY BY PERSONNEL WHO ARE FAMILIAR WITH THE SAFETY PRECAUTIONS AND TECHNIQUES FOR HANDLING GASEOUS OXYGEN, AND KNOW HOW TO OPERATE OXYGEN SERVICING EQUIPMENT.

Personnel must remain clear of oxygen system disc-type discharge indicator (green) on the left side of fuselage while oxygen system is being serviced.

C. Servicing Oxygen (Airplane with External Oxygen Filler Valve)

WARNING: KEEP OIL, GREASE, AND HYDRAULIC FLUID CLEAR OF OXYGEN.

NOTE: A man must be stationed in the airplane cabin during oxygen servicing to watch oxygen cylinder pressure gage and report pressure reading to prevent overfilling of cylinder.

(1) Bond airplane to oxygen servicing unit.

(2) Close all cylinder valves on servicing unit. Record pressure in each cylinder on separate tag and attach tag to corresponding cylinder.

(3) Have man in airplane make sure bypass (emergency) lever, if installed, on passenger oxygen control panel is at NORMAL (NORM) position.

(4) Open oxygen filler valve access door. Remove dust cap and attach servicing unit hose adapter to filler valve.

(5) Open servicing unit panel valve, then open valve on the servicing unit cylinder that is tagged with the lowest pressure that is higher than the pressure noted on the airplane oxygen cylinder pressure gage.

(6) Open valve on end of servicing unit hose. When servicing cylinder pressure gage shows no flow, the pressures in the servicing cylinder and in the airplane storage cylinder are equal.

(7) Close servicing cylinder and actuate another cylinder, always using a cylinder of progressively higher pressure until servicing unit pressure gage indicates about 1800 psi.

(8) Continue filling airplane storage cylinder until man in airplane reports pressure gage reads approximately 1850 psi. Then close valve on end of servicing hose.

(9) Close servicing unit panel valve.

(10) Disconnect servicing unit hose adapter from airplane filler valve. Open valve on hose slowly to allow hose pressure to be released.

(11) Screw dust cover on airplane filler valve and close valve access door.

(12) Disconnect bonding.

D. Servicing Oxygen (Airplane Without External Oxygen Filler Valve).

WARNING: KEEP OIL, GREASE, AND HYDRAULIC FLUID CLEAR OF OXYGEN.

(1) Bond airplane to oxygen servicing unit.

(2) Close all cylinder valves on servicing unit. Record pressure in each cylinder on separate tag and attach tag to corresponding cylinder.

(3) Remove cabin furnishings and/or trim-closeout panels necessary to gain access to airplane oxygen storage cylinder(s). Close shutoff valve on cylinder(s).

(4) Make sure by-pass (emergency) lever, if installed, on passenger oxygen control panel in cockpit is at NORMAL (NORM) position.

(5) Disconnect and remove airplane oxygen supply line from cylinder.

(6) Attach servicing unit hose adapter to storage cylinder supply fitting.

(7) Open servicing unit panel valve.

(8) Open shutoff valve on airplane oxygen cylinder(s), and note pressure indicated on cylinder pressure gage(s).

(9) Open valve on the servicing unit cylinder that is tagged with the lowest pressure that is higher than the pressure noted on the pressure gage on the airplane cylinder.

(10) Open valve on end of servicing unit hose. When servicing cylinder pressure gage shows no flow, the pressure in the servicing cylinder and in the airplane storage cylinder are equal.

(11) Close servicing cylinder and actuate another cylinder, always using a cylinder of progressively higher pressure until servicing unit pressure gage indicates about 1800 psi.

(12) Continue filling airplane storage cylinder until pressure gage on cylinder reads approximately 1850 psi. Then close valve on servicing unit end of hose.

(13) Close shutoff valve on airplane oxygen cylinder(s).

(14) Close servicing unit panel valve.

(15) Disconnect servicing unit hose adapter from storage cylinder supply fitting.

(16) Connect airplane oxygen supply line to cylinder and open shutoff valve on cylinder(s).

(17) Remove hose from airplane and slowly open valve on hose to allow hose pressure to be released.

(18) Replace cabin furnishings and/or trim-closeout panels.

(19) Disconnect bonding.

BATTERY SERVICING - MAINTENANCE PRACTICES

1. Servicing Battery

NOTE: Because the batteries in the Sabreliner may be procured from different manufacturers, the servicing procedures in this section are general in nature and are not intended to supersede the manufacturers specific requirements. For specific servicing procedures applicable to the type of battery being serviced, refer to the applicable manufacturers servicing instructions.

A. General

The nickel-cadmium batteries used in this airplane are virtually maintenance-free; however normal service procedure requires that the battery electrolyte level be checked at whatever interval is dictated by the existing operating and climatic conditions.

CAUTION: NEVER SERVICE A NICKEL-CADMIUM BATTERY WITH A SYRINGE, DROPPER, OR HYDROMETER WHICH HAS BEEN PREVIOUSLY USED TO SERVICE A LEAD-ACID BATTERY. SUCH PRACTICE WILL CONTAMINATE THE NICKEL-CADMIUM BATTERY ELECTROLYTE. ABSOLUTE CLEANNESS IN BATTERY SERVICING MUST BE MAINTAINED TO AVOID BATTERY MALFUNCTION DUE TO CONTAMINATION. ANY SUCH CONTAMINATION WILL CAUSE PERMANENT DAMAGE TO THE BATTERY AND RENDER IT USELESS IN A VERY SHORT PERIOD OF TIME.

On aircraft NA-370-1 through -9, the two batteries are in the lower part of the fuselage forward section, on the centerline of the airplane between stations 206 and 223. They may be reached through an access door on the bottom of the fuselage. The batteries are mounted on a movable tray, which may be raised or lowered, in and out of the fuselage, to facilitate servicing. On aircraft NA-380-1 and later, the batteries are located in the nose section forward of station 50 and are serviced in place.

On aircraft NA-370-1 through -9, the batteries are positioned for servicing as follows:

(1) Disengage Camlock fasteners that support battery sump jar vent lines.

(2) Remove battery access door.

(3) Remove battery tray support locknuts.

(4) Use a 7/16-inch socket wrench to turn battery hoist fitting (located in bay adjacent to battery compartment, and accessible through a small spring-loaded access door).

(5) Turn fitting clockwise to lower tray.

(6) Turn fitting counterclockwise to return battery tray to supports, engaging guide pins until ends of tray support bolts that extend below lower surface of tray.

CAUTION: DO NOT TURN HOIST FITTING UNTIL TRAY IS FORCED TO BOTTOM AGAINST TRAY SUPPORTS, BECAUSE OF POSSIBLE OVERTORQUING AND DAMAGE TO HOIST.

(7) Attach tray cradle locknuts, and tighten until tray is secured against cradle.

(8) Install access door, and secure sump jar vent line support Camlock fasteners.

B. Battery Servicing Precautions

Before attempting to service airplane batteries, observe the following precautions:

(1) Locate airplane in an area approved for maintenance.

(2) Place all airplane switches at OFF or airplane-parked position.

(3) Disconnect all external power from airplane.

(4) Tag airplane external power receptacles with a danger sign.

C. Determining Serviceability of Batteries

The serviceability of the airplane batteries can be determined while they are installed in the airplane. To determine serviceability, proceed as follows:

(1) Observe steps (1) through (3) of battery servicing precautions.

(2) Gain access to battery; **disconnect airplane wiring harness** from one battery. Temporarily insulate and stow this wiring harness quick-disconnect.

(3) In airplane cockpit, place "ELECTRIC MASTER" and "BATTERY" switches at ON.

(4) Create an approximate load on battery of 25 amperes by turning on aft fuselage lights (3.0 amperes), position lights (5.8 amperes), and one landing light (when extended, lamp only, 16.1 amperes).

(5) Note airplane "DC VOLTS" meter (on copilot's electrical power control panel). The voltmeter should read at least 22 volts dc if the battery is serviceable.

NOTE: A dc voltmeter reading of less than 22 volts dc indicates that the serviceability of the battery is not acceptable. A reading of approximately 24 volts dc indicates a fully charged battery.

(6) Turn off the airplane lights turned on in step (4), and place "BATTERY" and "ELECTRIC MASTER" switches at OFF.

(7) Reconnect airplane wiring harness quick-disconnect plug to airplane battery.

(8) Repeat steps (2) through (7) for other battery.

D. Charging Batteries in Airplane

Partially discharged batteries can be charged while they are in the airplane. Charge batteries as follows:

(1) Observe steps (1) and (2) of battery servicing precautions.

(2) Connect an external dc power source to airplane external dc utility power receptacle.

(3) Turn on external dc power source, and adjust to an output of 28.5 volts dc.

(4) Place airplane "ELECTRIC MASTER" and "BATTERY" switches at ON.

(5) Note dc voltmeter on external power source. If meter does not read 28.5 volts dc, adjust output voltage of external dc power source to obtain this reading. Maintain output voltage at 28.5 volts dc throughout this entire procedure.

(6) Note ammeter reading on external power source; write down this value. The meter should indicate the airplane dc current load.

(7) Place airplane "BATTERY" switch at OFF, and note ammeter reading on external power source; write down this value.

(8) Figure difference between ammeter readings in steps (6) and (7). If difference is greater than 10 amperes, place airplane "BATTERY" switch at ON and continue charging procedure. (Make sure that dc external power source output is 28.5 volts dc.)

(9) Interrupt charging procedure periodically and repeat steps (6) and (7). When difference in ammeter readings is less than 10 amperes [step (8)], discontinue charging procedure and place "BATTERY" and "ELECTRIC MASTER" switches at OFF.

NOTE: Batteries are considered charged when they draw less than 16 amperes (8 amperes each) of charging current from the external dc power source.

(10) Turn off external dc power source and disconnect it from airplane.

E. Checking Electrolyte Level in Battery

When batteries are installed in airplane, check electrolyte level in cells as follows:

(1) Place all airplane switches at OFF or in airplane-parked position, and disconnect all external power from airplane.

NOTE: Batteries should be fully charged, and then remain idle for at least 3 hours before electrolyte level in battery cells can be checked. The idle period is required because the charging operation drives the electrolyte out of the cell plates, and the level of the electrolyte does not stabilize until at least 3 hours after completion of the charging operation.

(2) Remove lids from battery cases.

(3) Remove vent caps from battery cells. (Use Sonotone Corp vent cap wrench.)

(4) Insert one end of a glass tube, about 6 inches long with an inside diameter of 1/8 inch, into cell vent. Lower tube into cell until it contacts nylon barrier inside cell.

(5) While holding glass tube in a vertical position, cover upper end of tube with a finger tip. Press finger down on tube and hold it firmly in this position.

(6) Remove glass tube from cell, while maintaining finger-tip pressure on end of tube.

(7) Note amount of electrolyte captured inside lower end of tube. If captured electrolyte extends up inside glass tube a distance of 1/16 to 1/8 inch, the level of the cell electrolyte is satisfactory.

(8) Lower end of glass tube back into cell and release finger pressure from top of tube, releasing captured electrolyte into cell. Remove tube from cell.

(9) Replace vent cap on battery cell and tighten, using vent cap wrench. (Do not overtighten cell vent cap.)

(10) Repeat steps (4) through (9) for remaining cells in both batteries.

NOTE: If any cell does not have required amount of electrolyte, remove battery from airplane, cycle battery, and add distilled water to electrolyte after an idle period of at least 3 hours.

(11) Install lids on battery cases and secure lids in position with tie-down rods.

F. Adding Distilled Water in Battery Cells

When a battery requires addition of distilled water in electrolyte, proceed as follows:

CAUTION: NEVER USE A SYRINGE, DROPPER, HYDROMETER, OR DISTILLED WATER WHICH HAS BEEN PREVIOUSLY USED TO SERVICE A LEAD-ACID BATTERY. SUCH PRACTICE WILL CONTAMINATE THE NICKEL-CADMIUM BATTERY ELECTROLYTE. CONTAMINATION WILL CAUSE PERMANENT DAMAGE TO THE BATTERY AND RENDER IT USELESS IN A VERY SHORT PERIOD OF TIME.

WARNING: IF ANY ELECTROLYTE IS SPILLED ON CLOTHING OR HANDS, FLOOD AFFECTED AREA IMMEDIATELY WITH CLEAN WATER. IF ANY ELECTROLYTE IS SPLASHED INTO EYES, FLOOD EYES IMMEDIATELY WITH CLEAN WATER (OR A 5 PERCENT SOLUTION OF BORIC ACID AND WATER) SEVERAL TIMES; THEN SEEK FIRST AID AS SOON AS POSSIBLE.

(1) Observe battery servicing precautions.

(2) Gain access to battery. Disconnect, temporarily insulate, and stow airplane wiring harness quick-disconnect plug from battery case.

(3) Remove battery from airplane and place in an upright position on bench in shop area. Remove lid from battery case.

(4) Cycle battery (Refer to 24-30-2).

(5) Allow battery to remain idle for at least 3 hours after completion of cycling procedure.

(6) Remove vent caps from battery cells (use Sonotone Corp vent cap wrench) and check electrolyte level in each cell.

(7) If electrolyte level is low in a battery cell (or cells), slowly add distilled water to cell, using a syringe, dropper, or hydrometer.

(8) Recheck level of electrolyte in battery cell. Readjust level, if necessary, using a syringe or hydrometer.

(9) Replace vent caps on battery cells. Use vent cap wrench; do not overtighten caps.

(10) Before replacing battery in airplane, cycle battery to thoroughly mix distilled water with cell electrolyte.

(11) Install lid on battery case and install battery in airplane.

(12) Reconnect airplane wiring harness quick-disconnect plug to battery receptacle.

(13) Perform an operational check-out of battery system.

G. Checking Specific Gravity of Battery Cell Electrolyte

NOTE: The specific gravity of the battery cell electrolyte cannot be checked until after the battery has been removed from the airplane, discharged, and then disassembled. This procedure is necessary because the small amount of electrolyte present above the cell plates is not enough to operate the hydrometer float. After the battery is disassembled, the electrolyte in each cell must be carefully dumped into an absolutely clean container before the specific gravity reading can be taken. For further information concerning this procedure, refer to 24-30-2.

H. Cleaning batteries

To remove accumulated corrosion from batteries, proceed as follows:

CAUTION: BEFORE ATTEMPTING TO REMOVE ANY CORROSION FROM BATTERY (OR BATTERIES), MAKE SURE ALL CELL VENT CAPS ARE PROPERLY INSTALLED AND TIGHTENED.

- DO NOT USE A WIRE BRUSH TO REMOVE CORROSION FROM TOP OF BATTERY CELLS; SEVERE ARCING WILL OCCUR. USE A DRY, STIFF-BRISTLED FIBER BRUSH TO LOOSEN CORROSION. REMOVE CORROSION FROM BATTERY BY WASHING BATTERY WITH CLEAN WATER.

(1) Observe battery servicing precautions.

(2) Gain access to battery tray. Disconnect, temporarily insulate, and stow airplane wiring harness quick-disconnect plugs from battery receptacles.

(3) Remove batteries from airplane.

(4) Temporarily cap or plug disconnected battery vent and drain lines to prevent dripping any accumulated electrolyte on airplane structure.

(5) Remove covers from battery cases; then check tops of batteries for evidence of corrosion.

(6) If any corrosion is present on tops of battery cells, loosen corrosion by brushing cell tops with a dry stiff-bristled fiber brush.

(7) Remove loose corrosion from tops of battery cells and inside of battery cases by washing with clean water.

(8) Dry battery cell tops and inside of battery cases.

(9) While batteries are still removed from airplane, check battery cell leakage current. (Refer to procedure for checking battery cell leakage current.)

(10) Replace covers on battery cases and install batteries in airplane.

(11) Remove temporary caps or plugs from vent and drain lines; then connect lines to battery cases.

(12) Connect airplane wiring harness quick-disconnect plugs to battery receptacles.

(13) Perform operational check-out of battery system.

I. Checking Battery Cell Leakage Current

NOTE: The battery cell leakage current should not exceed 50 milliamperes (0.050 ampere). Batteries having a cell leakage current in excess of this amount should be replaced by a serviceable battery.

To check battery cell leakage current, proceed as follows:

(1) If battery is installed in airplane, observe battery servicing precautions.

(2) If battery is installed in airplane, disconnect airplane wiring harness quick-disconnect plug from battery receptacle; then temporarily insulate and stow quick-disconnect plug.

(3) Use a multimeter to check battery cell leakage current flow from battery positive terminal (or negative terminal) to battery case.

(4) If leakage current is more than 50 milliamperes (0.050 ampere), replace battery. Be sure cover is installed on battery.

(5) Reconnect airplane wiring harness quick-disconnect plug to battery receptacle.

(6) If remaining battery is installed in airplane, repeat steps (2) through (5).

(7) Perform an operational check-out of battery system.

J. Servicing Battery Sump Jar

NOTE: During sump jar removal, keep jar in an upright position to avoid possible spillage of contents onto airplane structure. This is necessary because the sump jar lid is not held on securely after the tie-down rods have been disengaged from it.

- As soon as drain tubes are disconnected from sump jar, cap or plug tubes and secure them so that they cannot drip any accumulated moisture onto airplane structure.

CAUTION: MAINTENANCE PERSONNEL SHOULD WEAR RUBBER GLOVES, A RUBBER APRON, AND A FACE SHIELD TO PROTECT THEM FROM COMING INTO POSSIBLE CONTACT WITH ANY ACCUMULATED BATTERY ELECTROLYTE SOLUTION IN SUMP JAR.

(1) Observe battery servicing precautions.

(2) Disconnect drain tubes from jump jar, cap or plug tubes, and secure them so that they cannot drip onto airplane structure.

(3) Remove sump jar from airplane.

(4) Remove lid from sump jar.

(5) Note condition of pad inside sump jar. The pad should be moist with a solution of boric acid (3 percent by weight) and water. The jar should not contain any accumulation of electrolyte from battery case.

(6) If sump jar contains an accumulation of battery electrolyte, thoroughly wash out sump jar, and clean pad with clean water. Dry inside of sump jar, and squeeze excess moisture out of pad. Moisten pad with a 3 percent solution of boric acid and water; then return pad to inside of sump jar.

(7) Position lid on battery sump jar.

(8) Hold sump jar in an upright position, and placejar in its installed position on airplane structure.

(9) Secure sump jar to airplane structure with tie-down rods. (Be sure to safety tie-down rod wing nuts with lockwire.)

(10) Remove caps or plugs and connect drain tubes to drain connections on sump jar lid.

K. Servicing Battery System Vent and Drain Tubes

WARNING: DO NOT PUT MOUTH IN DIRECT CONTACT WITH AN OPEN END OF A BATTERY SYSTEM VENT OR DRAIN TUBE AS A MEANS TO BLOW CLEAR ANY OBSTRUCTION IN TUBE. FAILURE TO OBSERVE THIS WARNING CAN RESULT IN SEVERE MOUTH AND THROAT BURNS TO MAINTENANCE PERSONNEL.

(1) Observe battery servicing precautions.

(2) Disconnect lower end of vent or drain tube and place it in a glass jar.

NOTE: Do not permit any accumulation of electrolyte or moisture in vent or drain tube to drip onto airplane structure.

(3) Apply dry air pressure to upper end of vent or drain tube.

NOTE: Use a 2 to 5 psi dry-air pressure source to clear any obstructions in vent or drain tubes.

(4) Continue blowing dry air down vent or drain tube to remove obstruction.

(5) If obstruction cannot be removed, completely disconnect section of vent or drain tube and remove it from airplane. Replace faulty section with a new tube.

(6) Connect lower end of vent or drain tube; then remove glass jar from airplane.

ACCESS DOORS AND PANELS - DESCRIPTION

1. General

A. Access and Inspection Provisions

Removable panels and inspection doors are installed to provide access for inspection and maintenance of the airplane. The appropriate access doors or panels to use are referred to in pertinent chapters. Access door locations are shown in Chapter 6, figure 6-00-3.

AIRCRAFT CLEANING

MAINTENANCE PRACTICES

Refer to the Corrosion Control Manual, Report No. SR-76-023 for aircraft cleaning.

WASHING AIRPLANE EXTERIOR

Wash airplane exterior including wheel wells and landing gear.

CAUTION

Close wing slats while washing airplane to prevent washing lubrication from slat bearings.

Accomplish the following lubrication procedures immediately following the airplane exterior wash:

1. Lubricate Brake Control Valve per section 12-00-0, page 11.
2. Lubricate NLG and Doors per section 12-00-0, pages 8 thru 10.
3. Lubricate NLG Uplock per section 12-00-0, page 8.
4. Lubricate LH and RH MLG and Doors per section 12-00-0, page 12.

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CHAPTER

20

STANDARD PRACTICES (AIRFRAME)

CHAPTER 20

STANDARD AIRFRAME PRACTICES

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CHAPTER 20

STANDARD AIRFRAME PRACTICES

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STANDARD AIRFRAME PRACTICES

DESCRIPTION

GENERAL

Certain airframe structural repair practices may be considered new or unique for this airplane. The use of Chem-Milled and machine milled aluminum alloy skins in the basic airframe structure assemblies make their repair practices nonstandard. The repair of the aluminum honeycomb sandwich panels used in this airplane also may be considered nonstandard practice.

The fuselage is of all-metal semimonocoque construction. Chem-Milled aluminum alloy stressed skins, along with formed sheet metal and extruded frames (some of which are Chem-Milled), carry the shear loads. The major portion of the bending loads is carried by the four main longerons. The remainder of the bending loads is carried by the various stringers, intercostals, and secondary longerons. The thinned areas of the Chem-Milled fuselage skins have been reduced to a minimum gauge consistent with the applicable load and fatigue factors and crack propagation resistance.

The basic structure of the wing consists of front and rear spars, transverse ribs, and upper and lower machine-milled skins. This forms the main box beam which extends from wing tip to wing tip. The slats and leading edge skins are Chem-Milled. The trailing edge cantilevers from the rear spar of the main box section and is constructed of beams, ribs, skins, and honeycomb sandwich panels. The flaps and ailerons consist primarily of beams, transverse ribs, skins, and honeycomb sandwich panel trailing edges. Aileron aerodynamic balance is obtained by a series of steel weights attached inside the leading edge skin.

The empennage consists of a fixed vertical stabilizer, a rudder, a rudder trim tab, a trimmable horizontal stabilizer, an elevator, and fillets. The vertical and horizontal stabilizers are swept back 30 degrees at the 25 percent plane.

The vertical stabilizer is an all-metal, fully cantilevered structure. The main box structure consists of a full-length rear beam, a short lower front beam, transverse ribs, two longitudinal stringers, and side and leading edge skins. The box section above the rudder is formed by the rear beam, a honeycomb panel trailing edge, and side skins. The side skins are Chem-Milled. The vertical stabilizer is attached to the fuselage by bolts at the front and rear beams.

The rudder is a single-cell torque box consisting of a rear beam, two forward longitudinal stringers, transverse ribs, and side and leading edge skins. The rear beam consists of a sheet-metal channel and a honeycomb sandwich panel trailing edge, except in the area of the trim tab. The trim tab is a honeycomb sandwich panel structure with a sheet-metal channel leading edge.

The rudder is suspended from the stabilizer rear beam by three hinges and is rotated by a torque tube attached to the root rib. The rudder is both aerodynamically and statically balanced. Static balance is achieved by steel weights attached to the inside of the leading edge skin. The trim tab is statically balanced and rotates ±15 degrees.

The horizontal stabilizer is of full cantilever construction. It consists of two areodynamically neutral panels, connected by a pivoted beam through the fuselage. The complete assembly rotates about a pivot axis at FS 466.53 and water plane 20 for longitudinal trim. The structure is a single-cell box with a rear beam, two forward longitudinal stringers, and transverse ribs. The structure has upper and lower skins, and a leading edge skin, and a tip. The upper and lower skins are Chem-Milled.

The elevators are interconnected through a torque tube. The structure is a single-cell torque box consisting of a rear beam, two forward longitudinal stringers, and transverse ribs. The rear beam incorporates a sheet-metal channel and a honeycomb sandwich panel trailing edge. The structure is covered with upper and lower skins, a leading edge skin, and a tip. The upper and lower skins are Chem-Milled.

The elevator is suspended from the stabilizer rear beam by three hinges. The elevator is both aerodynamically and statically balanced. Static balance is achieved by a series of steel weights attached to the inner side of the leading edge skin.

REPAIRS

Repairs must be accomplished in accordance with Sabreliner Structural Repair Manual (Report No. NA-66-1032) instructions or by Federal Aviation Agency approved repair procedures for safe life structure.

PAINTING

It has been determined that polyurethane topcoat finishes are sufficient for all surfaces of the airplane. Refer to Sabreliner Corrosion Control Manual, SR-76-023, for a detailed description of painting procedures for all surfaces of the airplane.

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DESCRIPTION (Continued)

FLARED TUBING AND FLEX HOSE B-NUT TORQUE VALVES

The torque table is to be used as a guide in assembling flared tubes and flexible hoses.

It is absolutely necessary that open ends of tubing and all fittings on units be kept capped right up to the time they are connected into the system.

Route tubing and flexible hose so that they clear adjacent structure, hot-air ducts, electrical cables and equipment, flight control cables, and oxygen lines and equipment. All fluid and fuel lines should be routed at least 6 inches from and below electrical cables, if possible, to prevent fire due to line leakage.

Before assembly, inspect tube flares for cracks, burrs, sharp edges, and equal roundness with sleeves. Tight sleeves will not be considered cause for rejection. Nuts should turn freely on the sleeves.

When tube is in position, the tube flares should meet the fittings squarely and fully. Never use nut to draw a flare to fitting, as flare might be easily spun off or damaged.

Start nuts on fittings, and turn until flares and sleeves are firmly seated. Never use a wrench until nut is finger-tight. Use a wrench on the hex or flat of the body of the fitting to prevent turning of the fitting.

NOTE

The torque values given apply to the tubing material, regardless of the fitting or nut material. (Refer to "Identification of Tubing Material" in this section.)

FLARED TUBING AND FLEX HOSE B NUT TORQUE VALUES

TUBE OD INCHES	"B" NUT WRENCH SIZE	52S-0 AL ALLOY TUBING		6061-T6 AL ALLOY TUBING AND EQUIVALENT FLEX HOSE ASSEMBLY [1]		MIL-T-6845 STAINLESS STEEL TUBING		ALUMINUM ALLOY TUBING FOR USE ON OXYGEN LINES ONLY	
		IN. LB MIN	IN. LB MAX	IN. LB MIN	IN. LB MAX	IN. LB MIN	IN. LB MAX	IN. LB MIN	IN. LB MAX
1/8	3/8	20	25						
3/16	7/16	25	35	30	70	90	140		
1/4	9/16	40	65	70	120	135	185		
5/16	5/8	60	80	70	120	180	230	100	125
3/8	11/16	75	125	130	180	270	345	200	250
1/2	7/8	150	250	300	400	450	525	300	400
5/8	1	200	350	430	550	650	750		
3/4	1-1/4	300	500	650	800	900	1100		
1	1-1/2	500	700	900	1100	1200	1400		
1-1/4	2	600	900	1200	1450	1500	1800		
1-1/2	2-1/4	600	900	1550	1850	2000	2300		
1-3/4	2-5/8	700	1000	2000	2350	2600	2900		
2	2-7/8	800	1100	2500	2900	3200	3600		

[1] GENERALLY NOT USED IN FUEL, ENGINE, AND ACCESSORIES, OR HEAT AND VENT SYSTEMS. GEN-16E

Figure 20-00-1 — Flared Tubing and Flex Hose B-nut Torque Values

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DESCRIPTION (Continued)

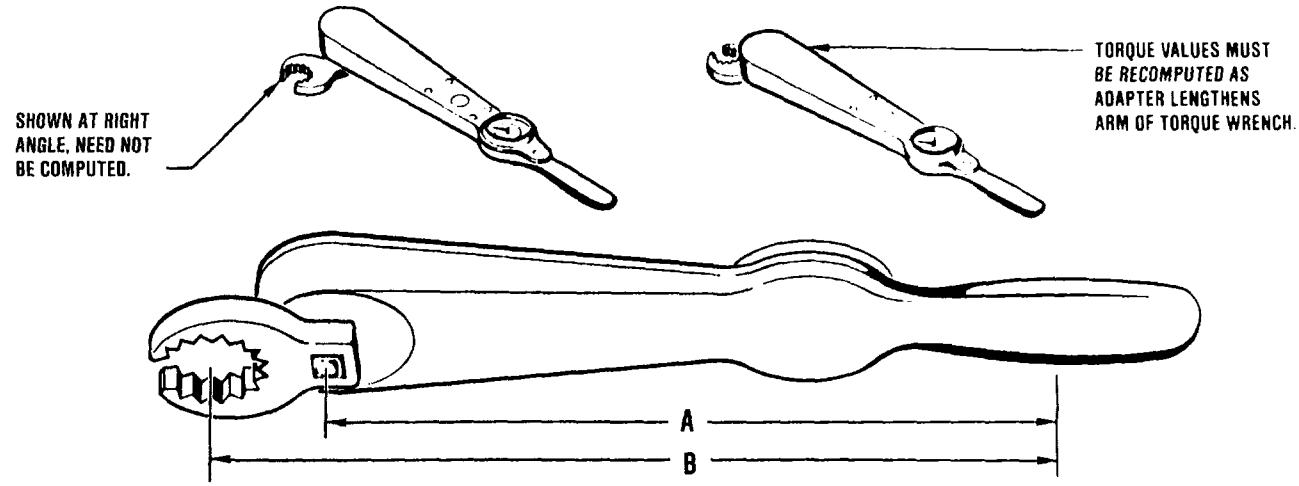
TORQUE WRENCHES AND CONVERTING TORQUE VALUES

Any type of torque wrench which is adequate for the job may be used for tightening "B" nuts. Table I lists several torque indicating handles that may be used.

Table II lists adapters (AN8508 crowfoot flare nut wrenches) which may be used with the torque indicating handles given in Table I.

The 1200 to 4800 inch-pounds torque indicating handle listed in Table I has a 3/4-inch square drive. Therefore, a square-drive adapter must be used when this handle is used with any of the AN8508 adapters listed in Table II.

Snap-type torque wrenches also may be used for tightening "B" nuts. Each of these wrenches is preset to a specific torque value.



CONVERTING TORQUE VALUE

Using any of the adapters listed in Table II lengthens the arm of the torque wrench. If such an adapter is used at right angles to the wrench handle as shown, it does not effectively lengthen the arm. However, if the adapter is used in such a way that it lengthens the arm of the torque wrench, torque values should be recomputed as follows:

1. Measure the length of the torque wrench from the center of the drive to the center of the handle. This is dimension A.
2. Measure the length of the torque wrench with the adapter attached. This measurement (dimension B) is taken from the center of the handle to the centerline of the adapter.
3. Divide dimension A by dimension B to get the conversion factor.
4. Multiply the conversion factor by the torque value given for the "B" nut to get the torque reading that should be used with the adapter on the wrench.

Example: Dimension A on a torque wrench is 12.5 inches. With an adapter installed, dimension B is 14.3 inches. A ² B = 12.5 ² 14.3 = 0.87. This is the conversion factor.

Let's assume the torque wrench with the adapter installed is to be used to tighten "B" nuts for which the torque value is 1325 inch-pounds. Therefore, 0.87 ×1325 = 1153. This is the torque reading to be used when this particular adapter is installed on the torque indicating handle and the torque called for is 1325 inch-pounds.

If the torque indicating handle and adapter in the preceding example are used, and the torque value is not recomputed, actual torque applied to the nut would be approximately 1516 inch-pounds. With a tolerance of ±125 inch-pounds, actual torque would exceed the maximum permissible torque by approximately 66 inch-pounds.

TABLE I — SOCKET WRENCH TORQUE INDICATING HANDLES

CAPACITY (IN. LB)	SQUARE DRIVE	SPECIFICATION	STOCK NO.
5-150	1/4	MIL-H-4034	5120-221-7944
100-750	3/8	MIL-H-4034	5120-904-1022
700-1600	1/2	MIL-H-4034	5120-270-3121
1200-4800	3/4	MIL-H-4034	5120-221-7945

TABLE II — AN8508 ADAPTERS (CROWFOOT FLARE NUT WRENCHES)

HEX SIZE	SQUARE DRIVE	PART NO.	STOCK NO.
9/16	1/4	AN8508A9	5120-541-4074
11/16	3/8	AN8508-11	5120-189-7896
7/8	3/8	AN8508-14	5120-181-6765
1-1/2	1/2	AN8508-24	5120-181-6755
2	1/2	AN8508-32	5120-181-6747
2-1/4	1/2	AN8508-36	5120-184-8385

Figure 20-00-2 — Torque Wrenches and Converting Torque Values

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DESCRIPTION (Continued)

TORQUE VALUES OF WIGGINS FITTINGS

Connector	Size									
	-8	-10	-12	-16	-20	-24	-28	-32	-40	
3605 (*) 3615 (*) 3655 (*) 3665 (*) 3608 (*) 3658 (*)	4	5	7	13	18	30	32	33	35	Torque In Foot-Pounds

* Indicates dash number of fitting.

NOTES

- *Refer to Illustrated Parts Catalog for component parts of each fitting.*
- *Lubricate O-rings with VV-P-236 Petrolatum.*
- *Lockwire nuts to each other or to the fixed (stationary) fitting.*

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CHAPTER

21

AIR CONDITIONING

CHAPTER 21

AIR CONDITIONING

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CHAPTER 21

AIR CONDITIONING

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21-32-01	101	Oct 11/76
21-32-02	201	Sep 2/88
	202	Sep 2/88
	203/204	Sep 2/88

* The asterisk indicates pages revised, added, or deleted by the current revision.

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21-60-01	101	Jul 6/73
21-60-02	201	Oct 11/76
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	202	Oct 11/76
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* The asterisk indicates pages revised, added, or deleted by the current revision.

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AIR CONDITIONING - DESCRIPTION

1. General

Air for the environment and pressurization system is extracted from the engine compressors. This bleed air is manifolded at each engine and is routed through the engine nacelle pylon into the fuselage rear section. At this point, the ducts from each engine are joined at one common duct which supplies the air for the environment and pressurization system. In this manner, either engine is capable of furnishing air to the complete system.

Air to the cabin and crew compartment is a mixture of bleed air which has been cooled by the refrigeration unit and bleed air which is routed directly from the extraction ports of the engine.

Air from the extraction ports of the left engine is manifolded, passed through an engine bleed-air shutoff valve and a reverse-flow check valve, and joined with the air which was similarly routed from the right engine. The engine bleed-air shutoff valves are energized to open, and are controlled by the CABIN AIR rotary switch located on the environmental control panel of the engine control lever pedestal in the cockpit. The shutoff valves may be closed separately if a downstream duct ruptures, if air contamination occurs, or if an engine fails. The reverse-flow check valves located at the "T" fitting, downstream of the engine bleed-air shutoff valves, protect against reverse flow during single-engine operation. (See figures 21-00-1 or 21-00-1A and 21-00-2 or 21-00-2A.)

Because the engine bleed air has been put to various uses, the air conditioning and pressurizing system consists of the following subsystems:

- Air supply
- Emergency air supply
- Ram-air supply
- Air distribution system
- Cabin pressurization control system
- Electronic nose compartment pressurization system
- Inflatable door seal system
- Cooling air system
- Electronic equipment compartment cooling system
- Cabin temperature control system
- Cockpit temperature control system - Aircraft NA-370-1, -7, NA-380-1 and later
- Inflatable wheel seal system - Aircraft NA-370-1 through -8 not changed by Service Bulletin 72-8

The components of these systems and their locations are as follows:

ORIGINAL AS
RECEIVED

COMPONENT NOMENCLATURE	NO. PER AIRPLANE	LOCATION
Engine bleed-air shutoff valve	2	Right and left engine compressor bleed-air supply ducting
Emergency bleed-air shutoff valve	1	Right engine pylon, station 363
Emergency bleed-air overpressure switch	1	Rear section of fuselage, station 357, right side
Emergency ram-air supply valve	1	Rear section of fuselage, in duct downstream from ram-air inlet duct
Flow limiter valve	1	Rear section of fuselage, upstream from refrigeration unit

COMPONENT NOMENCLATURE	NO. PER AIRPLANE	LOCATION
Cabin flood flow valve	1	Rear section of fuselage, downstream from heat exchanger
Rate-of-change control	1	Cockpit/right console or center pedestal
Isobaric control and ground test valve	1	Fuselage station 30, right side
Cabin pressure regulator	1	Fuselage station 76, right side
Pneumatic relay	1	Fuselage station 85, right side
Cabin air dump and safety valve	1	Fuselage station 76, left side
Air filter	1	Fuselage station 85, right side
Limit control	1	Fuselage station 85, right side
Cabin air dump solenoid control valve	1	Fuselage station 76, left side
Nose compartment absolute-pressure controller 2	1	Fuselage station 32, left side
Nose compartment pressure regulator 2	1	Fuselage station 50, left side
Nose compartment low pressure switch 2	1	Fuselage station 45, center
Nose compartment relief valve 2	1	Fuselage station 45, right side
Door seal pressure regulator	1	Rear section of fuselage, on bulkhead, left side, aft of refrigeration unit
Door seal pressure reservoir	1	Rear section of fuselage, left side, outboard of refrigeration unit
Door seal manually operated dump valve	1	Bulkhead aft of entrance door
Water separator	1	Rear section of fuselage, part of refrigeration unit
Water separator delta P sensor 1	1	Rear section of fuselage, above water separator
Water separator anti-ice valve 1	1	Rear section of fuselage, upstream of water separator
Refrigeration unit	1	Rear section of fuselage
Ground cooling blower	1	Fuselage station 25, center
Nose compartment check valve	1	Fuselage station 23, center
Cabin hot-air mixing valve	1	Rear section of fuselage, center, inboard and forward of refrigeration unit, station 360

1 S/N 370-2 thru -6, -8 and 9
2 S/N 380-1 thru -23 Airplanes not changed by S/L 7 or S/B 74-1

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COMPONENT NOMENCLATURE	NO. PER AIRPLANE	LOCATION
Vacuum pump	1	Fuselage nose section station 30
Vacuum regulator	1	Fuselage station 99, right side
Vacuum regulator valve	1	Center pedestal
Dehydrator	1	Fuselage station 365, right side
Filter	1	Fuselage station 365, right side
35°F temperature control valve [2]	1	Fuselage rear section on refrigeration unit
Pressure regulator [2]	1	Fuselage rear section
35°F temperature control sensor [2]	1	Fuselage station 364 RH
Cockpit temperature control unit [2]	1	Fuselage station 194 RH
Cockpit duct air temperature sensor [2]	1	Fuselage station 333 RH
Cockpit hot-air mixing valve [2]	1	Fuselage station 357
Cabin air temperature control	1	Center pedestal
Cockpit air temperature control [2]	1	Center pedestal
Cabin air temperature sensor/blower	1	Aft side of forward cabin bulkhead — LH

[2] S/N 370-1, -7, and S/N 380-1 and Subs

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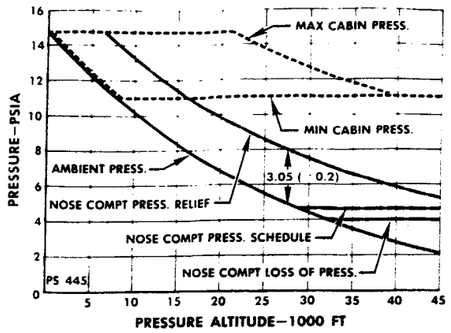
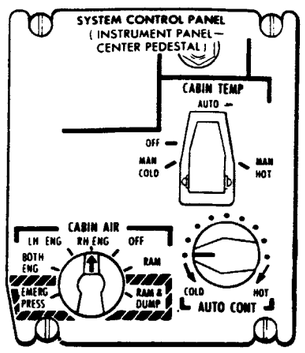
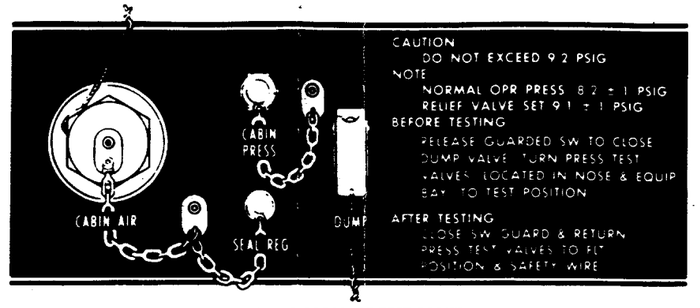




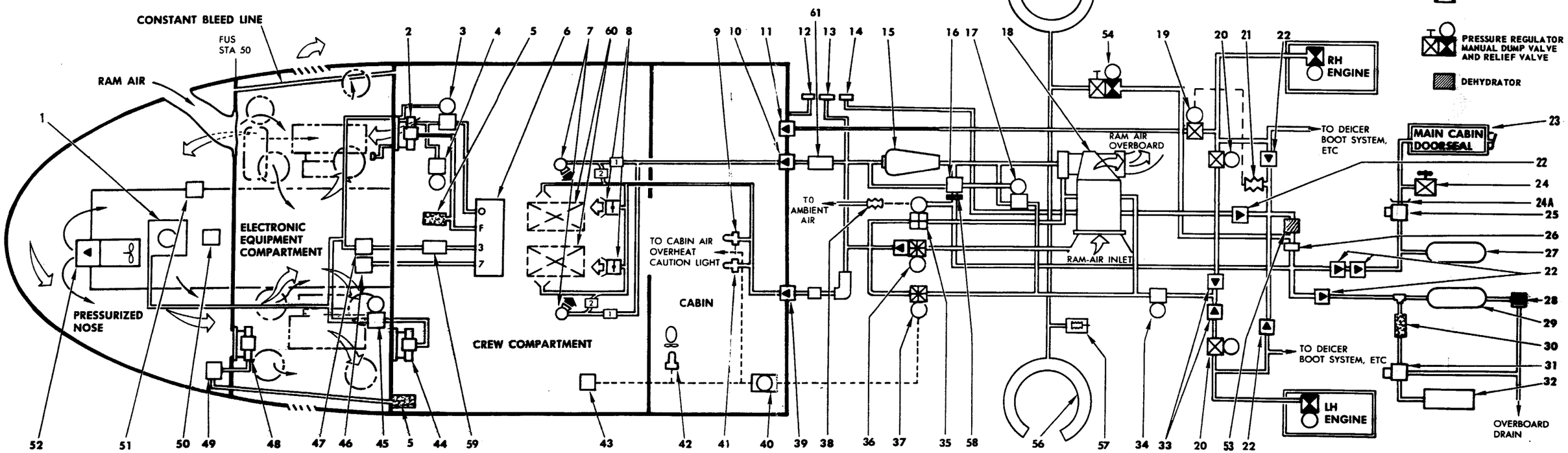

CABIN AIR
CABIN PRESS
SEAL REG
DUMP
CAUTION
DO NOT EXCEED 9.2 PSIG
NOTE
NORMAL OPR PRESS 8.2 ± 1 PSIG
RELIEF VALVE SET 9.1 ± 1 PSIG
BEFORE TESTING
RELEASE GUARDED SW. TO CLOSE DUMP VALVE. TURN PRESS TEST VALVES LOCATED IN NOSE & EQUIP BAY TO TEST POSITION
AFTER TESTING
CLOSE SW. GUARD & RETURN PRESS TEST VALVES TO FLT POSITION & SAFETY WIRE

GROUND PRESSURE TEST PANEL

[1] AIRPLANES NOT CHANGED BY SB 74-27

[2] AIRPLANES CHANGED BY SB 74-27

ELECTRICALLY OPERATED VALVE
PNEUMATICALLY OPERATED VALVE
SOLENOID-PILOTED PNEUMATIC VALVE
PNEUMATIC VALVE WITH MANUAL SOLENOID PNEUMATIC PRESSURE CONTROL
BUTTERFLY (MANUAL)
PRESSURE SWITCH
PRESSURE LIMITING AND SHUTOFF VALVE
VENTURI-TYPE JET PUMP
AIR FILTER
HEAT SENSING ELECTRICAL INDICATION
CHECK VALVE
DUMP VALVE (MANUAL)
PRESSURE REGULATOR MANUAL DUMP VALVE AND RELIEF VALVE
DEHYDRATOR

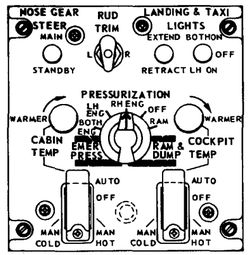


1 VACUUM PUMP
2 CABIN PRESSURE REGULATOR
3 PNEUMATIC RELAY
4 LIMIT CONTROL
5 AIR FILTER
6 RATE-OF-CHANGE CONTROL
7 COCKPIT COOLING-AIR OUTLET
8 AIR EYEBALL OUTLETS
9 DUCT AIR TEMPERATURE SENSOR
10 CABIN COOLING-AIR SUPPLY CHECK VALVE
11 EMERGENCY CABIN AIR PRESSURE SUPPLY CHECK VALVE
12 GROUND FITTING – CABIN PRESSURE
13 GROUND FITTING – CABIN AIR
14 GROUND FITTING – SEAL REGULATOR
15 WATER SEPARATOR
16 WATER SEPARATOR DELTA P SENSOR
17 WATER SEPARATOR ANTI-ICE VALVE
18 REFRIGERATION UNIT
19 EMERGENCY BLEED-AIR SHUTOFF VALVE
20 ENGINE BLEED-AIR SHUTOFF VALVE
21 EMERGENCY BLEED-AIR OVERPRESSURE SWITCH
22 PNEUMATIC PRESSURE CHECK VALVE
23 MAIN CABIN DOOR SEAL
24 MANUALLY OPERATED DUMP VALVE
24A DISCONNECT FOR DOOR SEAL TEST
25 DOOR SEAL PRESSURE REGULATOR
26 FILTER
27 DOOR SEAL PRESSURE RESERVOIR
28 MANUALLY OPERATED AIR DRAIN VALVE
29 AIR RESERVOIR – HYDRAULIC SYSTEM
30 AIR FILTER – HYDRAULIC SYSTEM
31 AIR PRESSURE REGULATOR – HYDRAULIC SYSTEM
32 PRESSURIZED HYDRAULIC RESERVOIR
33 REVERSE-FLOW CHECK VALVES
34 FLOW LIMITER VALVE
35 CABIN FLOOD FLOW VALVE
36 EMERGENCY RAM-AIR SUPPLY VALVE
37 HOT-AIR MIXING VALVE
38 ALTITUDE PRESSURE SWITCH
39 CABIN AIR SUPPLY CHECK VALVE
40 CABIN AIR TEMPERATURE CONTROL BOX
41 CABIN AIR SUPPLY OVERHEAT THERMOSWITCH
42 CABIN AIR TEMPERATURE SENSOR & BLOWER
43 CABIN AIR TEMPERATURE SELECTOR
44 CABIN AIR DUMP AND SAFETY VALVE
45 CABIN AIR DUMP SOLENOID CONTROL VALVE
46 CABIN AIR ISOBARIC CONTROL AND GROUND TEST VALVE
47 GROUND TEST SHUTOFF VALVE
48 NOSE COMPARTMENT PRESSURE REGULATOR
49 NOSE COMPARTMENT ABSOLUTE-PRESSURE CONTROLLER
50 NOSE COMPARTMENT LOW-PRESSURE SWITCH
51 NOSE COMPARTMENT PRESSURE RELIEF VALVE
52 ELECTRONIC EQUIPMENT COOLING-AIR BLOWER & CHECK VALVE
53 DEHYDRATOR
54 PRESSURE REGULATOR MANUAL DUMP AND RELIEF VALVE *
55 RIGHT HAND WHEEL SEAL *
56 LEFT HAND WHEEL SEAL *
57 PRESSURE SWITCH *
58 FILTER
59 VACUUM REGULATOR
60 HEATING PAD
61 COLD AIR DUCT MUFFLER
62 CONDITIONED AIR MUFFLED
* NA-370-2 THROUGH -6, AND -8 AIRPLANES NOT CHANGED BY SERVICE BULLETIN 728

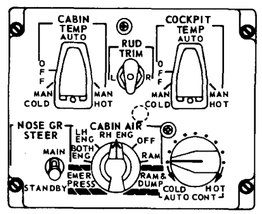
S370-2-53-3B

Air Conditioning System
Aircraft NA-370-2 through -6, -8 and -9
Figure 21-00-1

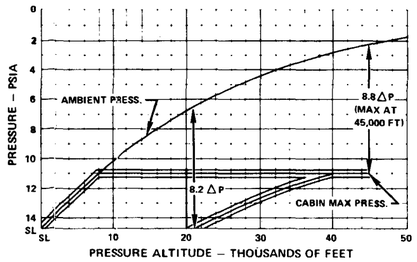
SEE BLOW-UP FICHE NO. ASE0411-ITEM A



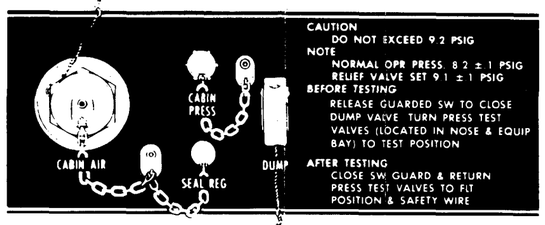
NA-370-1, -7 AND NA-380-4 AND LATER



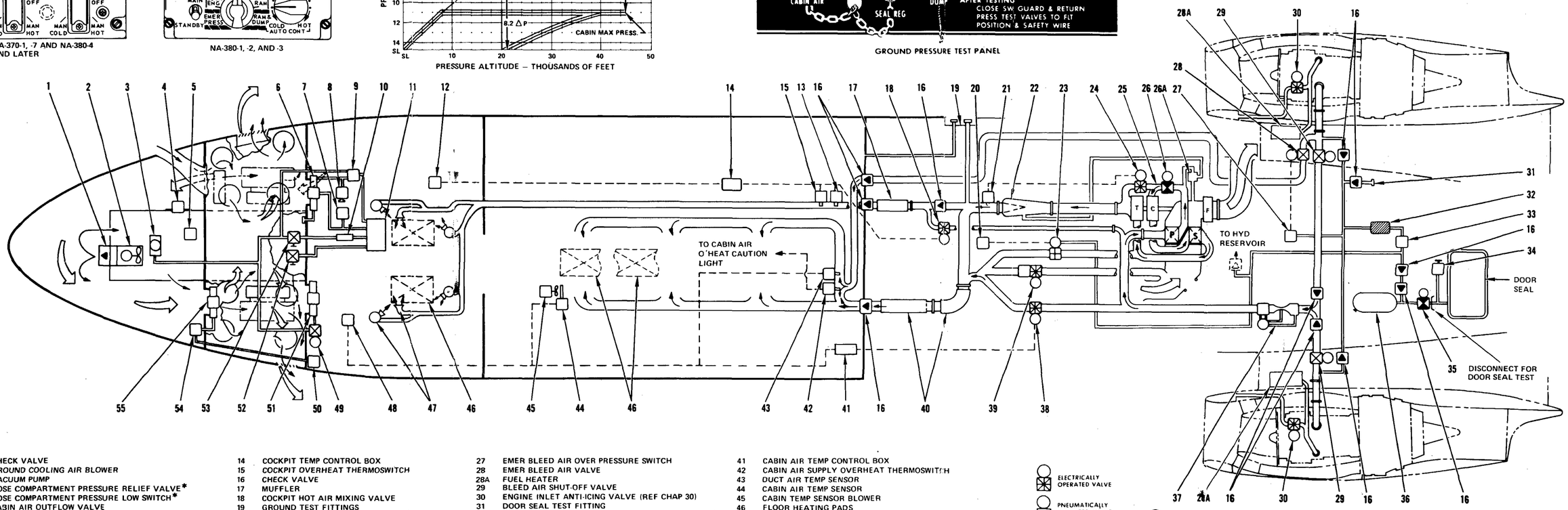
NA-380-1, -2, AND -3



PRESSURE ALTITUDE – THOUSANDS OF FEET


CABIN AIR
CABIN PRESS
SEAL REG
DUMP
CAUTION
DO NOT EXCEED 9.2 PSIG
NOTE
NORMAL OPR PRESS 8.2 ± 1 PSIG
RELIEF VALVE SET 9.1 ± 1 PSIG
BEFORE TESTING
RELEASE GUARDED SW TO CLOSE DUMP VALVE TURN PRESS TEST VALVES (LOCATED IN NOSE & EQUIP BAY) TO TEST POSITION
AFTER TESTING
CLOSE SW GUARD & RETURN PRESS TEST VALVES TO FLT POSITION & SAFETY WIRE

GROUND PRESSURE TEST PANEL



1 CHECK VALVE
2 GROUND COOLING AIR BLOWER
3 VACUUM PUMP
4 NOSE COMPARTMENT PRESSURE RELIEF VALVE*
5 NOSE COMPARTMENT PRESSURE LOW SWITCH*
6 CABIN AIR OUTFLOW VALVE
7 AIR FILTER
8 LIMIT CONTROL
9 PNEUMATIC RELAY
10 VACUUM REGULATOR
11 CABIN PRESS. & RATE OF CHANGE CONTROL
12 COCKPIT AIR TEMP SELECTOR
13 COCKPIT DUCT AIR TEMP SENSOR
14 COCKPIT TEMP CONTROL BOX
15 COCKPIT OVERHEAT THERMOSWITCH
16 CHECK VALVE
17 MUFFLER
18 COCKPIT HOT AIR MIXING VALVE
19 GROUND TEST FITTINGS
20 ALTITUDE PRESSURE SWITCH
21 35° F TEMPERATURE CONTROL/SENSOR
22 WATER SEPARATOR
23 FLOOD FLOW VALVE
24 35° TEMPERATURE CONTROL VALVE
25 3 WHEEL REFRIGN PACK
26 PRESSURE REGULATOR
26A WATER INJECTOR (ASPIRATOR)
27 EMER BLEED AIR OVER PRESSURE SWITCH
28 EMER BLEED AIR VALVE
28A FUEL HEATER
29 BLEED AIR SHUT-OFF VALVE
30 ENGINE INLET ANTI-ICING VALVE (REF CHAP 30)
31 DOOR SEAL TEST FITTING
32 DEHYDRATOR
33 FILTER
34 RELIEF VALVE
35 DOOR SEAL PRESSURE REGULATOR
36 DOOR SEAL RESERVOIR
37 FLOW LIMIT CONTROL
38 CABIN HOT AIR MIX VALVE
39 EMER RAM AIR SUPPLY VALVE
40 MUFFLERS
41 CABIN AIR TEMP CONTROL BOX
42 CABIN AIR SUPPLY OVERHEAT THERMOSWITCH
43 DUCT AIR TEMP SENSOR
44 CABIN AIR TEMP SENSOR
45 CABIN TEMP SENSOR BLOWER
46 FLOOR HEATING PADS
47 AIR OUTLETS
48 CABIN AIR TEMP SELECTOR
49 CABIN DUMP SOLENOID CONTROL VALVE -
50 FILTER*
51 CABIN AIR DUMP & SAFETY VALVE
52 CABIN AIR ISOBARIC CONTROL & GROUND TEST VALVE
53 GROUND TEST SHUTOFF VALVE
54 NOSE COMPARTMENT ABSOLUTE PRESSURE CONTROLLER*
55 EQUIPMENT BAY OUTFLOW VALVE*

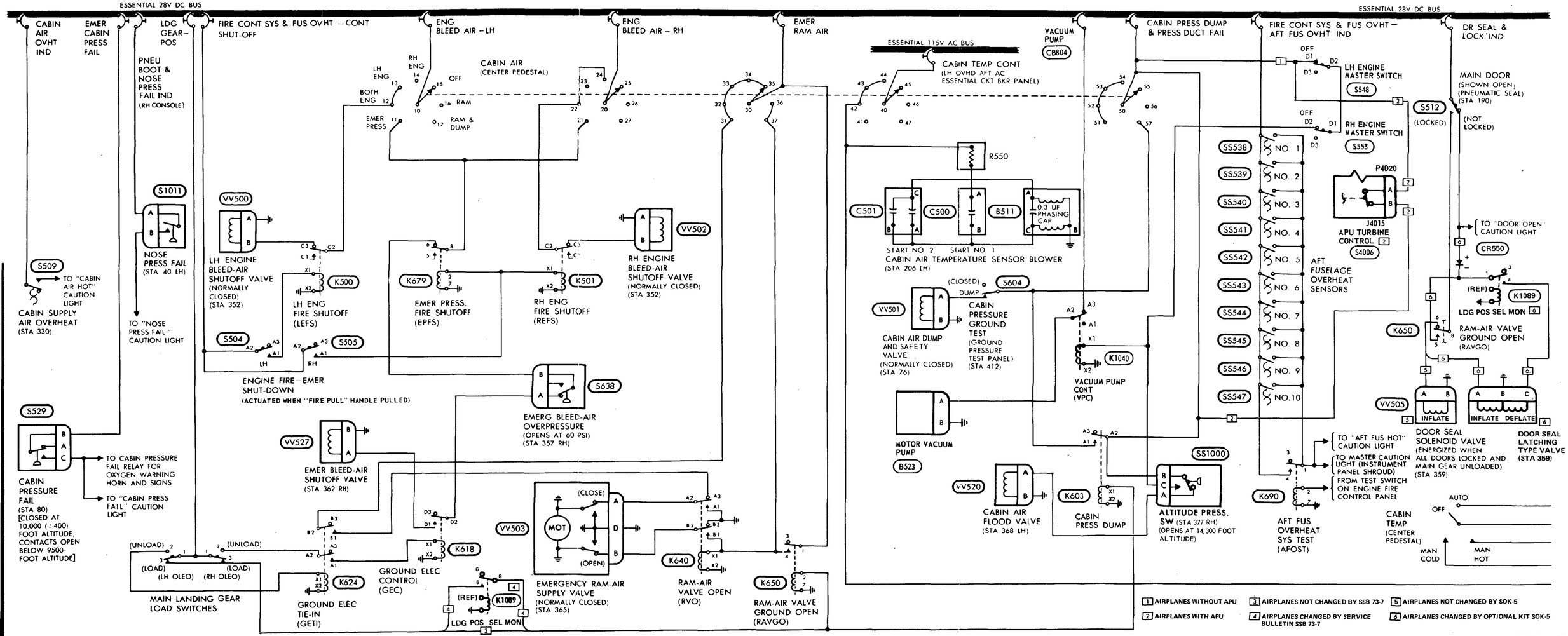
ELECTRICALLY OPERATED VALVE
PNEUMATICALLY OPERATED VALVE
SOLENOID PILOTED PNEUMATIC VALVE
PNEUMATIC VALVE WITH MANUAL SOLENOID PNEUMATIC PRESSURE CONTROL
PRESSURE LIMITING AND SHUTOFF VALVE
CHECK VALVE
DUMP VALVE (MANUAL)

T-EXPANSION TURBINE
C-COMPRESSOR
F-FAN-DRAWS AMBIENT AIR OVER SECONDARY HEAT EXCHANGER
P-PRIMARY HEAT EXCHANGER
S-SECONDARY HEAT EXCHANGER

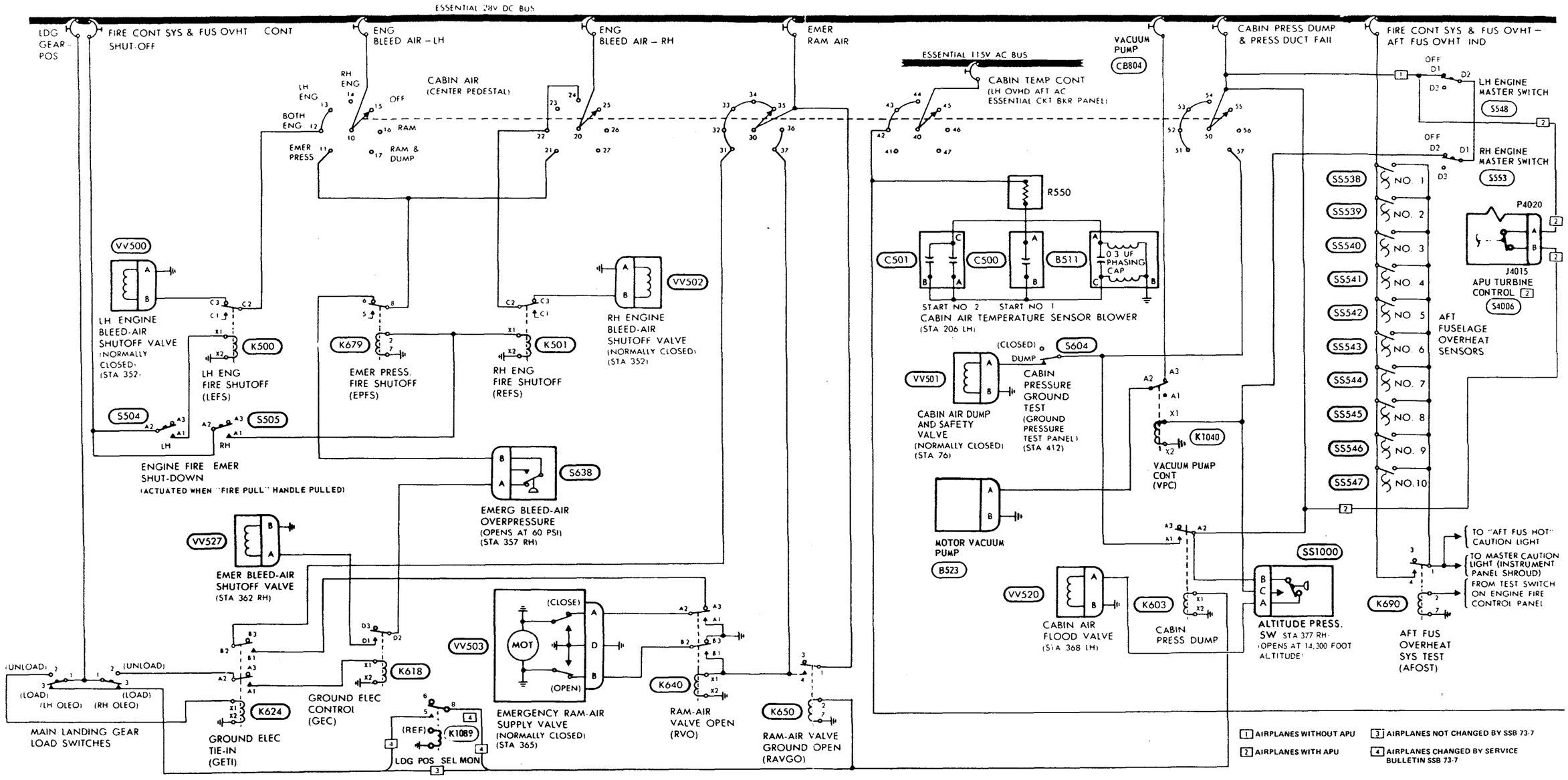
* ON AIRPLANES NA-370-1, -7 AND NA-380-1 THRU -23 THAT ARE NOT CHANGED BY SSL-7

S380-2-53-1A

Air Conditioning System -
Aircraft NA-370-1, -7, NA-380-1 and Later
Figure 21-00-1A

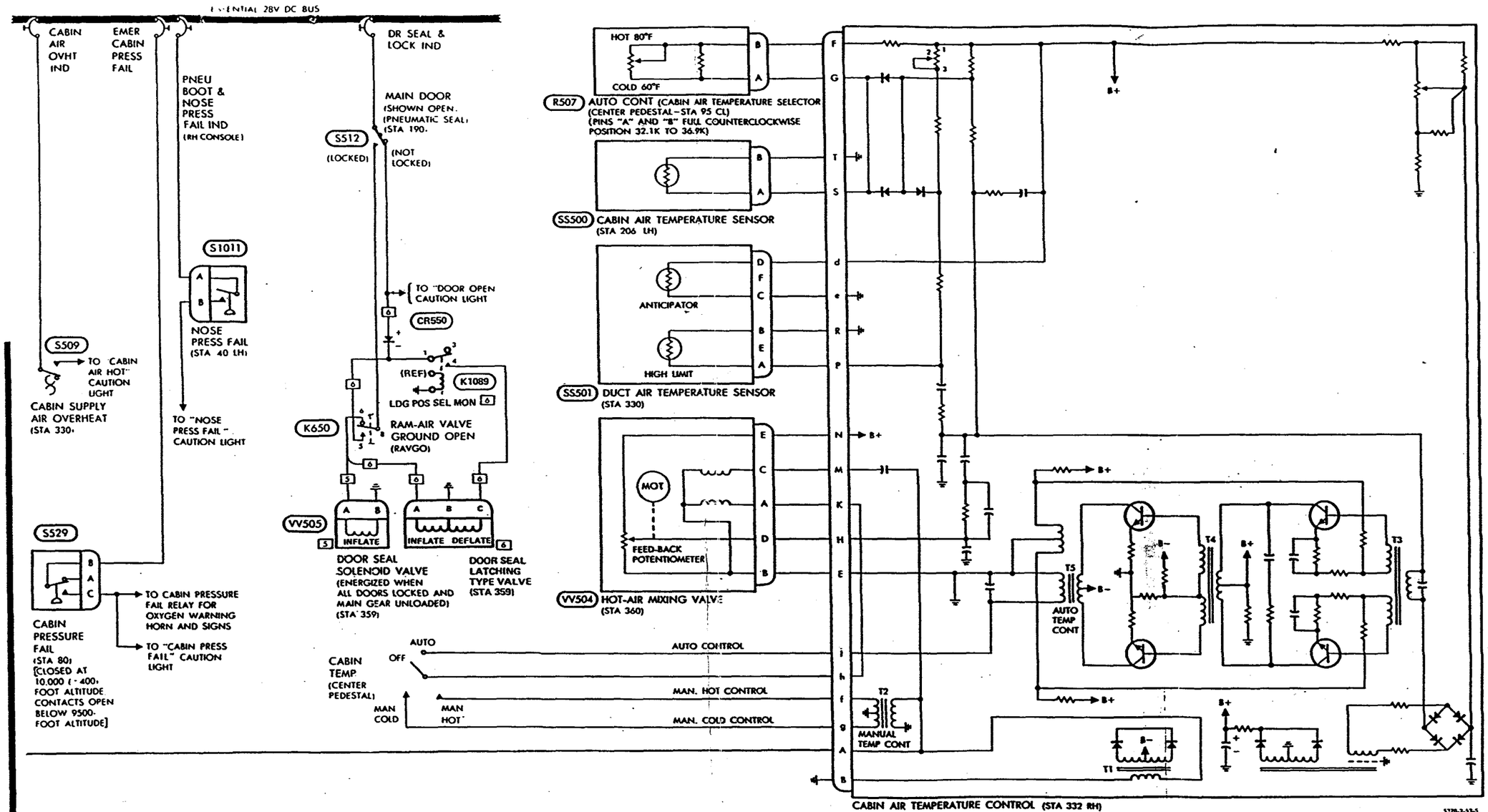


Air Conditioning System Electrical Schematic -
Aircraft NA-370-2 Through -6, -8 and -9
Figure 21-00-2 (Sheet 1 of 2)



Air Conditioning System Electrical Schematic
Aircraft NA 370-2 thru -6, -8 and -9
Figure 21-00-2 (Sheet 1 of 2)

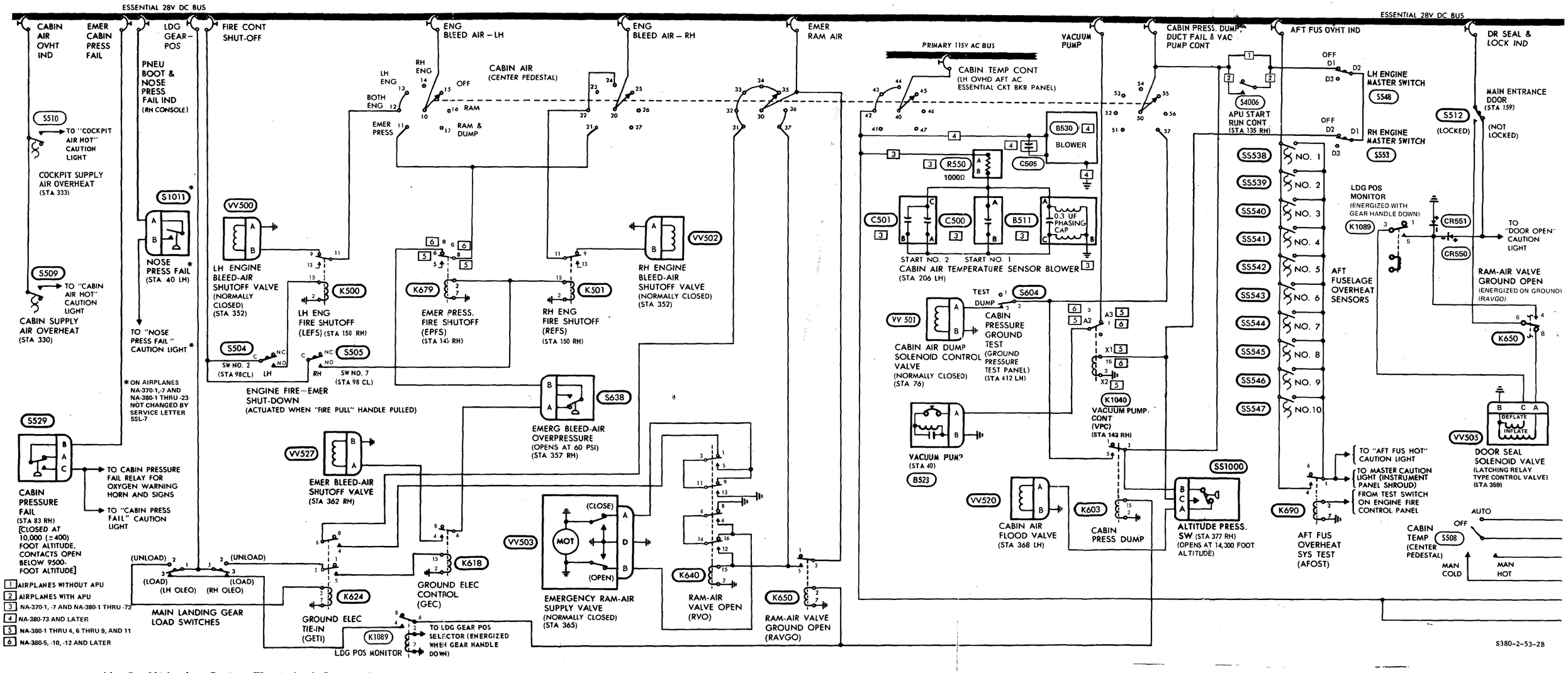
Oct 11/76



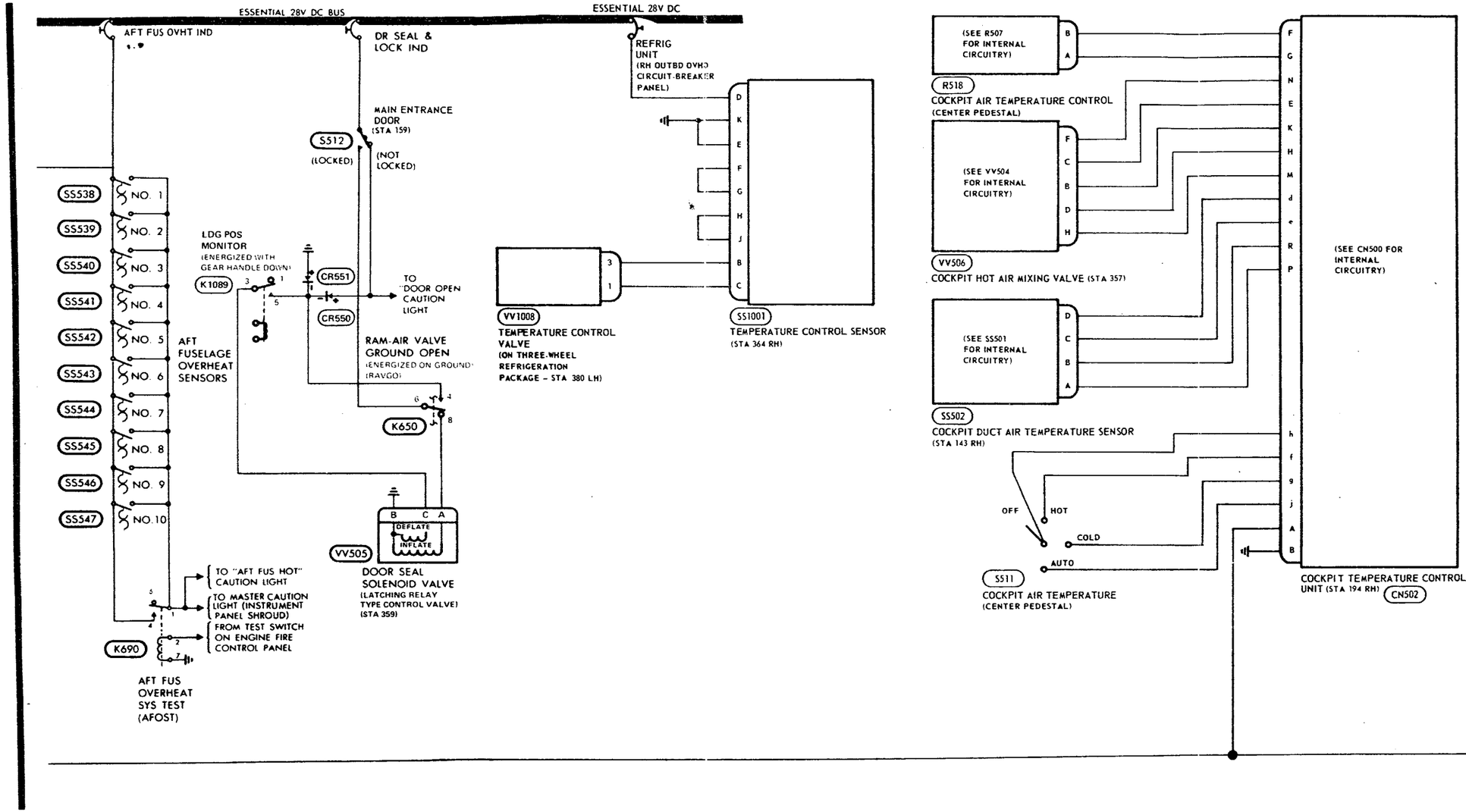
[5] AIRPLANES NOT CHANGED BY SOK-5

[6] AIRPLANES CHANGED BY OPTIONAL KIT SOK-5

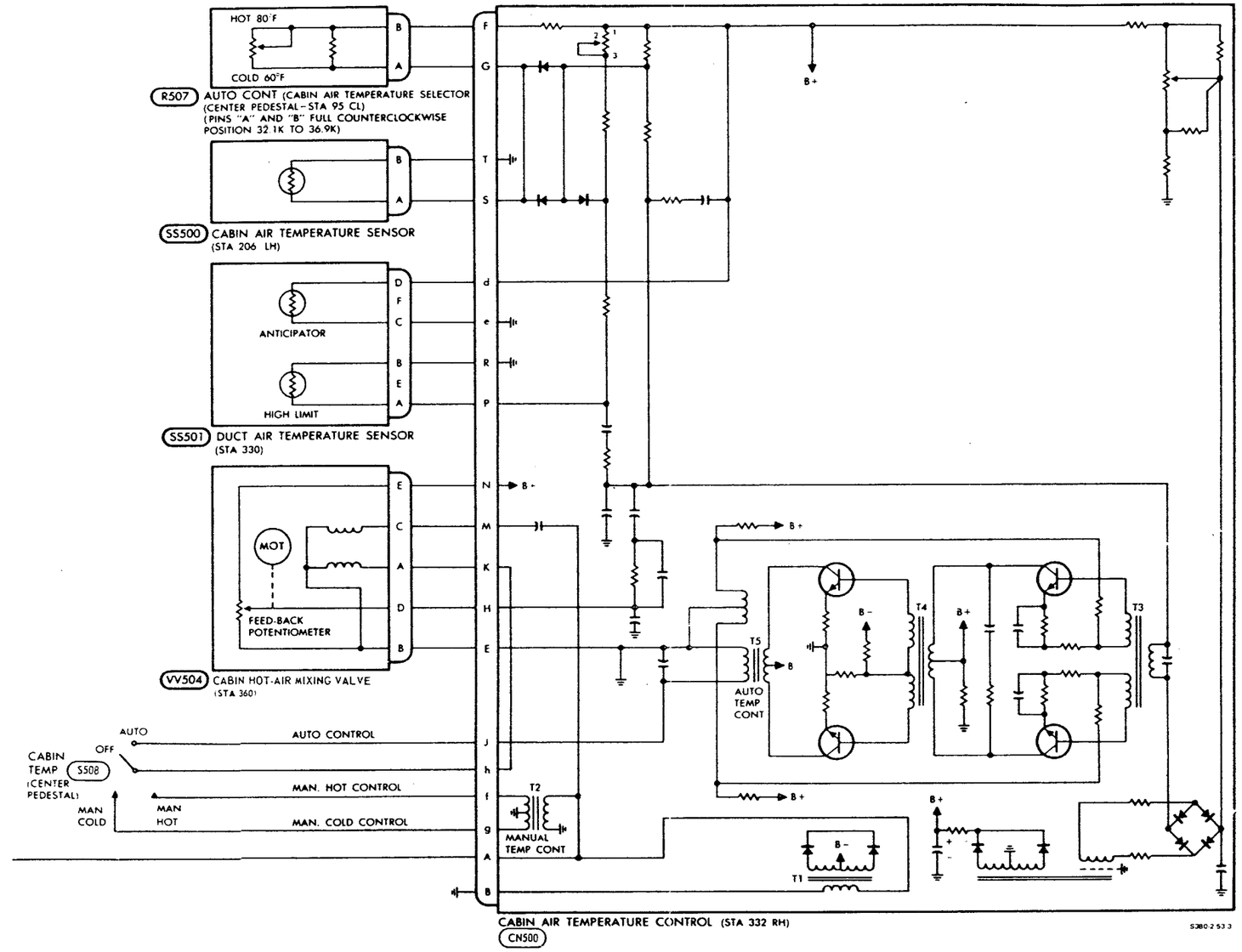
Air Conditioning System Electrical Schematic
Aircraft NA 370-2 thru -6, -8 and -9
Figure 21-00-2 (Sheet 2 of 2)



Air Conditioning System Electrical Schematic - Aircraft NA-370-1, -7, NA-380-1 and Later
Figure 21-00-2A (Sheet 1 of 2)



Air Conditioning System Electrical Schematic
Aircraft NA 370-1, -7, NA 380-1 and Later
Figure 21-00-2A (Sheet 2 of 3)



Air Conditioning System Electrical Schematic
Aircraft NA 370-1, -7, NA 380-1 and Later
Figure 21-00-2A (Sheet 3 of 3)

COMPRESSION

DESCRIPTION AND OPERATION

1. AIR SUPPLY

Engine bleed-air comes from the engine compressor through extraction ports on each engine. These ports are externally manifolded at each engine and form a single duct which is routed through the engine pod pylon into the fuselage rear section. At this point, the single ducts from each engine are joined to one common duct, supplying air for the air conditioning and pressurizing systems. In this manner, either engine is capable of maintaining the air supply to the various systems.

The major portion of the engine bleed air flows to the flow-limiting control valve. Downstream from the flow-limiting valve, the air flow branches. A portion of it goes to the refrigeration unit to be cooled; the remainder is used as heating air. Engine bleed air is also ducted to the emergency bleed-air shutoff valve, cabin flood flow valve and differential-pressure sensing valve, inflatable door seal system, de-icer boot system, and hydraulic reservoir air pressure system.

2. EMERGENCY AIR SUPPLY

The CABIN PRESS FAIL warning light and the AFT FUS HOT caution light provide all necessary duct failure warning indications. If leakage from the pressurization air supply duct becomes excessive in the fuselage rear section, sensing switches near the supply ducts may light the AFT FUS HOT caution light. With either indication of trouble, the cabin pressurization selector switch should be positioned to EMER PRESS; this opens the emergency bleed-air shutoff valve and admits engine bleed-air from upstream of the right engine bleed-air shutoff valve only, through a smaller line and check valve directly into the cabin, thus providing emergency pressurization. An emergency bleed-air overpressure switch closes the emergency bleed-air shutoff valve if the pressure exceeds 60 (±5) psi. Pressure about 60 psi would supply excessive temperature. The maximum temperature may be 350°F to the cabin.

3. RAM AIR SUPPLY

The ram air system is provided to cool the engine bleed-air as it passes through the heat exchanger of the refrigeration unit, to furnish an emergency source of air to the cabin if engine failures occur, and to scavenge fumes or contaminated air from the cabin. When the airplane is on the ground (with no aerodynamically induced ram air), air is drawn through the heat exchanger by the compressor fan. This will make the system operational to provide cooling. Engine speed may have to be increased above idle, on a hot day, to provide a cabin temperature of 75°F.

The ram air, used as an emergency air source, is directed into the main air supply ducting and is routed to the cabin through the normal pressure distribution ducting.

The ram air source is routed from the ram-air inlet scoop to an emergency ram air supply valve and through a pneumatic pressure check valve, from where it flows into the main air supply line to the cabin. The emergency ram air supply valve opens when the pilot selects the RAM & DUMP position of the CABIN AIR control.

4. ENGINE BLEED-AIR SHUTOFF VALVE

The engine bleed-air shutoff valves are installed in each engine bleed-air supply duct. The valves are solenoid-piloted, pneumatically actuated, and the fail-safe position is closed.

At idle rpm, these normally closed valves open in approximately 4.5 seconds upon actuation of the CABIN AIR rotary switch. Positioning the CABIN AIR switch at RH ENG opens the right engine bleed-air shutoff valve, positioning at LH ENG opens the left engine bleed-air shutoff valve, and positioning at BOTH ENG opens both (right and left) engine bleed-air valves. A valve position indicator consists of a pin which protrudes from the actuator housing when the valve is open. If the valve should fail to open within the allowable time, the valve should be removed and the air-bleed hole cleaned. The valves are closed separately if an engine fire occurs, by use of the fire pull engine fire emergency shut-down controls. The valves close in 0.25 second. Electrical power for operation of engine bleed-air shutoff valves is obtained from either the essential bus or the standby power bus (on NA 265-70 airplanes equipped with a standby power bus.)

5. ENGINE BLEED-AIR DUCT REVERSE-FLOW CHECK VALVES

The engine bleed-air reverse-flow check valves are installed in the inboard ends of the right and left engine bleed crossover ducts. The valves are accessible by removing the engine bleed crossover T duct. The valve assemblies are threaded into each duct, and seat against a washer and O-ring seal. The assembly includes a ring and valve seat asssembly on which four spring-loaded flapper valves operate. Two 1/8-inch holes are provided in the ring to accommodate a spanner wrench necessary for removal and installation.

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DESCRIPTION AND OPERATION (Continued)

6. EMERGENCY BLEED-AIR SHUTOFF VALVE

The emergency bleed-air shutoff valve is similar to, but smaller than, the engine bleed-air shutoff valves. (Refer to description of engine bleed-air shutoff valves.) It has no position indicator. It is used, when necessary, to bleed air from the right engine only and is positioned upstream from the right engine bleed-air valve.

The emergency bleed-air shutoff valve is opened when the airplane is airborne and the CABIN AIR switch is positioned at EMERG PRESS. The emergency bleed-air shutoff valve cannot be actuated when the airplane is on the ground because the ground electrical tie-in (GETI) relay (K624) and ground electrical control (GEC) relay (K618) are not energized until the main gear struts are extended. When the valve is in the open position, it can be closed when the right-hand ENGINE FIRE—EMER SHUT-DOWN handle is pulled or if the emergency overpressurization switch opens because of pressure above 60 psi.

7. EMERGENCY BLEED-AIR OVER-PRESSURE SWITCH

The emergency bleed-air overpressure switch is in the fuselage rear section, station 357, right side. It is diaphragm-actuated and pressure-controlled, differentiating between ambient air and the air pressure within the emergency air supply plumbing.

When the CABIN AIR switch is actuated to EMER PRESS, power from the ENG BLEED AIR — LH and ENG BLEED AIR — RH circuit breakers is routed through the unenergized contacts of the emergency pressure fire shutoff (EPFS) relay (K679), the normally closed contacts of the emergency bleed-air overpressure switch (opens at 60 psi), and the energized contacts of the ground electrical control (GEC) relay, to the emergency bleed-air shutoff valve. The emergency bleed-air shutoff valve is actuated open. When the emergency bleed-air overpressure switch senses an overpressure condition, the circuit is broken and the emergency bleed-air shutoff valve will go to a closed position.

8. EMERGENCY RAM AIR SUPPLY VALVE

The electrically operated emergency ram air supply valve is in the aft equipment compartment. It is installed in a duct that is connected to the refrigeration unit ram air supply duct. It is downstream from the ram air inlet duct in the fuselage rear section. When the normally closed valve has been actuated open, ram air flows into the duct that carries conditioned air to the cabin. Positioning the CABIN AIR switch at RAM or RAM & DUMP actuates the ram air supply valve to open, but the RAM & DUMP position also actuates the cabin air dump and safety valve to dump cabin air pressure.

This valve is a motor-operated iris-type valve combined with sectored swing check valves. The iris valve consists of six blades or disks actuated by a ring gear (similar to a camera stop adjustment ring) connected to a gear train and motor. Limit switches stop the motor at end-of-travel. The check valve sector flappers are hinged at the circumference and swing open from the center to the side of the body. They are spring-loaded closed. Their function is to prevent loss of conditioned air if the iris valve should remain open and to prevent reverse loading of the iris blades.

9. DOOR SEAL AND SENSING AIR DEHYDRATOR

The door seal and sensing air dehydrator is a silica-gel filled container mounted in the aft fuselage equipment compartment. The dehydrator collects and prevents moisture from entering the entrance door seal systems and distribution system sensing lines. The dehydrator must be inspected at periodic intervals and refilled with silica-gel when required. An in-line filter is installed at the output of the dehydrator.

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COMPRESSION - TROUBLE SHOOTING

1. Trouble Shooting Ram-air Supply

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. NOT ENOUGH FRESH AIRFLOW TO CABIN WHEN RAM AIR IS SELECTED.		
(1) Faulty emergency ram-air supply valve or electrical control circuit, or obstruction in ram-air inlet ducting.	See figure 21-00-2 or 21-00-2A and check electrical control circuit to valve. If power is available at valve disconnect plug and valve does not respond, valve is faulty. Check for ram-air inlet duct obstruction.	Replace faulty emergency ram-air supply valve, or repair electrical control circuit, as required.
(2) Control relay malfunction.	Check relay for burned points.	Replace relay.

ORIGINAL AS
RECEIVED

COMPRESSION

MAINTENANCE PRACTICES

REPLACEMENT OF ENGINE BLEED-AIR SHUTOFF VALVE

Removing Engine Bleed-air Shutoff Valve

1. Remove accesses on lower sides of either inboard pylon.
2. Disconnect electrical control wires.
3. Loosen clamps at both ends of valve and slide clamps back on ducting, allowing valve to be removed.
4. Cover openings of valve and ducting to prevent entry of foreign material.

Cleaning Engine Bleed-air Shutoff Valve

1. Remove valve from airplane. (Refer to procedure for removing engine bleed-air shutoff valve.)
2. Wash valve body in dry-cleaning solvent (Stoddard). Locate 0.026-inch bleed hole in nose cone in upstream side of valve.
3. Clean out bleed hole. (Use a 0.025-inch wire, or smaller.)
4. Install valve. (Refer to procedure for installing engine bleed-air shutoff valve.)
5. Check valve operation time by feeling for protrusion of indicator on valve when valve is operating. Normal operating time is 0.05 second; maximum operating time is 2.4 seconds.

Installing Engine Bleed-air Shutoff Valves

1. Remove protective covering from valve and ducting.
2. Place gaskets in position on valve flanges. New gaskets (Marman 51134-175-A) should be used for each installation.
3. Place valve in position between ducts, the valve actuator in a horizontal position and extending inboard, its arrow pointing aft.
4. Install two coupling clamps (Marman 51394H-175) with bolts parallel to pylon skin, and torque them to 45-55 inch-pounds.
5. Connect electrical wires, and install terminal cover.

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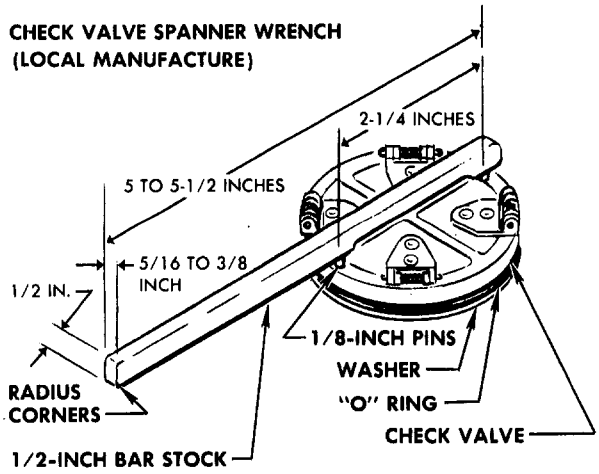


5370-2-53-7

Check Valve Spanner Wrench (Locally Manufactured)
Figure 21-10-401

2. Replacement of Engine Bleed-air Duct Reverse-flow Check Valves

A. Removing Engine Bleed-air Duct Reverse-flow Check Valves

(1) Open aft fuselage access door.

(2) On NA-370 airplanes, remove battery.

(3) Remove insulation cover (when present) and Marman clamp on left end of flow-limiting valve.

(4) Remove clamp insulation covers (when present) and Marman clamps connecting right and left engine bleed-air ducts to "T" duct.

(5) Remove the two AN3 bolts holding "T" duct to bracket.

(6) Separate ducts and carefully work crossover "T" duct with valve out from between right and left ducts, without damaging check valves in right and left ducts.

(7) Using spanner wrench (figure 21-10-401), turn check valves counter-clockwise and remove valves.

(8) Lift out O-ring seal and washer, and check condition for re-use.

B. Installing Engine Bleed-air Duct Reverse-flow Check Valves

(1) Place washer, then O-ring seal, in air duct.

(2) Turn check valve assembly into air duct. Use spanner wrench. Valve ring should be flush with flanges of duct coupling. Do not over-tighten.

(3) Visually check to see that valve flappers all seat properly after installing.

(4) Insert "T" duct with valve attached between left and right engine bleed-air ducts.

(5) Carefully place Conoseal gaskets in coupling flanges of "T" duct and pressure-limiting valve.

(6) Install Marman clamps over coupling flanges with clamp bolts positioned so that lap on insulation covers will not catch liquids.

(7) Install the two AN3 bolts through duct support and bracket.

(8) Torque "T" duct and valve coupling clamps to 60-70 inch-pounds.

(9) Install duct clamp insulation covers with split in cover to the bottom.

(10) Secure clamp insulation covers with two clamps each.

(11) Reinstall any other equipment removed.

(12) Close accesses.

3. Replacement of Emergency Bleed-air Shutoff Valve

A. Removing Emergency Bleed-air Shutoff Valve

(1) Remove access door on lower side of right hand pylon.

(2) Disconnect electrical connector to valve.

(3) Remove Marman clamps connecting valve to lines.

(4) Remove valve from bay.

(5) Cap lines and valve openings.

B. Installing Emergency Bleed-air Shutoff Valve

(1) Remove caps from valve and ducts.

(2) Place gaskets (Conoseal 51134-100A) in flanges of valve.

(3) Place valve in position between ducts. The arrow on the body must point to the fuselage.

(4) Install clamps (Marman 51394-H100, using Conoseal 51134-1000 gaskets). Torque bolts to 35-40 inch-pounds.

(5) Connect electrical connector to valve; then safety with lockwire.

(6) Check electrical operation of valve by feeling valve for solenoid operation.

(7) Install access.

4. Replacement of Wheel Seal*, Door Seal, and Sensing Lines Dehydrator

A. The dehydrator is located in the aft fuselage equipment compartment on the right side. The dehydrator contains a silica gel dessicant which must be replaced when moisture saturated (when it changes color from blue to pink). To replace dehydrator, proceed as follows:

(1) Open aft fuselage access door.

(2) Disconnect inlet and outlet lines from dehydrator capsule.

(3) Remove bolts securing capsule to shelf bracket and remove capsule.

(4) Remove line fittings from capsule and install in replacement capsule

*Aircraft NA-370-1 through -8 not changed by Service Bulletin 72-8

NOTE: The capsule may be installed with either end facing forward.

(5) Position capsule on shelf bracket, and install and tighten attaching bolts.

(6) Connect and tighten inlet and outlet lines to capsule.

(7) Perform check-out of pressurization system.

(8) Close access doors.

5. Servicing Wheel Seal*, Door Seal, and Sensing Lines Dehydrator

A. The following materials are required for servicing the dehydrator: two gaskets, Part No. RB8089-1; and silica gel, Specification MIL-D-3716, Grade H, Type IV. The RB8089-1 gaskets may be purchased from Lear Siegler, Inc, Romec Division, Fort Worth, Texas. Service dehydrator capsule as follows:

(1) Remove dehydrator capsule from airplane.

WARNING: CAPSULE END PLATES ARE SPRING LOADED.

(2) Remove nuts and washers from the bolts and remove capsule end plates.

(3) Remove compression spring, spring plate, felt pads, filter pad, and sieve; retain for reinstallation.

(4) Remove dessicant.

(5) Remove and replace RB8089-1 gaskets. (See figure 21-10-402.)

(6) Place end cap with new gasket on one end of capsule.

(7) Install filter pad and sieve, respectively; then install tie bolts with washers.

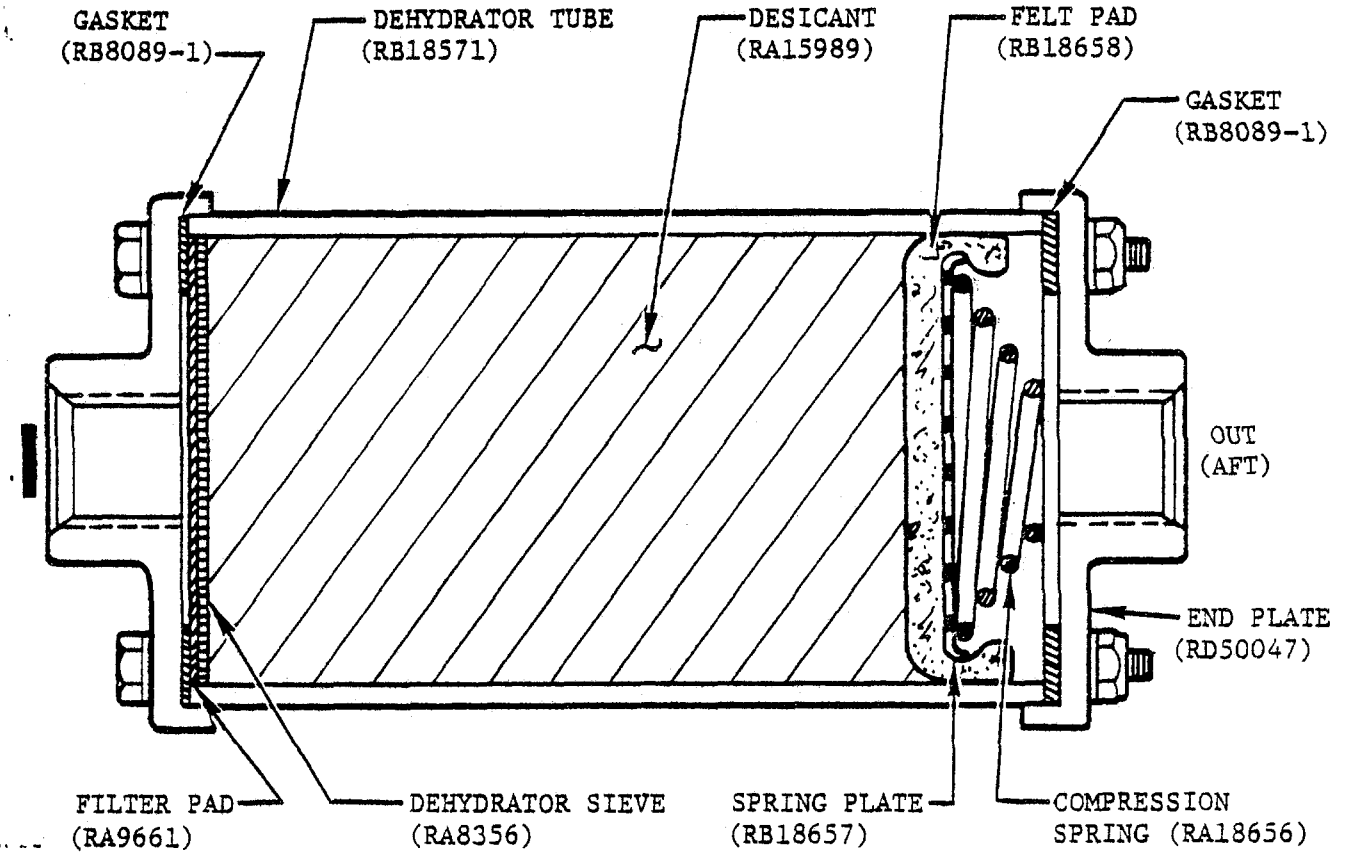
(8) Fill capsule with desiccant (about 13 ounces) so that compression spring must be compressed when installing end cap.

(9) Install felt pad, spring plate and compression spring, respectively.

(10) Install end cap with gasket over tie bolts and capsule, and secure with washers and nuts. Torque nuts to 12-20 inch-pounds.

(11) Install dehydrator in airplane.

*Aircraft NA-370-1 through -8 not changed by Service Bulletin 72-8



Dehydrator, Door Seal and Sensing Lines
Figure 21-10-402

DISTRIBUTION - DESCRIPTION AND OPERATION

1. Air Distribution System - Aircraft NA-370-2 Through -6, -8 and -9

Bleed air from one or both engines passes through the flow limiter valve, then divides into two duct branches. One branch routes the hot air to a hot-air mixing valve; the other branch routes air to the refrigeration unit for cooling. (See figure 21-00-1.)

The hot-air branch from the flow limiter valve passes through the hot-air mixing valve, then rejoins the cooled air from the refrigeration unit. The hot-air mixing valve operates automatically to admit the proper amount of hot air into the cool air supply line in response to the cabin air temperature selector in the cockpit center pedestal and a cabin air temperature sensor in the cabin.

Cooled air from the refrigeration assembly is routed into a "Y" duct. This divides the airflow into two branches. One branch joins the main system hot-air line downstream of the hot-air mixing valve. The other branch routes the air directly into the cabin and cockpit through a cabin cooling-air supply check valve. This cabin cooling air is then ducted to two cooling-air eyeball outlets in the crew compartment. These outlets are operated individually at the discretion of the crew.

The mixed, thermostatically controlled air, which has been limited to the desired airflow by the flow-limiting control and mixed to the proper temperature by the action of the hot-air mixing valve, enters the cabin and cockpit through a cabin cooling-air supply check valve. The air is routed through ducting, entering the cabin at low velocity through flush outlets at about head level, with additional air outlets at floor level locations.

The right-hand section of the conditioning air supply duct continues through the cabin into the crew compartment. The crew compartment has two adjustable eyeball air outlets located on the compartment aft bulkhead, one behind and to the left of the pilot's seat, the other behind and to the right of the copilot's seat. Foot outlets are provided in the left and right side consoles and are sized for maximum effect without adjustment.

1A. Air Distribution System - Aircraft NA-370-1, -7, NA-380-1 and Later

Bleed air from one or both engines passes through the flow limiter valve, then divides into two branches. One branch routes air to the cabin hot-air mixing valve. The other branch routes air to the cockpit hot-air mixing valve through a smaller branch duct, but the major portion is supplied to the refrigeration unit for cooling. Duct size properly proportions the distribution of hot air from the flow limiter to the mixing valves and refrigeration unit. (See figure 21-00-1A.)

The cooled air output from the refrigeration unit is also divided into two branches with the major portion routed to the cabin and the remainder to the cockpit, again properly proportioned by duct size. Prior to entering the cabin and cockpit the cooled air ducts are rejoined by the hot air ducts from the hot air mixing valves. At these junctions the cooled and hot air are mixed. Duct air temperature sensors, cabin and cockpit air temperature sensors and the cockpit and cabin air temperature selectors work in conjunction with the cockpit and cabin temperature control boxes (see figure 21-00-2A) to regulate the opening of the hot-air mixing valves. The valves are opened only enough to add the hot air required to raise the temperature of the cooled air to the temperature setting of the air temperature selectors.

Check-valves in the separate air ducts are provided to prevent reverse airflows and in-line mufflers reduce airflow noises. Outlets are branched off the tempered air ducts throughout the cabin and in the cockpit to properly distribute the conditioned air.

Two air supply overheat switches are provided, one for the cabin air supply and one for the cockpit. They are installed in the separate air supply ducts downstream of the reverse airflow check valves. The switches are set to close at 265°F. If either switch closes, the corresponding "CABIN AIR HOT" or "COCKPIT AIR HOT" warning light will come on to warn the pilot of the overheat condition.

The cabin flood valve is provided to permit air which has been partially cooled in the heat exchanger to bypass the expansion turbine and water separator and enter the cabin and cockpit air supply lines downstream of the hot-air mixing valves. The valve is normally open being closed when bleed air pressure is applied, and operates pneumatically in response to a decrease in bleed-air pressure, bypassing the two high-pressure-drop items (expansion turbine and water separator). The valve automatically provides an adequate amount of air to the cockpit and cabin even though the supply bleed air pressure is only slightly higher than cabin pressure. This condition could occur during a rapid descent from high altitude. With decreasing bleed-air pressure, this valve will start to open at 15 psi and will be fully open at 12 psi. When the airplane is below 14,300' or on the ground, a barometrically controlled altitude pressure switch closes an electrical circuit to the valve, energizing it. When energized, the valve will not start to open until the bleed air pressure decreases to 8 psi, being fully open at 3.5 psi. The change prevents bumps in cabin pressure during approach power setting changes.

This pressure switch also keeps the cabin flood flow valve in the closed position until the airplane reaches a 14,300-foot altitude. At this altitube the switch opens, allowing the valve to operate normally on bleed air pressure.

DESCRIPTION AND OPERATION (Continued)

2. FLOW LIMITER VALVE

The pneumatically operated flow-limiting control is installed in the bleed-air supply ducting, automatically maintaining the proper bleed-air flow rate to the air conditioning and pressurizing systems at all airplane altitudes. The flow varies with the temperature and pressure present at the valve inlet. The flow-limiting control valve is in the rear fuselage section.

3. CABIN FLOOD FLOW VALVE

The cabin flow valve is installed in the engine compressor bleed-air ducting and is provided to permit air which has been partially cooled in the heat exchanger to bypass the expansion turbine and water separator and enter the cabin and crew compartment main air supply line downstream of the hot-air mixing valve. The cabin flood flow valve operates pneumatically in response to a decrease in bleed-air pressure, bypassing the two high-pressure-drop items (expansion turbine and water separator). The valve automatically provides an adequate amount of air to the cabin even though the supply bleed-air pressure is only slightly higher than cabin pressure. (This condition could occur during a rapid descent from high altitude.) With decreasing bleed-air pressure, this valve will start to open at 15 psi and will be fully open at 12 psi. When both engines are switched off, the cabin flood flow valve pressure setting of 12 to 15 psi is changed by an internal solenoid action to 3.5-8 psi.

An altitude pressure switch controls the electrical circuit to the cabin flood flow valve. This barometrically controlled pressure switch keeps the cabin flood flow valve electrically energized and in the closed position until the airplane reaches a 14,300 foot altitude. At this altitude the switch opens, allowing the valve to operate normally on bleed-air pressure. When the altitude pressure switch is not actuated (below 14,300-foot altitude), the cabin flood flow valve pressure setting is 3.5 to 8 psi. Valve is fully closed when the sense pressure line pressure is 15 psi or higher (solenoid de-energized). Solenoid energized, pressure sense line pressure of 3.5 or above will allow the valve to close. The valve allows bleed air to enter the cabin (valve modulates open) when the supply pressure is lower than normal. With normal pressure, the valve will always be closed.

When the airplane is on the ground, the solenoid of the flood flow valve is energized through the altitude pressure switch. This will allow the flood flow valve to close when 5 psi air pressure is applied during APU operation.

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DISTRIBUTION - TROUBLE SHOOTING

1. Trouble Shooting Air Distribution System

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. NO HEATED OR COOLED AIRFLOW TO CABIN.		
(1) Faulty engine bleed-air shutoff valve or valves. Faulty valve electrical operating circuit, or loose pneumatic control line fittings.	If engine compressor bleed air can not be obtained first, check that engine fire emergency shutdown controls have not been actuated; then see figure 21-00-2 or 21-00-2A and check for electrical power at both engine bleed-air shutoff valve disconnect plugs. Also, check pneumatic control line fittings for tightness.	Remove engine bleed-air shutoff valve, and check 0.026-inch bleed hole in nose cone. If it is clogged, remove obstructions with a smaller wire than the hole size. If electrical power is at valve disconnect plugs and if pneumatic control line fittings are tight, remove and replace engine bleed-air shutoff valves.
B. EXCESSIVE CABIN PRESSURE FLUCTUATION DURING RAPID DESCENT FROM ALTITUDE AND NOT ENOUGH FRESH AIR ENTERING CABIN AFTER LANDING AIRPLANE.		
(1) Faulty cabin flood flow valve, loose pneumatic control lines, faulty electrical control circuit, or faulty flow-limiting control.	Check cabin flood flow valve position. Valve should be open on ground. If valve is found in closed position, see figure 21-00-2 or 21-00-2A and check for proper operation of valve control circuit. If electrical power is found at valve disconnect plug, the valve may be checked for electrical operation. However, it does not prove valve pneumatic operation.	Make necessary checks to determine proper valve operation, check electrical and pneumatic control circuit and lines, and replace any faulty units.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
	Therefore, it may be necessary to perform a pneumatic check for proper valve operation. For valve pneumatic check, refer to air distribution check-out procedure.	
(2) Faulty operation of relay, indicated by erratic operation of regulator.	Check for fluctuating flow of exhaust air and loose connections on control line.	Replace relay or tighten loose connections.
C. FOG EMITTING FROM AIR DISTRIBUTION SYSTEM IN AFT CABIN.		
(1) Clogged water drain line and/or fitting.*	Remove fitting (with screen) and line, and check for obstruction.	Clean screen and blow out line.
(2) Contaminated water separator.*	Check water separator drain, on bottom of fuselage just aft of station 357. If fuselage skin is streaked with dirt or foreign matter in the area of the drain discharge, the water separator condenser is contaminated.	Clean or replace water separator condenser.

*Aircraft NA-370-2 through -6, -8 and -9

DISTRIBUTION

MAINTENANCE PRACTICES

REPLACEMENT OF CABIN FLOOD FLOW VALVE

Removing Cabin Flood Flow Valve

1. Open aft fuselage access door.
2. Disconnect sensing line and electrical wires from flow valve. Loosen clamps at flow valve inlet and outlet duct joints, and slide clamps back on ducting.
3. Remove flow valve from ducting and, to prevent entry of foreign material, cover flow valve, sensing line, and ducting openings.

Installing Cabin Flood Flow Valve

1. Remove protective seals from valve, pneumatic pressure line, and ducts.
2. Insert gaskets in flanges of valve; then place valve in position between ducts, checking that pressure line and pneumatic pressure fitting align.
3. Install clamps (Marman clamp, 51394H-200, using Conoseal-Marman gasket, 51134-200A). Torque clamps with steel flange to 55-60 inch-pounds; with aluminum flange, to 40-50 inch-pounds.
4. Connect pneumatic pressure line to outboard side of valve. Torque pneumatic pressure fitting to 40-65 inch-pounds.
5. Connect electrical wires to cabin flood flow valve.
6. Close accesses.

REPLACEMENT OF FLOW-LIMITING CONTROL

Removing Flow-Limiting Control

1. Open aft fuselage access doors.
2. Remove clamp insulation covers.
3. Disconnect clamps at both ends of control and slide clamps back on ducting. Note position of clamps to ensure proper positioning during installation.
4. Remove flow-limiting control from airplane.
5. Cover openings in duct to prevent foreign object entry.

Installing Flow-Limiting Control

1. Remove protective coverings from serviceable control and ducting.
2. Place gaskets in valve connecting flanges.

CAUTION

Ensure that flow-limiting control is properly positioned in ducting.

3. Check that mounting flanges are clean, then position limit control between duct openings.
4. Place clamps on flanges with connecting bolts positioned to clear insulation covers.
5. Torque clamp bolts to 60-70 inch pounds.
6. Place clamps on flanges with connecting bolts positioned to prevent entry of fluids inside of cover.
7. Install clamp insulation covers.
8. Operate engines and check for excessive air leaks.
9. Close aft fuselage access doors.

AIR DISTRIBUTION SYSTEM CHECK-OUT PROCEDURE

Checking for Proper Operation of Control Valves, Check Valves, and for Airflow — (S/N 370-2 thru -6, -8 and -9)

CAUTION

The cockpit is to be occupied by qualified (medically approved) personnel. All openings must be closed, including main door. Check that DOOR OPEN caution light is off.

1. Push in the following circuit breakers:
   - ENG BLEED AIR - LH
   - ENG BLEED AIR - RH
   - CABIN TEMP CONTROL
   - EMER RAM AIR
   - CABIN - PRESS DUMP & PRESS DUCT FAIL
   - CABIN - AIR OVHT IND
   - DR SEAL & LOCK IND
   - EQUIP GRD BLO - SECOND CONT
   - EQUIP GRD BLO - POWER
   - NO. 1 INV CONT
   - NO. 1 INV PWR

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MAINTENANCE PRACTICES (Continued)

Checking for Proper Operation of Control Valves, Check Valves, and for Airflow — (S/N 370-2 thru -6, -8 and -9) (Cont'd)

2. Set environmental and pressurizing controls as follows:
   - CABIN TEMP switch at AUTO
   - AUTO CONT rheostat at COLD
   - CABIN AIR switch at OFF
3. Connect external DC electrical power to airplane and position INVERTER switch at NO. 1. Start both engines and operate them at 80% to 85% rpm or higher. Check that no airflow is obtained at the cabin air outlets.

NOTE

No airflow indicates that the engine bleed-air shutoff valves are closed.

4. With both engines operating at 80% to 85% rpm or higher, position CABIN AIR switch in turn to RH ENG, OFF, LH ENG, OFF, and BOTH ENG. Check that cool air emits from cabin air outlets when switch is placed at RH ENG, LH ENG, and BOTH ENG, and that there is no airflow when switch is placed at OFF.

NOTE

This check indicates that the engine bleed-air shutoff valves open and close properly and that the refrigeration unit operates properly.

5. Pull cotter pins and disconnect landing gear load switches, on right and left oleo torque links, position to flight configuration, engage LDG GEAR-POS circuit breaker, and position CABIN AIR switch at EMER PRESS. Check that air ceases to flow from cabin and cockpit outlets and enters through a single outlet in aft end of cabin. The air will be warm-to-hot. Reconnect landing gear load switches, install cotter pins, and pull out LDG GEAR-POS circuit breaker.
6. With CABIN AIR switch at BOTH ENG, rotate AUTO CONT rheostat between COLD and HOT, and check that cabin supply temperature changes accordingly. Select a comfortable cabin temperature.

NOTE

Changes in temperature indicate that the hot-air mixing valve and cabin air temperature control system operate properly.

7. Position CABIN AIR switch at RAM and then at RAM & DUMP. Check that both engine bleed-air shutoff valves close at either switch position. Return switch to BOTH ENG and check that valves open.

NOTE

No airflow from the cabin air outlets indicates that both engine bleed-air shutoff valves close.

8. Move CABIN TEMP switch to MAN HOT; hold at MAN HOT for about 30 seconds and check that temperature of cabin supply air increases.

NOTE

Increase in cabin air temperature indicates that the hot-air mixing valve opens when the switch is moved to MAN HOT.

9. Move CABIN TEMP switch to MAN COLD; hold it at MAN COLD for about 30 seconds and check that temperature of cabin supply air decreases.

NOTE

Decrease in cabin air temperature indicates that the hot-air mixing valve closes when the switch is moved to MAN COLD.

10. Return CABIN TEMP switch to OFF. Shut down engines.
11. Remove DC power from airplane bus system, turn off AC power, disengage circuit breakers used for this check, and position control switches at OFF or normally maintained positions.

Checking for Proper Operation of Control Valves, Check Valves, and for Airflow (S/N 370-1 and -7, 380-1 and Subs)

CAUTION

The cockpit is to be occupied by qualified (medically approved) personnel. All openings must be closed, including main door. Check that DOOR OPEN caution light is Off.

1. Push in the following circuit breakers:
   - ENG BLEED AIR - LH
   - ENG BLEED AIR - RH
   - CABIN TEMP CONTROL
   - EMER RAM AIR
   - CABIN - PRESS DUMP & PRESS DUCT FAIL
   - CABIN - AIR OVHT IND

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MAINTENANCE PRACTICES (Continued)

Checking for Proper Operation of Control Valves, Check Valves, and for Airflow — (S/N 370-1 and -7, 380-1 and Subs) (Cont'd)

- AFT FUS OVHT IND
- DR SEAL & LOCK IND
- FIRE DET TEST
- EQUIP GRD BLO — SECOND CONT
- EQUIP GRD BLO — POWER
- NO. 1 INV CONT
- NO. 1 INV PWR
- REFRIG UNIT

2. Set environmental and pressurizing controls as follows:
   - CABIN TEMP switch at AUTO
   - COCKPIT TEMP switch at AUTO
   - Cabin AUTO CONT rheostat at COLD
   - Cockpit AUTO CONT rheostat at COLD
   - CABIN AIR switch at OFF
3. Connect external DC electrical power to airplane and position INVERTER switch at NO. 1. Start both engines and operate them at 80% to 85% rpm or higher. Check that no airflow is obtained at the cabin and cockpit air outlets. Check the position indicator of the cabin flood flow valve. It should be closed.

NOTE

No airflow indicates that the engine bleed-air shutoff valves are closed.

4. Check that the forward fuselage equipment ground cooling blower is operating.
5. With both engines operating at 80% to 85% rpm or higher, position CABIN AIR switch in turn to RH ENG, OFF, LH ENG, OFF, and BOTH ENG. Check that cool air emits from cabin and cockpit air outlets when switch is placed at RH ENG, LH ENG, and BOTH ENG, and that there is no airflow when switch is placed at OFF.

NOTE

This check indicates that the engine bleed-air shutoff valves open and close properly and that the refrigeration unit operates properly.

6. With CABIN AIR switch at BOTH ENG, rotate CABIN TEMP — AUTO CONT and COCKPIT TEMP AUTO CONT rheostat between COLD and HOT, and check that cabin and cockpit supply temperature changes accordingly. Select a comfortable cabin and cockpit temperature.

NOTE

Changes in temperature indicate that the hot-air mixing valves and the air temperature control systems operate properly.

7. Position CABIN AIR switch at RAM and then at RAM & DUMP. Check that both engine bleed-air shutoff valves close at either switch position. Return switch to BOTH ENG and check that valves open.

NOTE

No airflow from the cabin and cockpit air outlets indicates that both engine bleed-air shutoff valves close.

8. Position CABIN AIR switch to EMER PRESS. Check that both engine bleed-air shutoff valves close and the emergency bleed-air shutoff valve opens.

NOTE

Airflow from the emergency air pressure outlet in cabin indicates that emergency bleed air shutoff valve opens.

9. Position CABIN AIR switch to BOTH ENG and check that the engine bleed-air shutoff valves open and the emergency bleed-air shutoff valve closes.
10. Move CABIN TEMP and COCKPIT TEMP switch to MAN HOT; hold at MAN HOT for about 30 seconds and check that temperature of cabin and cockpit supply air increases.

NOTE

Increase in cabin and cockpit air temperature indicates that the hot-air mixing valves open when the switches are moved to MAN HOT.

11. Move CABIN TEMP and COCKPIT TEMP switches to MAN COLD; hold at MAN COLD for about 30 seconds and check that temperature of cabin and cockpit supply air decreases.

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MAINTENANCE PRACTICES (Continued)

Check for Proper Operation of Control Valves, Check Valves, and for Airflow — (S/N 370-1 and -7, 380-1 and Subs) (Cont'd)

NOTE

Decrease in cabin and cockpit air temperature indicates that the hot-air mixing valves close when the switches are moved to MAN COLD.

12. Select a comfortable cabin and cockpit temperature. Position engine power levers at IDLE and position the ANTI-ICE-ENGINE INLET switches to ON.
13. If the RH or LH ENG DE-ICE FAIL caution lights illuminate, advance engine power levers until both caution lights extinguish. Both caution lights should go out before 65% engine rpm is reached.
14. Position ANTI-ICE-ENGINE INLET switches to OFF. Check that both caution lights remain out. Position CABIN TEMP and COCKPIT TEMP switches to OFF.
15. Shut down engines and remove DC power from airplane bus system, turn off AC power, disengage circuit breakers used for this check, and position control switches at OFF or normally maintained positions.

FUNCTIONAL CHECK OF CABIN FLOOD FLOW VALVE

To perform a functional check of the cabin flood flow valve, proceed as follows:

1. Remove cabin flood flow valve from airplane.
2. Apply a 5 psig regulated source of dry air to supply (pressure) port. Check that valve is open.
3. Using 24 VDC electrical power, energize solenoid of valve. Check that valve closes.
4. De-energize valve. Check that valve opens.
5. Increase supply pressure to 12 psig. Check that valve is open.
6. Energized solenoid of valve. Check that valve closes.
7. De-energize valve. Check that valve is open.
8. Increase supply pressure to 15 psig. Check that valve starts to close from 12 to 15 psig and is fully closed at 15 psig.
9. Decrease supply pressure to zero. Check that valve opens.
10. Remove air and electrical source from valve. Replace valve if above checks are not met.
11. Install serviceable valve in airplane.
12. Leak check valve during engine run.

NEW

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PRESSURIZATION CONTROL - DESCRIPTION AND OPERATION

1. Cabin Pressurization Control System

The pressure in the cabin and crew compartment is controlled by the rate-of-change control and maintained by the cabin pressure regulator. (See figure 21-00-1 or 21-00-1A.) Pressurization and rate of change may be selected by setting the rate-of-change controls. A cabin altitude of sea level to 8000 feet, and the desired rate of pressurization to the selected altitude, may be chosen up to an 8000-foot altitude. As the airplane climbs, the cabin altitude will also climb at the rate selected, about 50 to 2000 feet per minute until a differential pressure of 8.20 (±0.1) psi and/or an altitude of 40,000 feet is reached. A sea level altitude can be maintained up to an airplane altitude of about 21,200 feet.

The cabin outflow valve is regulated to a 15,000-foot isobaric schedule by the limit control, in case of failure in the open position at altitudes in excess of 13,500 (±1500) feet. The pneumatic relay, in the line between the controller and the outflow valve, increases the sensitivity to altitude and/or pressure changes, thereby reducing cabin fluctuations. The vacuum pump supplies the negative pressure required to operate the outflow valve at low altitude.

The cabin air dump and safety valve, commonly called a dump valve, serves as a combination pressure relief, vacuum relief, and pressure dump valve. It is on the forward cockpit pressure bulkhead and discharges into the electronic equipment compartment. If the cabin air pressure regulator or cabin air isobaric control should fail, this valve would prevent cabin pressure from exceeding 9.1 (±0.1) psi above outside atmospheric pressure.

The vacuum relief feature of the cabin air dump and safety valve prevents the atmospheric pressure from exceeding cabin pressure during rapid descent from altitude.

The emergency dump feature is operated by the cabin air dump solenoid control valve. If it is desired to dump the pressure in the cabin, the "CABIN AIR" control is positioned at RAM & DUMP. The engine bleed-air shutoff valves will close, and the emergency ram-air supply valve will open to admit ram air into the cabin simultaneously with the opening of the cabin air dump and safety valve.

Provisions are made in the air conditioning and pressurizing system ducting to permit ground pressurizing and conditioning by external means. The ground fittings are accessible through fuselage rear section access doors. A cabin pressure dump switch is provided, on the ground pressure test panel, to check cabin air dump valve operation.

2. Rate-of-Change Control

The rate-of-change control consists of a housing assembly enclosing three separate pressure control systems: selective isobaric, selective pressure rate of change, and fixed differential. The housing assembly is provided with a front panel containing rate control and altitude selector knobs, a dial with an isobaric pointer, and a window for indicating airplane altitude at maximum pressure differential. All system connections and a rate filter are at the rear of the housing assembly. An electrical connector for dial lights is mounted on the side of the housing assembly.

The selective isobaric system contains an isobaric bellows with rocker arm, fulcrum, and isobaric calibration spring, positioned on the rate-of-change reference chamber side of the rate control diaphragm. An isobaric metering valve contacts the diaphragm on the reference chamber side.

The selective pressure rate-of-change system consists of a rate control diaphragm containing two poppet valves, a capillary tube, the rate filter, and a rate metering valve. The diaphragm divides the housing assembly into two chambers, a reference chamber and a rate-of-change reference chamber. The capillary tube interconnects both of these chambers.

The fixed differential system is positioned on the reference chamber side of the unit and consists of a differential diaphragm and metering valve. This system also incorporates a fail-safe feature consisting of an emergency differential control seat and orifice.

The cabin altitude is controlled by turning the altitude selector knob until the isobaric pointer indicates the desired altitude on the dial. The maximum airplane altitude for the selected cabin altitude is shown in the window cutout on the dial face. The rate at which the cabin altitude increases to the selected altitude is then set by adjusting the rate control knob.

Before the start of control operation, pressure on both sides of the rate control diaphragm is equalized at approximately atmospheric pressure. The isobaric bellows is compressed, and the isobaric metering valve is open. Cabin air flows through the orifice into the reference chamber and out through the isobaric metering valve. The reference chamber pressure is sensed by the outflow valve which operates according to the control schedule of the outflow valve control. During this phase of operation, the airplane cabin will remain essentially unpressurized, provided that the rate of airplane ascent does not exceed the selected rate.

When the airplane rate of climb exceeds the selected rate, the pressure in the reference chamber is reduced, causing a corresponding decrease in the rate-of-change reference chamber. The rate of decrease of pressure in the rate-of-change reference chamber is limited to a fixed value as determined by the position of the rate metering valve, which is adjusted by the rate control knob. Pressure change in the rate-of-change reference chamber lags behind that of the reference chamber, resulting in a pressure differential across the rate control diaphragm. In response to this pressure differential, the rate control diaphragm moves the isobaric metering valve to limit the

flow of air from the reference chamber until the pressure in both chambers is substantially equal. The cabin outflow valve acts to control the rate of change of cabin pressure accordingly.

As the pressure in the reference chamber continues to decrease and as the selected cabin altitude is approached, the isobaric bellows expands to override the rate control diaphragm and operate the isobaric metering valve. Action of the isobaric bellows during this phase of operation keeps the reference pressure at an essentially constant value. The outflow valve then reacts to variations in cabin pressure to maintain the selected cabin altitude.

As the airplane continues to ascend and reaches an altitude at which the pressure differential between cabin and atmosphere approaches the calibrated setting, the pressure differential across the differential diaphragm opens the metering valve and releases air in the reference chamber to atmosphere. This results in a decrease in reference chamber pressure and causes cabin pressure to open the outflow valve. The unit incorporates a fail-safe feature which prevents loss of reference pressure in case of a ruptured differential diaphragm by sealing off the reference chamber. With the diaphragm assembly seated on the emergency differential control seat, the reference pressure loss is through the orifice only. During a rapid descent, the differential metering valve closes because of increased atmospheric pressure. If the resultant rate of increase in reference pressure exceeds the selected rate, the rate control diaphragm opens the isobaric metering valve and air from the reference chamber is discharged to atmosphere. The outflow valve reacts to the reference pressure to limit the rate of increase in cabin pressures.

3. Isobaric Control and Ground Test Valve

When the airplane is on the ground and from ground up to 8000 feet and with an 8000 feet cabin selected, altitude air flows from the cabin through the cabin pressure regulator outflow gate; therefore, no pressure is built up in the pressure compartments. This air flows to the reference pressure chamber of the absolute pressure control and manual shutoff valve. During the unpressurized range of operation, reference pressure in the absolute pressure control and manual shutoff valve is enough to hold the isobaric metering valve open by preventing the isobaric control bellows from expanding, thus allowing reference pressure air to flow to atmosphere. At this time, reference pressure is about the same as atmospheric pressure. Reference pressure is sensed by the cabin pressure regulator outflow gate through a sensing line connected to the pneumatic relay and the isobaric control and ground test valve. Cabin pressure is slightly higher than atmospheric pressure during this range of operation even though it is considered to be the unpressurized range.

When the rate-of-change control is used to select a desired altitude between sea level and a 21,200-foot altitude, the rate-of-change control programs to the isobaric control the desired pressure ratio differential, causing the cabin air pressure regulator to control the cabin air accordingly.

When the airplane approaches the isobaric range (40,000 to 45,000 feet), pressure in the isobaric control reference pressure chamber decreases to allow

the isobaric control bellows to expand. This force, aided by the isobaric calibration spring, moves the isobaric metering valve toward its seat. The movement of the isobaric metering valve toward its seat limits flow of reference pressure air to atmosphere. This controls reference pressure at a substantially constant pressure throughout the isobaric range of operation. During this range of operation, variations in cabin pressure cause compensating movements of the system isobaric control (outflow valve control), pneumatic relay, and cabin air pressure regulator to maintain pressure within the compartments at a constant value.

NOTE: The absolute pressure control and manual shutoff valve must be safetied at flight position for proper in-flight operation.

When the airplane reaches altitude where the pressure differential between inside and outside atmosphere approaches an isobaric pressure differential control setting of 8.2 (±0.1) psi of the isobaric control and ground test valve, the pressure differential between reference pressure and true static atmosphere is enough to cause the differential control diaphragm to overcome force of the differential calibration spring. This moves the differential metering valve off its seat, allowing reference pressure air to escape to atmosphere. The resulting decrease in reference pressure is enough to further expand the isobaric control bellows and seat the isobaric metering valve. With the isobaric control system thus made inoperative, cabin-to-atmosphere pressure differential is governed by the differential control system and is maintained at essentially a constant value. This is done by metering reference pressure air through the differential metering valve to the pneumatic relay, resulting in a constant reference-to-atmosphere and cabin-to-atmosphere pressure differential while the airplane remains in the differential range.

4. Cabin Pressure Regulator (Air Outflow Valve)

When the airplane pressure system is being used, cabin air bleeds into the reference chamber of the cabin air pressure regulator, through the air filter. Cabin air also flows through passages in the poppet valve into the regulator cabin pressure chamber. The pressure differential between the reference chamber and the cabin pressure chamber controls the opening and closing of the regulator air outflow gate during unpressurized or pressure relief range from ground up to 8000 feet altitude.

During this time, with any increase in cabin pressure in the cabin pressure chamber, cabin-to-atmosphere pressure differential across the differential control diaphragm also increases. When this pressure differential reaches the calibrated pressure relief setting, the pressure relief valve unseats, allowing reference chamber air to flow to atmosphere, in turn decreasing reference pressure. The pressure differential across the regulator outflow valve cover plate and outflow valve diaphragm increases, opening the regulator poppet outflow valve. The outflow valve modulates open and closed to control outflow of air from the cabin, releasing pressure and preventing build-up in the pressure compartments. When the airplane is at 8000 feet and above, cabin air bleeds through the regulator air filter and orificed into the regulator reference chamber, into the pneumatic relay, and on through the isobaric control and ground test valve to atmosphere. The cabin

DESCRIPTION AND OPERATION (Continued)

4. CABIN PRESSURE REGULATOR (AIR OUTFLOW VALVE) (Cont'd)

air passing through the orificed fitting reduces the pressure, creating a reference pressure in the reference chamber. To maintain reference pressure at an approximately constant value, the cabin air isobaric control limits the outflow of reference pressure air. The pneumatic relay amplifies the reference pressure. Any change in cabin pressure changes the cabin-to-reference pressure differential to modulate the cabin air pressure regulator poppet outflow valve open and closed, limiting and regulating outflow of cabin pressure.

5. PNEUMATIC RELAY

In normal operation, three pressures are applied to the pneumatic relay: a control reference pressure from the cabin air pressure isobaric and differential control unit, atmospheric pressure, and reference pressure of the cabin air pressure regulator.

Air at cabin pressure enters the isobaric control through an orifice, creating a reference pressure at a value slightly less than cabin pressure. This reference pressure is sensed in the relay through the control connection on the relay. Reference pressure in the regulator is sensed in the relay through the outflow valve (regulator) connection on the relay. This reference pressure is the same as cabin pressure until a specific pressure differential exists across the relay diaphragm assembly. As cabin pressure increases, reference pressure in the relay control pressure chamber is less than cabin pressure within the relay outflow valve (regulator) reference pressure chamber. When this pressure differential (regulator reference pressure to isobaric control reference pressure) increases sufficiently to overcome the force of the relay calibration spring, the relay diaphragm moves toward the calibration spring and allows the metering valve to move toward a metering position. This metering action of the metering valve regulates the flow of the regulator reference chamber air to atmosphere to establish the regulator pressure for operation of the regulator. When atmospheric pressure exceeds cabin pressure, the check valve in the relay is positioned against the valve seat and prevents the reverse flow of air from the atmopsphere through the relay regulator reference chamber to the regulator.

6. VACUUM PUMP

An electrically driven vacuum pump and a vacuum regulator provide a reliable vacuum source or 5.5 to 6.0 in. Hg. This constant vacuum supplies the negative pressure required to operate the outflow valve on the ground and at altitudes below 14,300 feet. On the ground and with one or two engines operating, the vacuum pump enables the outflow valve to be open, thus preventing a cabin pressure bump when the entrance door is closed and air inflow into the cabin is initiated. The vacuum pump does not operate if both egines are switched to OFF or if airplane altitude is above 14,300-feet. The vacuum pump, on airplanes S/N 380-57 and subs, and airplanes changed by S/L 36, incorporates an internal circuit breaker with a reset push button.

7. CABIN AIR DUMP AND SAFETY VALVE

The cabin air dump and safety valve is identical to the cabin air pressure regulator in operation but is set to relieve at a higher pressure. The safety valve relieves at 9.1 (±0.1) psi at any altitude. However, the valve can be dumped at any time it is necessary whenever the CABIN AIR switch is positioned at RAM & DUMP.

During normal operation of the cabin air dump and safety valve, the pressure differential acting on the differential control diaphragm is not enough to overcome force exerted by the pressure relief calibration spring. When cabin-to-atmosphere pressure differential reaches or exceeds the calibrated value, cabin pressure in the cabin pressure chamber is enough to force the differential control diaphragm in a direction that opens the ball pressure relief valve. When this occurs, air escapes from the reference chamber to atmosphere through the outflow valve guide and the atmosphere sensing line. This establishes a cabin-to-reference pressure differential across the outflow valve cover plate and actuator section of the outflow diaphragm, opening the poppet outflow valve and allowing cabin air to flow to atmosphere. Once a reference pressure has been established, the poppet outflow valve is held in the approximate position that maintains differential pressure at the calibrated value.

When atmospheric pressure exceeds cabin pressure, pressure on the vacuum relief and balance section of the outflow valve diaphragm overcomes cabin pressure on the opposite side of the diaphragm. The outflow diaphragm is forced against the outflow valve cover plate and overcomes the pressure in the reference chamber and the force of the poppet valve return spring. The poppet outflow valve moves toward the reference chamber, and enough air to restore the calibrated ratio of cabin pressure to atmospheric pressure flows through the valve and into the cabin. If the pneumatic control line to the cabin air dump and safety valve is open during vacuum relief operation, a check valve prevents air to atmospheric

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DESCRIPTION AND OPERATION (Continued)

7. CABIN AIR DUMP AND SAFETY VALVE (Cont'd)

pressure from entering the reference chamber and closing the poppet outflow valve.

The cabin air dump and safety valve dump operation is done by electrically or manually opening the pneumatic control line. This allows air from the reference chamber to rapidly escape to atmosphere. The cabin-to-reference pressure differential across the outflow valve cover plate then causes the outflow poppet valve to fully open, allowing cabin air to flow (dump) to atmosphere.

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PRESSURIZATION CONTROL - TROUBLE SHOOTING

1. Trouble Shooting Pressurization Control System

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. REGULATOR OR CABIN AIR DUMP VALVE FAILS TO RELEASE AIR.		
(1) Pneumatic line to controller loose or defective.	Feel for loose or damaged lines and "B" nuts.	Tighten nuts or replace damaged lines.
(2) Associated cabin pressure control equipment does not function properly.	Replace suspected item and check operation.	Replace defective item.
(3) Ground-test valve at ground-test position.		Position valve to flight position and safety with lockwire.
B. DIFFERENTIAL-PRESSURE AND ISOBARIC CONTROL FAILS TO CONTROL CABIN PRESSURE.		
(1) True static atmosphere line to pneumatic relay loose or obstructed.	Check for restrictions or loose lines.	Tighten, remove restriction, or replace lines.
(2) Pneumatic connections leaking.	Check for loose connections.	Tighten connections.
C. LOSS OF COMPARTMENT PRESSURIZATION WHILE IN FLIGHT.		
(1) Inflatable pressure seals deflated.	See figure 21-00-2. Refer to procedures for pressure and leakage check of inflatable seals and adjusting cabin door inflatable seal switch.	Adjust main door switch when necessary. Replace cabin door seal.
(2) Faulty door seal solenoid pressure regulator.	See figure 21-00-2 and run continuity check. Refer to check-out procedure.	Replace seal pressure regulator if faulty.
(3) Faulty cabin air dump and safety valve, or this valve dumping cabin pressure through improper control of electrical or manually controlled pneumatic control line.	Refer to check-out procedure. Also, make sure all electrical control switches and control circuits affecting cabin air dump and safety valve operation are not faulty.	Replace faulty cabin air dump and safety valve or correct cabin air dump manual control valve adjustment.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
(4) Pressure compartment leakage exceeding normal leakage rate.	Refer to procedure for checking cabin pressurization.	Refer to procedure for cleaning surfaces and repairing leaks in compartment pressure sealing.
D. CABIN FAILS TO PRESSURIZE.		
(1) Excessive cabin leakage.	Listen for leaks.	Locate and seal cabin leaks.
(2) Inadequate air supply to cabin.	Check engine bleed-air supply.	Clean or replace valve.
(3) Filter clogged.	Check filter inlet.	Replace filter element.
(4) Foreign objects between outflow valve seats of regulator or dump valve.	Check visually.	Remove objects and clean valve seats.
(5) Defective regulator or cabin air dump and safety valve.	Check air exhaust from regulator or dump valve.	Replace regulator or cabin air dump valve.
(6) Differential-pressure and isobaric control lever improperly positioned.	Check for proper ground-test or flight position.	Position properly, and safety with lockwire for flight.
E. OSCILLATION AND SURGING OF CABIN PRESSURE - Aircraft NA-370-2 Through -6, -8 and -9.		
(1) Plugged or restricted supply pressure orifice in water separator delta P sensor.	Check supply pressure orifice in delta P sensor for contaminants. Check delta P sensor supply line in-line filter for contamination.	Clear supply pressure orifice and clean in-line filter NOTE: Filter is installed with free-flow to supply pressure orifice.

TROUBLESHOOTING (Continued)

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
F. VACUUM PUMP INOPERATIVE		
(1) Vacuum pump motor burnt out.	With power applied to airplane, position either engine master switch at ON. Check for 28 VDC at pin A of electrical plug 889 on vacuum pump motor.	If 28 VDC is available at pin A or P889, replace vacuum pump motor.
(2) Circuit breaker on vacuum pump released. [1]	Check reset pushbutton.	Reset pushbutton on pump motor.
G. VACUUM PUMP DOES NOT OPERATE WHEN EITHER ENGINE MASTER SWITCH IS TURNED ON		
(1) Faulty vacuum pump control relay.	With power applied to airplane and both engine master switches at OFF, check for 28 VDC at terminal X1 of vacuum pump control relay (K1040).	If 28 VDC is available at terminal X1 of relay K1040, replace faulty relay.
H. RAPID LOSS OF PRESSURIZATION WHEN CABIN AIR SELECTOR IS POSITIONED AT OFF		
(1) Faulty cabin conditioned air check valves.	Remove outlet duct from water separator; then apply air source to eyeball outlet and check that no air flow exists from separator duct.	Replace check valve.
I. POPPING SOUNDS IN CABIN DURING TAKEOFF		
(1) Faulty water separator by-pass valve.	Remove valve and inspect for proper spring tension.	Replace spring or complete valve.
(2) Faulty delta P anti-ice valve or 35° temp. sensor.		Replace applicable valve.

[1] S/N 380-57 and subs, and airplanes changed by S/L 36.

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PRESSURIZATION CONTROL - MAINTENANCE PRACTICES

1. Replacement of Cabin Pressure Regulator

A. Removing Cabin Pressure Regulator

(1) Remove access 9. (Refer to Chapter 6.)

(2) Remove the three lines attached to regulator. Cap and/or plug "B" nuts and fittings.

(3) Support regulator; then remove clamp attaching regulator to duct. Remove regulator.

B. Cabin Pressure Regulator Noise Suppression Ring Maintenance

If the polyurethane noise suppression ring should become disconnected from the valve baffle or if a new ring is to be installed, proceed as follows:

(1) Remove cabin pressure regulator from the airplane.

(2) Clean both surfaces to be cemented.

(3) Apply Stabcnd'C242 cement to both surfaces.

Coating thickness should be medium and two coats should be applied. When the adhesive obtains a tacky condition, the surface should be joined.

NOTE: Stabond C242 is manufactured by Stabond Corporation. 14010 So. Western Avenue, Gardena, 90249.

(4) The 146037 noise suppression ring is mounted level with the outer-lip of the baffle plate.

(5) A torn or loose ring may be recemented using the above method. Normally the ring is assembled and bonded to the baffle when the valve is disassembled. With care a new ring could be attached by the above method if split once. However, because of the confined area of the ring in the valve, ring replacement is not recommended as a standard field repair.

(6) Reinstall cabin pressure regulator.

C. Installing Cabin Pressure Regulator

(1) Remove caps, plugs, or coverings from regulator, lines, and fittings.

(2) Place gasket in position on duct flange. Support and position regulator to duct flange; then install clamp loosely.

(3) Shift regulator as necessary, to align it with pneumatic control lines. Start "B" nuts and tighten with fingers.

(4) Torque mounting clamp to 25-35 inch-pounds.

(5) Torque "B" nuts to 40-65 inch-pounds.

(6) Install access.

2. Replacement of Cabin Air Dump and Safety Valve

A. Removing Cabin Air Dump and Safety Valve

(1) Open access 12. (Refer to Chapter 6.)

(2) Disconnect pneumatic control lines from cabin air dump and safety valve.

(3) While supporting body of safety valve, loosen mounting clamp. Disconnect flex line from safety valve and remove cabin air dump and safety valve through access.

(4) Cap and/or plug all lines and fittings.

B. Installing Cabin Air Dump and Safety Valve

(1) Remove all coverings, caps, and/or plugs from valve and lines.

(2) Place gasket in groove on adapter.

(3) Connect hose assembly to fitting in spider of outflow gate. Torque "B" nut to 40-65 inch-pounds.

MAINTENANCE PRACTICES (Continued)

Removing Cabin Air Dump and Safety Valve (Cont'd)

4. Place valve in position on adapter, and loosely install mounting clamp.
5. Rotate regulator as necessary to align fittings with lines. Install lines and torque to 40-65 inch-pounds.
6. Tighten mounting clamp; then torque it to 25-35 inch-pounds.
7. Install access.

CLEANING CABIN PRESSURE REGULATOR AND CABIN AIR DUMP AND SAFETY VALVE

To clean valves use dry cleaning solvent (Federal Specification P-D-680 or equivalent) or a suitable household detergent (Joy) mixed one part detergent to two parts water. Proceed as follows:

1. Remove valve from airplane. (Refer to Replacement of Cabin Pressure Regulator or Replacement of Cabin Air Dump and Safety Valve.)

CAUTION

Do not disassemble valve.

CAUTION

Do not use any tools or other objects to open the poppet valve(s). Move valve with fingers, avoid uneven loading which will place a side load on the poppet valve guide. Bending of this guide will result in valve malfunction. To hold valve open during cleaning use a soft padded horseshoe-shaped spacer which will assure even loading of poppet.

CAUTION

Avoid using excessive solution. Use care to prevent solution from entering vent bleed holes and areas which cannot be wiped clean.

2. Open valve as required. Apply cleaning solution to valve spider, diaphragm, and seat with a soft lint-free cloth. (See Figure 21-30-401.)
3. Allow solution to stand until dirt and tar are loosened; then wipe parts with a soft lint-free cloth. Repeat cleaning procedure as required.
4. Install valve in airplane. (Refer to Replacement of Cabin Pressure Regulator or Replacement of Cabin Air Dump and Safety Valve.)

REPLACEMENT OF CABIN AIR DUMP SOLENOID CONTROL VALVE

Removing Cabin Air Dump Solenoid Control Valve

1. Remove access 13. (Refer to chapter 6, Figure 6-00-3.)
2. Remove necessary electrical wires and equipment.
3. Work rubber cap on top of valve back onto wires; then disconnect terminals from valve.

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(4) Remove mounting screws and remove valve.

(5) Remove filter screen and fitting from bottom of valve. Clean screen and check for re-use.

B. Installing Cabin Air Dump Solenoid Control Valve

(1) Check O-ring gasket on filter assembly, and install filter fitting in bottom of valve.

(2) Place valve in position on bulkhead and install two screws and washers, being sure to clamp wire.

(3) Attach line and torque "B" nut to 40-60 inch-pounds.

(4) Connect electrical wires to valve; then install rubber cap.

(5) Install electrical wires.

(6) Install access.

4. Inspect Cabin Air Dump and Safety Valve Area

A. To inspect cabin air dump and safety valve area, proceed as follows:

(1) Open applicable access doors.

(2) Inspect accessible flight control bellcrank and push-pull control rods for security, chafing and proper safeties.

(3) Inspect pneumatic lines for kinks, chafing, proper routing and security.

(4) Inspect structure for corrosion, cracks and distortion.

(5) Inspect cabin air dump valve static air line filter as follows:

(a) Remove filter screen from static line fitting.

(b) Clean screen with dry cleaning solvent and dry.

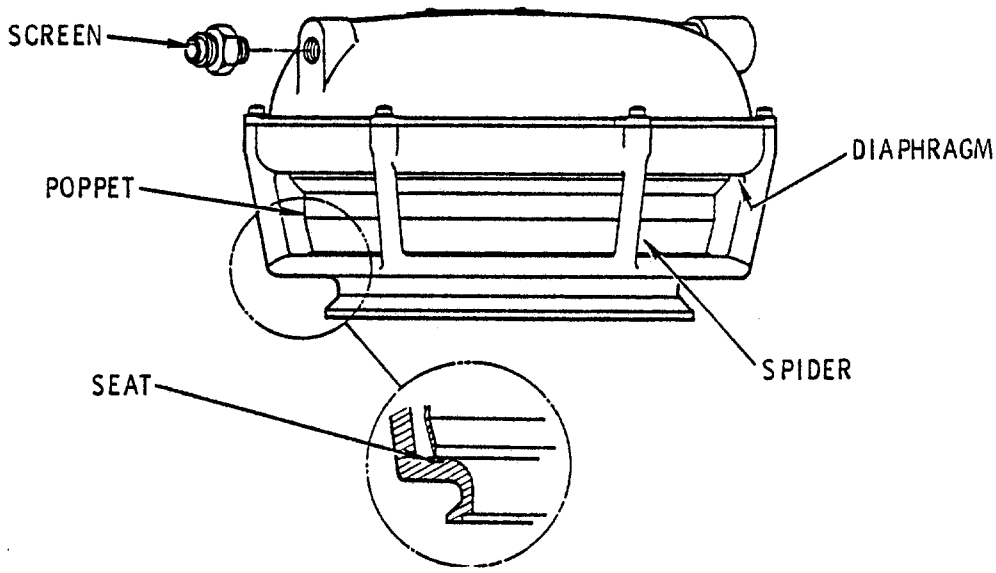
(c) Replace screen if damaged.

(d) Install filter on static line fitting and torque "B" nut.

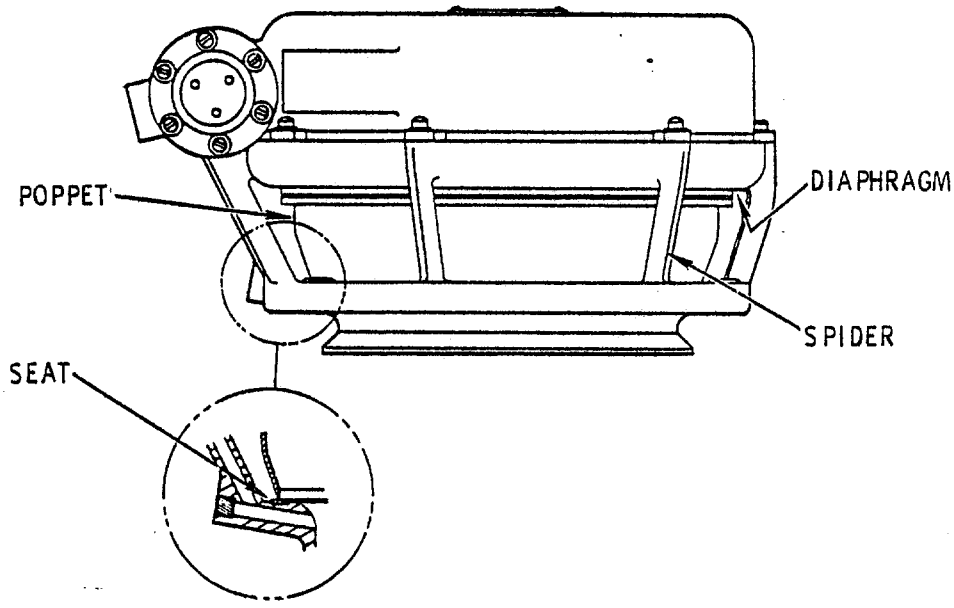
(6) Clean dump and safety valve as required. (Refer to "Cleaning Cabin Pressure Regulator and Cabin Air Dump and Safety Valve".)

(7) Remove cabin air dump and safety valve chamber screen (plug) and clean unit with dry cleaning solvent. Reinstall chamber screen (plug) and safety.

(8) Inspect accessible cables, cable pulleys, and teleflex controls for security chafing and proper safeties.



CABIN AIR DUMP AND SAFETY VALVE (P/N 103238-())



CABIN OUTFLOW VALVE (P/N 103316-())

306-2-53-19

Cabin Air Dump and Safety Valve and Cabin Outflow Valve
Figure 21-30-401

(9) Check area for foreign objects and install access doors.

5. Replacement of Cabin Pressure Regulator Filter Cartridge

(1) Remove access door 9.

(2) Remove cartridge from filter housing.

(3) Install new cartridge. Ensure that cartridge is firmly seated in housing with small screened opening out.

(4) Install access door 9.

5A. Inspect Cabin Air Pressure (Outflow) Valve Area

A. To inspect cabin air pressure (outflow) valve area, proceed as follows:

(1) Open applicable access door.

(2) Inspect accessible flight control bellcranks and push-pull control rods for security, chafing and proper safeties.

(3) Inspect flex lines for deterioration, kinks, security, proper routing and chafing. Check pneumatic (hard) lines for security and damage

(4) Inspect limit control for security and shock mounts for deterioration

(5) Inspect structure for corrosion, cracks and distortion.

(6) Clean outflow valve as required. (Refer to "Cleaning Cabin Pressure Regulator and Cabin Air Dump and Safety Valves".)

(7) Inspect outflow valve and components for damage, cleanliness, cracks, corrosion, security, and proper safety.

(8) Inspect all electrical wiring and connections for cleanliness, deterioration, chafing, and security.

(9) Check for foreign objects and install access doors.

6. Replacement of Vacuum Pump

A. Removing Vacuum Pump

(1) Remove access door. (Refer to Chapter 6.)

(2) Remove electrical plug from vacuum pump motor.

(3) Remove and cap vacuum line attached to pump.

(4) Remove four mounting screws securing vacuum pump to structure.

(5) Remove vacuum pump from airplane.

(6) Note position of fittings in pump, then remove fittings and retain for installation in replacement pump.

B. Installing Vacuum Pump

(1) Install fittings in pump.

NOTE: Before installing vacuum pump, ensure that rubber mounts are serviceable, and that pipe plug on side of pump is tight. Remove the filter screen fitting from the outlet port (if installed).

(2) Install vacuum pump in airplane with four mounting screws and eight washers. Place the one-inch diameter washers (2W1AL-18-32-63) between screw heads and grommets and place the other washers (LD153-0011-0012) between grommets and pump support bracket.

(3) Install and tighten vacuum pump line.

(4) Connect electrical plug to vacuum pump motor. Lockwire electrical plug to pipe plug on side of vacuum pump.

(5) Install access door.

(6) Perform operational check out of vacuum pump.

MAINTENANCE PRACTICES (Continued)

AIR LEAKAGE IN PRESSURIZATION CONTROL SYSTEM

Locating Leaks in Pressure Compartment Sealing

1. All possible points of air leakage around the pressurized compartment areas must be sealed to maintain pressure. Three methods of sealing are used. Rubber seals are used at all points where frequent breaking of the seal is necessary (such as entry and escape doors). Sealing compounds are used at points where no breaking of the seal is expected, except for maintenance. Special seals are used for passing cables, tubing, mechanical linkages, and wiring out of the pressurized area. The wires and tubing are passed through pressure seals on bulkhead fittings to which the wires or tubes are fastened from each side. The flight controls are passed through the bulkheads by special seals.
2. With the compartments pressurized, isolate leaks on inside by running a piece of rubber tubing or a stethoscope along seams in leakage area. Listen for a change in sound when tube is passed over a leak. Leaks may be verified by an assistant on outside of airplane.

   While the compartments are pressurized, a thorough interior and exterior inspection of all accessible fuselage seams, structure and points of attachment to and through the exterior surfaces of the pressurized area should be accomplished. Particular attention should be given to proven areas of leakage to determine that leakage is not the result of cracks, loose or missing fasteners or damaged seals.

WARNING

When anyone is inside, the compartments must never be subjected to a higher pressure than has previously been safely established with the compartments empty.

WARNING

It is recommended that only personnel who are medically checked and approved be permitted to occupy compartments when they are pressurized on the ground.

WARNING

Anyone who has recently had a cold, or who has sinusitis or nasal congestion should not work in the compartments when they are pressurized.

WARNING

An experienced operator should be present at the ground tester pressurizing equipment control panel at all times while the compartments are pressurized.

WARNING

With personnel inside pressure compartments, the pressure change should never exceed 1000-foot-per-minute altitude change, as shown on a rate-of-change indicator.

Cleaning Surfaces and Repairing Leaks in Compartment Pressure Sealing.

1. To ensure maximum adhesion between the sealing compound and the surface, all surfaces to be sealed must first be cleaned.
2. With a vacuum cleaner, thoroughly clean all loose dirt from surface.
3. Moisten a clean cloth with naptha or solvent. Clean area around spot to be sealed.
4. To clean thoroughly, repeat step 3 as often as necessary. Never use a soiled cloth. Discard cleaning cloth as soon as it becomes soiled.

NOTE

Repair sealing compound should not be used for basic sealing, because of marginal adhesion to bare or metal primed surfaces. However, adhesion of the repair sealing compound to that of basic sealing compound is excellent. Therefore, beads of repair sealing compound should be limited to a maximum of 2 inches in length on bare or primed metal surfaces.

5. Seal holes and gaps up to 1/16 inch in width with a fillet of repair sealing compound. No backing strip is necessary.

NOTES

- *Reference the Structural Repair or Corrosion Control Manual for sealant types.*
- *Sealing compound must always be applied on the pressure side. A second coat of sealing compound may be applied after the first coat has been allowed to cure completely. Curing time is 24 hours at about 77°F, provided 4-hour work-life material is used. If applicator guns are used to apply the sealant, they should be cleaned as soon*

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MAINTENANCE PRACTICES (Continued)

Cleaning Surfaces and Repairing Leaks in Compartment Pressure Sealing (Cont'd)

NOTES (Cont'd)

as possible with cleaning solvent; otherwise, any sealant that cures inside the gun must be cleaned out by hand. Do not apply sealing compound by picking it up after wetting fingers in water, soap or protective hand cream. A film from any of these solutions will cause the compound to adhere poorly. Leaks can be repaired from inside while the cockpit is pressurized. Repairing under pressure is recommended because it is faster and the results can be checked immediately.

6. Seal holes and gaps from 1/16 inch to 3/16 inch in width with a backing strip of pressure-sensitive tape. Apply tape over opening on pressure side, and apply a fillet of general-purpose sealing compound over tape, allowing a 1/8-inch overlap from edge of tape.

NOTES

Remove compound from tools within one hour after use. Removal of the compound is difficult after it begins to cure. To prevent skin irritation, personnel should wear gloves while using cleaning compounds.

7. To repair cracks or damage in structure, refer to Sabreliner Structural Repair Manual Report No. NA-66-1032.
8. Within one hour after mixing repair compound and accelerator, the sealing equipment should be cleaned, using toluene solvent. Otherwise, the compound will harden and can only be removed by mechanical means. However, cleaning of equipment may be delayed for about 72 hours, provided the uncleaned equipment is placed in refrigeration at zero to 10°F.

PRESSURIZATION CONTROL SYSTEM CHECKOUT PROCEDURE

Checking Cabin Pressurization

NOTE

The maximum allowable leak rate is 144 cubic feet per minute (cfm).

1. Make sure the following circuit breakers are pushed in:
   - DR SEAL & LOCK IND
   - CABIN — PRESS DUMP & PRESS DUCT FAIL
   - LDG GEAR — POS

CAUTION

To prevent damage to equipment and installations, make certain that step 2 is completed before proceeding with cabin pressurization checks.

2. Remove the following instruments (refer to chapter 31) and plug or cap all lines disconnected from removed instruments:
   - Cabin altimeter and differential pressure indicator
   - Cabin pressure rate-of-change indicator

WARNING

To prevent damage to equipment and possible injury to personnel, observe all instructions given on ground pressure test panel.

3. Connect a controllable source of air pressure to CABIN AIR fitting on cabin ground pressure test panel. Do not apply pneumatic pressure. (See Figure 21-30-606.)
4. Install a 1 to 10 psi pressure gauge (capable of providing pressure differential readings in 0.1 psi increments) to CABIN PRESS fitting on cabin ground pressure test panel.
5. Remove access panel 7 (see Figure 6-00-3, sheet 2 of 4 or Figure 6-00-4, sheet 2 or 4).
6. Remove lockwire from ground pressure test valve on cabin air isobaric valve and gound test shutoff valve. Turn valves to test position.

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MAINTENANCE PRACTICES (Continued)

PRESSURIZATION CONTROL SYSTEM CHECKOUT PROCEDURE (Cont'd)

7. On ground pressure test panel, cut lockwire to release guarded DUMP switch. Position switch at power-off position.
8. Disconnect right and left main landing gear load switch links (refer to 32-60-01). Leave load switch links in load (struts compressed) position.
9. Connect external 28 VDC electrical power to airplane (refer to chapter 24).

WARNING

During steps 10 thru 21, only medically checked and approved personnel should be allowed to occupy ground-pressurized compartments. Pressure changes inside occupied compartments must not exceed 1000-foot-per-minute altitude changes as shown on a pressure rate-of-change indicator. While compartments are pressurized, an experienced operator must be posted at all times at the pressurizing equipment panel.

10. Close all doors and windows in the cabin and cockpit. Make sure that entrance door handle is in fully locked position. Check that floor drain valve is closed.
11. Close all accesses in the nose section (S/N 370-1 thru -7 and 380-1 thru -23 not changed by S/L No. 7).
12. Connect a 25 to 200 psi, controllable, air pressure source to SEAL REG fitting on the ground pressure test panel. Apply 90 psig air pressure at SEAL REG fitting.

CAUTION

Pressure at this fitting should not exceed 175 psig or be less than 30 psig.

13. Position and leave left main gear load switch in unload (airborne) position. Check to ensure that no change takes place.
14. Position and leave right main gear load switch in unload (airborne) position. Check to ensure that main entrance door seals inflate.
15. Position either main gear load switch to load position. Check that seal deflates. Reposition switch to unload position. Check that seal inflates.
16. Move entrance door handle out of locked position. Check to ensure seals deflate.
17. Lock main entrance door. Check to ensure seals inflate.
18. After observing placard on ground pressure test panel and being thoroughly acquainted with warning preceding step 10, pressurize cabin to 8.8(+-0.1) psi above atmospheric pressure. Check that leakage rate does not exceed 10.8 pounds per minute, or 144 cubic feet per minute.

WARNING

To prevent possible injury to personnel, make certain that all personnel have left the cabin and cockpit before proceeding with the next step.

19. If the Auxiliary Fuel Cell (Bladder) is removed from the fuselage, spray the aft side of the F.S. 333 bulkhead with a Non-Corrosive Gas and Leak Detector, Sherlock, P/N CG1, or equivalent. Any leaks that are discovered should be marked and repaired from inside of the cabin after the test is completed. If no leaks are detected on the F.S. 333 bulkhead, proceed to step 21.

WARNING

Absolutely no leakage is allowed through the F.S. 333 bulkhead because, in the event of a fuel cell bladder leak, fuel or fuel fumes could enter the cabin environment.

20. If the Auxiliary Fuel Cell (Bladder) is still installed on the aircraft, open the gravity refueling port on the right side of the fuselage above the pylon. Use a strong flashlight and examine the inside of the auxiliary fuel cell for signs of collapse. If there is a pressurization leak through the F.S. 333 bulkhead, the auxiliary fuel cell may be compressed to the point where the mounting springs pull free of their attach points, allowing the fuel cell bladder to fall away from its normal location (droop). If the auxiliary fuel cell (bladder) is found to be even partially collapsed, it must be completely removed from the aircraft and the leak in the F.S. 333 bulkhead repaired. If this condition is found to exist, fueling of the auxiliary fuel cell should be avoided until repairs are made, preventing a puncture and subsequent fuel leak condition. Repeat this test until no leaks are detected in the F.S. 333 bulkhead.

NOTE

This allowable leakage rate includes 6.7-cubic-feet-per-minute leakage from cabin pressure regulator, 6.7-cubic-feet-per-minute leakage from cabin safety and dump valve, and 2.4-cubic-feet-per-minute leakage from water separator drain line. If leakage is excessive, refer to paragraph 1 for information on locating leaks, and for cleaning and resealing procedures.

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MAINTENANCE PRACTICES (Continued)

PRESSURIZATION CONTROL SYSTEM CHECKOUT PROCEDURE (Cont'd)

21. Increase pressure in cabin and check that pressure relief feature of cabin air dump and safety valve maintains the pressure between 9.0 and 9.2 psig. The valve must relieve cabin pressure at 9.1 (±0.1) psig.

CAUTION

Exercise extreme care to ensure that only enough air pressure is applied to permit pressure relief through valve. Under no circumstance is the cabin pressure to exceed 9.5 psig. If 9.5 psig is exceeded, all equipment vented to cabin should be removed and checked for calibration.

NOTE

If inflatable door seals require replacement, refer to 21-32-02.

ELECTRONIC NOSE COMPARTMENT PRESSURIZATION CHECKOUT (EXTERNAL AIR SOURCE) (S/N 370-1 THRU -9, 380-1 THRU -23 NOT CHANGED BY S/L No. 7)

Perform nose compartment leakage test as follows:

1. Remove cabin pressure regulator (air outflow valve) (ref. 21-31-01) and replace and seal with a plate having an external air fitting.
2. Install access doors and covers required to close out pressurized nose compartment.

CAUTION

Do not exceed 3.30 psig during steps 3 and 4.

3. Pressurize nose compartment to 2.43 psig (through plate fitting) and note leakage. Maximum allowable leakage (in-service airplanes) is 1.05 pounds per minute (14 CFM based on standard pressure at 70°F). If leakage is found excessive, reseal compartment as required following procedures outlined under **Cleaning Surfaces and Repairing Leaks in Compartment Pressure Sealing.**
4. After leakage rate has been established within limits, gradually pressurize compartment to 3.05 (±.2) psig and verify that relief valve relieves within noted range (as evidenced by no further increase in compartment pressure).
5. Open access door just forward of landing light and reset relief valve (per instructions on decal adjacent to valve).
6. Remove equipment installed for test, then install cabin pressure regulator and access doors.

IN-FLIGHT PRESSURE LEAKAGE CHECK

Perform an in-flight pressure leakage check of the air conditioning and pressurization systems as follows:

1. Conduct the following test at 25,000 feet without passengers, with pilot, copilot on oxygen, and with inspector (if applicable) on portable oxygen bottle supply.
2. Prior to takeoff, check that both engine bleed-air valves close properly (as indicated by no air flow from cockpit air outlet ports).
3. With cabin air selector switch at BOTH ENG, set cabin pressure controller at 8,000 feet, and rate of change control to 2,000 feet.
4. Position passenger oxygen switch at CLOSED.

NOTE

Close all cabin and cockpit fresh air outlet valves prior to step 5.

5. Check that cabin altitude is 8,000 (±240) feet, then position cabin air selector at OFF. Cabin altitude should not exceed 19,000 feet in three minutes.

NOTE

Accomplish step 6 provided step 5 limit is satisfactory.

6. With cabin PRESSURIZATION OFF, open one cabin fresh air outlet valve. Using hand, piece of paper or cloth, check for reverse flow of cabin air into supply line due to leaking one-way check valve at FS 333.

NOTE

Accomplish step 7 provided step 5 limit was satisfactory during step 6.

7. With cabin PRESSURIZATION OFF, close all cabin fresh air outlet valves and open one cockpit fresh air outlet valve. Using hand, piece of paper or cloth, check for reverse flow of cabin air into supply line due to leaking one-way check valve at FS 333.
8. Reduce right engine power to 75% rpm. Position cabin air selector switch to EMER PRESS. Cabin altitude should decrease. Position right engine power lever as required.
9. Position cabin air selector switch to BOTH ENG. Cabin altitude should descend rapidly to 13,500 (±1500) feet, may hold up to 6 minutes, then rate towards 8,000 feet. Cabin altitude should reach 8,000 (±240) feet in about 8 minutes total elapsed times.

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INSPECTION/CHECK (Continued)

IN-FLIGHT PRESSURE LEAKAGE CHECK (Cont'd)

10. Resume normal flight.

FLIGHT CHECK OF CABIN AND COCKPIT COMPARTMENT PRESSURIZATION (S/N 370-1, -7, 380-1 and Subs)

To perform a flight check of cabin and cockpit pressurization system, proceed as folows:

1. Position environmental system controls as follows:
   - CABIN TEMP and COCKPIT TEMP switches at AUTO.
   - CABIN TEMP and COCKPIT TEMP rheostat at desired temperature.
   - CABIN AIR switch — BOTH ENG.
   - CABIN PRESSURE CONTROL — cabin altitude — 8,000 feet. Rate of change — 500 feet/min.
2. With airplane flying at 20,000 feet, the cabin pressure should reach 8,000 feet after approximately 16 minutes.
3. Select sea level cabin altitude and adjust the rate of change controller to a cabin rate of descent of 250 feet per minute.
4. Observe the rate of change for 2 minutes (by stop watch) and note that the cabin altitude decreases 500 feet.
5. Readjust rate of change controller to give a cabin rate of descent of 1000 feet per minute and note rate of change for one minute (by stop watch). Check that the cabin altitude decreases 1000 feet.
6. Allow cabin pressure to stabilize at sea level ±240 feet.
7. Select a cabin altitude of 4000 feet and check that the cabin altitude stabilizes at 4000 feet in 4 minutes.
8. Select a cabin altitude of 8000 feet and set the rate of change control to 2000 feet per minute and check that cabin altitude stabilizes at 8000 feet in 2 minutes.

CAUTION

Oxygen masks must be worn during the following test.

9. Climb to 25,000 feet. The cabin pressure should remain constant at 8000 ±240 feet. Position CABIN AIR switch to RAM & DUMP. Check that cabin altitude increases from 8000 to approximately 25,000 feet.
10. Position CABIN AIR switch to BOTH ENG. Check that cabin altitude returns to 8000 feet in approximately 9 minutes.
11. Reduce right engine rpm to 75%. Position CABIN AIR switch to EMER PRESS. Check that cabin altitude decreases to 13,500 ±1500 feet for approximately 4 minutes, then rate toward 8000 feet.
12. Position CABIN AIR switch to BOTH ENG. Cabin altitude should return to 8000 feet.
13. Climb to 45,000 feet. Check that cabin altitude remains at 8000 ±240 feet. Select a sea level cabin setting.

NOTE

If field of departure and landing is at an altitude greater than sea level, set the cabin altitude for 500 feet higher than field elevation.

14. Position environmental system controls as required.
15. Resume normal flight.

OPERATIONAL CHECKOUT OF VACUUM PUMP

1. Check that VACUUM PUMP circuit breaker is engaged (pushed in).
2. Connect external 28 VDC power to airplane.
3. Position LH ENG MASTER switch to ON.
4. Check that vacuum pump is operating.
5. Position LH ENG MASTER switch to OFF. Vacuum pump operation should cease.
6. Disconnect external 28 VDC power from airplane.

GROUND PRESSURE LEAKAGE CHECK — ENGINE OPERATING

WARNING

Anyone who has recently had a cold or who has sinusitis or nasal congestion should not work in the compartments when they are pressurized.

1. Position airplane in a location where ground runup is permitted.
2. Pull the landing gear position circuit breakers.
3. Open access panel 7. (Refer to Chapter 6.)
4. Remove lockwire from ground pressure test valve. Set valve to ground test position.
5. Open pilot's sliding window. Close main entrance door. Start right engine.

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INSPECTION CHECK (Continued)

GROUND PRESSURE LEAKAGE CHECK — ENGINE OPERATING (Cont'd)

6. Set pressurization switch to RH ENG. Toggle CABIN TEMP control to MAN COLD and hold for 10 seconds, then return to OFF.
7. Slowly close pilot's sliding window and allow cabin pressure to build to 5 psi differential.
8. Toggle CABIN TEMP control towards MAN HOT (helps prevent pressure bump) and increase engine rpm until cabin pressure is 8.2 psi differential.
9. Extend speed brake.
10. Secure RH engine. Select cabin air to OFF. Set inverter to OFF.

NOTE

If battery switch is selected to OFF, cabin pressure will dump.

11. Ensure battery switch remains ON.
12. Check for fuselage air leaks, especially around the sealed escape door and at all duct connections in the aft fuselage compartment. Mark any leaks for later resealing.
13. After fuselage leak check, slowly release cabin pressure.
14. Push in LDG GEAR POS circuit breaker. Set ground test valve to OFF and resafety wire.

FUNCTIONAL CHECK OF FLOW LIMITING CONTROL

The functional check of the flow limiting control verifies that the control is operating smoothly. To perform a functional check of the flow limiting control, proceed as follows:

1. Open the pilot's sliding window and close the main cabin entrance door.
2. Set cabin pressurization controller 5,000 ft. above field elevation, and rate at maximum. Set cabin air selector to OFF.
3. Open aft lower fuselage doors (access to flow limiting control).
4. Start both engines (Refer to Chapter 71). Select NO. 1 or main inverter.
5. Position CABIN AIR selector to BOTH.
6. Station a helper in the aft fuselage to monitor operation of flow limiting control.

WARNING

To prevent possible injury to personnel, use extreme caution when working adjacent to an operating engine.

7. Position cabin heat control to manual hot and hold for approximately 10 seconds. This will fully open the heat control valve which allows maximum air flow.
8. While monitoring clevis end of flow limiting control, advance RH engine power lever until 81 to 85% engine RPM is reached. Check that valve clevis moves slowly and smoothly form 3/8 to 1/2 inch.
9. Slowly reduce RH engine power lever to IDLE. Check that valve moves smoothly to the full open position.
10. Shut down both engines (Refer to Chapter 71).
11. Position all switches and controls in their normal OFF position.
12. Close access doors.

FUSELAGE AUXILIARY FUEL CELL BAY PRESSURIZATION AND LEAKAGE TEST

NOTES

- *Removal of F.S. 333 interior is required to perform this test.*
- *AUX Fuel Cell must be removed to perform this test.*
- *Contact Sabreliner Corporation for Test Cover Plate-Kit (P/N 265-048001) before performing this test.*
- *Use Non-corrosive Gas and Leak Detector, Sherlock, P/N CG1, or equivalent, or Amprobe/ Pasar-TMULD-100 Testmaster Kit by AMPROBE-Lynbrook, NY. (Ultrasonic)*

Procedure A — Utilizing Standard NR leak test panel No. TSC-10083 or equivalent.

1. Install test cover plate (Kit P/N 265-048001). See Figure 21-30-601.
2. Connect an external tube type water manometer with a water level of 60 to 80 inches of water, and a 0 to 15 psi air pressure gauge.

CAUTION

Do not exceed 2.5 psi as AUX Fuel Cell Bay structural damage could occur.

3. Connect Test Panel No. TSC-10083 to the test cover plate as illustrated on Figure 21-30-601.
4. Pressurize Fuel Cell Bay as follows:

NOTE

The leak test unit contains two systems, a R.H. system color coded blue, and a L.H. system color coded yellow. Use only the blue system. The red colored controls and gage in the center are used for tester supply air pressurization. The manometer for the blue coded system has a range of 0 to 16.5 inches of water.

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INSPECTION CHECK (Continued)

FUSELAGE AUXILIARY FUEL CELL BAY PRESSURIZATION AND LEAKAGE TEST (CONT'D)

A. Check to ensure air shut off valve No. 1 and shut off valve No. 2 are closed. Open shut off valve No.'s 3 and 4. See Figure 21-30-603.

B. Connect an air pressure source to the leak tester unit.

C. Open the single point pressurization valve (red trimmed). Open the refueling regulator (red handle) until 25 psi is indicated on the red colored pressure gage.

D. Slowly open shut off valve No. 1, Monitor the Fill Pressure Gage, Sensing Pressure Gage, and external water manometer unit (pressure relief) to determine that pressure is being supplied and that the pressure does not exceed 2.5 psi.

WARNING

No leakage is allowed from the fuel cell bay forward into the cabin, because in the event of a Fuel Cell bladder leak, fuel fumes could enter the cabin environment.

5. Apply leak detector to the entire forward surface of the F.S. 333 bulkhead and check for leaks. If leaks are detected, depressurize, seal leaks, and retest until no leaks into the cabin are detected.

NOTE

Proceed to Step 6 only if no leaks are detected on the F.S. 333 bulkhead.

6. Both pressure gages should be stabilized at 2.5 psi, and external water manometer should be stabilized at 68 to 70 inches of water to verify pressure gages.

CAUTION

External water manometer is to be used ONLY as pressure relief protection. DO NOT use the external water manometer as a pressure monitor.

7. Close shut-off valves 3, 4, and 1 in the order listed.
8. Close single point pressurization valve, and main air supply valve (red handles). Disconnect air supply from tester unit. Check fitting at leak tester unit for leakage with a bubble solution.
9. Monitor the test unit water manometer and record pressure value indicated.
10. After 30 minutes has elapsed, record pressure again. If pressure has decayed more than 4.0 inches of water, leakage is excessive.
11. If leakage is excessive, locate the leakage areas using bubble solution or Ultrasonic Leak Detector. De-pressurize, reseal and retest until pressure decay is within allowable range (0 to 4.0 in. of water).
12. When a successful test has been done, open shut-off valves 2 and 4 to relieve the pressure in the system.
13. Disconnect Leak Test Unit and Water Manometer/Pressure Gage.
14. Remove test cover plate, and secure aircraft as desired.

PROCEDURE B — Using locally fabricated Leak Test Panel (See Figure 21-30-604) and Cabin Pressurization Tester P/N 1126-17-Type MB-3.

1. Install test cover plate (Kit P/N 265-048001) (See Figure 21-30-601).
2. Connect an external tube type water manometer with a water level of 60 to 80 inches of water, and a 0 to 15 psi air pressure gage.

CAUTION

Do not exceed 2.5 psi as AUX Fuel Cell Bay structural damage could occur.

3. Connect leak test panel to the test cover plate as illustrated on Figure 21-30-601.
4. Connect Cabin Pressurization Tester to leak test panel as illustrated on Figure 21-30-604.
5. Pressurize Fuel Cell Bay as follows:

A. Check to ensure that test panel valves No. 1 and No. 2 are closed. Open test panel valves No. 3 and No. 4.

B. With Cabin Pressurization Tester in operation, close Tester Regulator Valve (No. 5) so that air can be supplied to the test panel.

C. Slowly open test panel valve No. 1. Monitor the Fill pressure Gage, Tank Pressure Gage, and external Water Manometer Unit (Press. Relief) to determine that pressure is being supplied and that the pressure does not exceed 2.5 psi.

WARNING

No leakage is allowed from the fuel cell bay forward into the cabin, because in the event of a Fuel Cell bladder leak, fuel fumes could enter the cabin environment.

6. Apply leak detector to the entire forward surface of the F.S. 333 bulkhead and check for leaks. If leaks are detected, de-pressurize, seal leaks and retest until no leaks into the cabin are detected.

NOTE

Proceed to Step 7 only if no leaks are detected on the F.S. 333 bulkhead.

7. Both pressure gages should be stabilized at 2.5 psi, and external water manometer should be stabilized at 68 to 70 inches of water to verify pressure gage.

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INSPECTION CHECK (Continued)

FUSELAGE AUXILIARY FUEL CELL BAY PRESSURIZATION AND LEAKAGE TEST (CONT'D)

CAUTION

External water manometer is to be used ONLY as a pressure relief protection. DO NOT use the external manometer as a pressure monitor.

8. Close shut-off valves, 3,4, and 1 in the order listed.
9. Open Tester Regulator Valve (No. 5) and shut down cabin Pressurization Tester unit.
10. Disconnect Pressure Tester Unit from the Leak Test Panel. Check fitting at Leak Test Unit for leakage with a bubble solution.
11. Monitor Test Unit water manometer and record pressure value indicated.
12. After 30 minutes has elapsed, record pressure again. If pressure has decayed more than 4.0 inches of water, leakage is excessive.
13. If leakage is excessive, locate the leakage areas using bubble solution or Ultrasonic Leak Detector. De-pressurize, reseal, and retest until pressure decay is within allowable range (0 to 4.0 in. of water).
14. When a successful test has been done, open shut-off valves 2 and 4 to relieve the pressure in the system.
15. Disconnect Leak Test Unit and Water Manometer/Pressure Gage.
16. Remove Test Cover Plate, and secure aircraft as desired.

PROCEDURE C — Using locally fabricated Leak Test Panel (See Figure 21-30-605) and a Shop Vacuum (Sears Craftsman 3.0 HP-16 gal-0 to 3 Psig Max. or equivalent)

NOTE

A No. 200 Mesh Screen should be installed in line between shop vacuum and Leak Test Panel. (Spark Arrestor).

1. Install test cover plate (Kit P/N 265-048001) (See Figure 21-30-601).
2. Connect an external tube type water manometer with a water level of 60 to 80 inches of water, and a 0 to 15 psi air pressure gage.

CAUTION

Do not exceed 2.5 psi. as AUX Fuel Cell Bay structural damage could occur.

3. Connect Leak Test Panel to the test cover plate is illustrated on Figure 21-30-601.
4. Connect Shop Vacuum to the Leak Test panel as illustrated on Figure 21-30-605.
5. Pressurized Fuel Cell Bay as follows:
   A. Check to ensure that Test Panel valves No.1 and No. 2 are closed, open Test Panel valves No. 3 and No. 4.
   B. With Shop Vacuum in operation, close the Regulator Valve (No. 5) so that air can be supplied to the test panel.
   C. Slowly open Test Panel Valve No. 1. Monitor the Fill Pressure Gage, Tank Pressure Gage, and external water Manometer Unit (Pressure Relief) to determine that pressure is being supplied and that pressure does not exceed 2.5 psi.

WARNING

No leakage is allowed from the fuel cell bay forward into the cabin, because in the event of a Fuel Cell bladder leak, fuel fumes could enter the cabin environment.

6. Apply leak detector to the entire forward surface of the F.S. 333 bulkhead, and check for leaks. If leaks are detected, de-pressurize, seal leaks, and retest until no leaks into the cabin are detected.

NOTE

Proceed to Step 7 only if no leaks are detected on the F.S. 333 bulkhead.

7. Both pressure gages should be stabilized at 2.5 psi, and external water manometer should be stabilized at 68 to 70 inches of water to verify pressure gages.

CAUTION

External Water Manometer is to be used ONLY as a pressure relief protection. DO NOT use the external water manometer as a pressure monitor.

8. Close shut-off valves 3, 4 and 1 in the order listed.
9. Open Regulator Valve (No. 5) and shut down the Shop Vacuum unit.
10. Disconnect Shop Vacuum unit from the Leak Test Panel. Check fitting at Leak Test Unit for Leakage with bubble solution.
11. Monitor Leak Test Unit water manometer, and record pressure value indicated.
12. After 30 minutes has elapsed, record pressure again. If pressure has decayed more that 4.0 inches of water, leakage is excessive.
13. If leakage is excessive, locate the leakage areas using bubble solution or Ultrasonic Leak Detector. De-pressurize, reseal, and retest until pressure decay in within allowable range (0 to 4.0 inch of water).
14. When a successful test has been done, open shut-off valves 2 and 4 to relieve pressure in the system.
15. Disconnect Leak Test Unit and Water Manometer/Pressure Gage.
16. Remove Test Cover Plate, and secure aircraft as desired.

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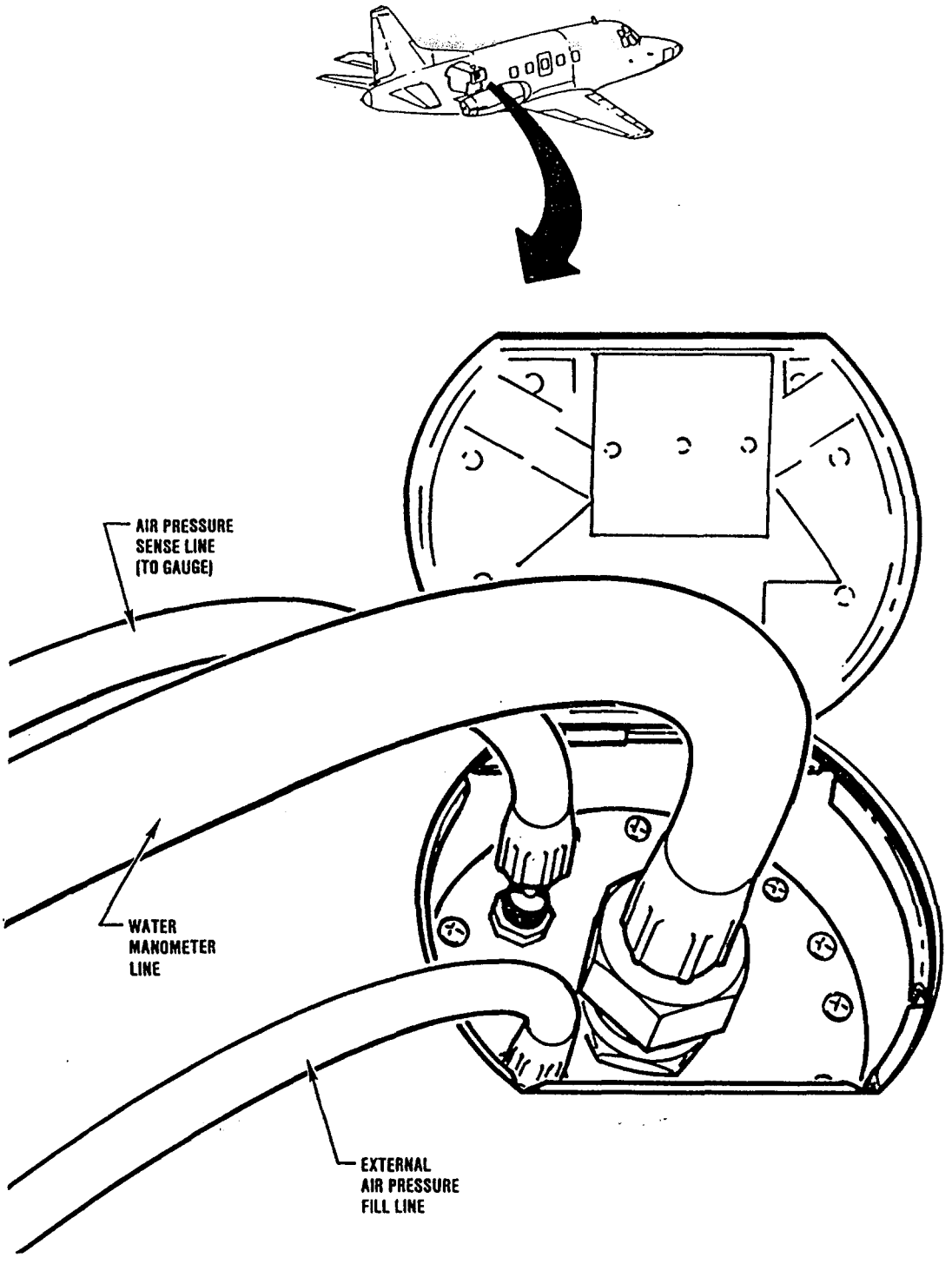

Figure 21-30-601 — Fuselage Auxiliary Fuel Cell Bay Pressurization and Leakage Test — Equipment Hookup

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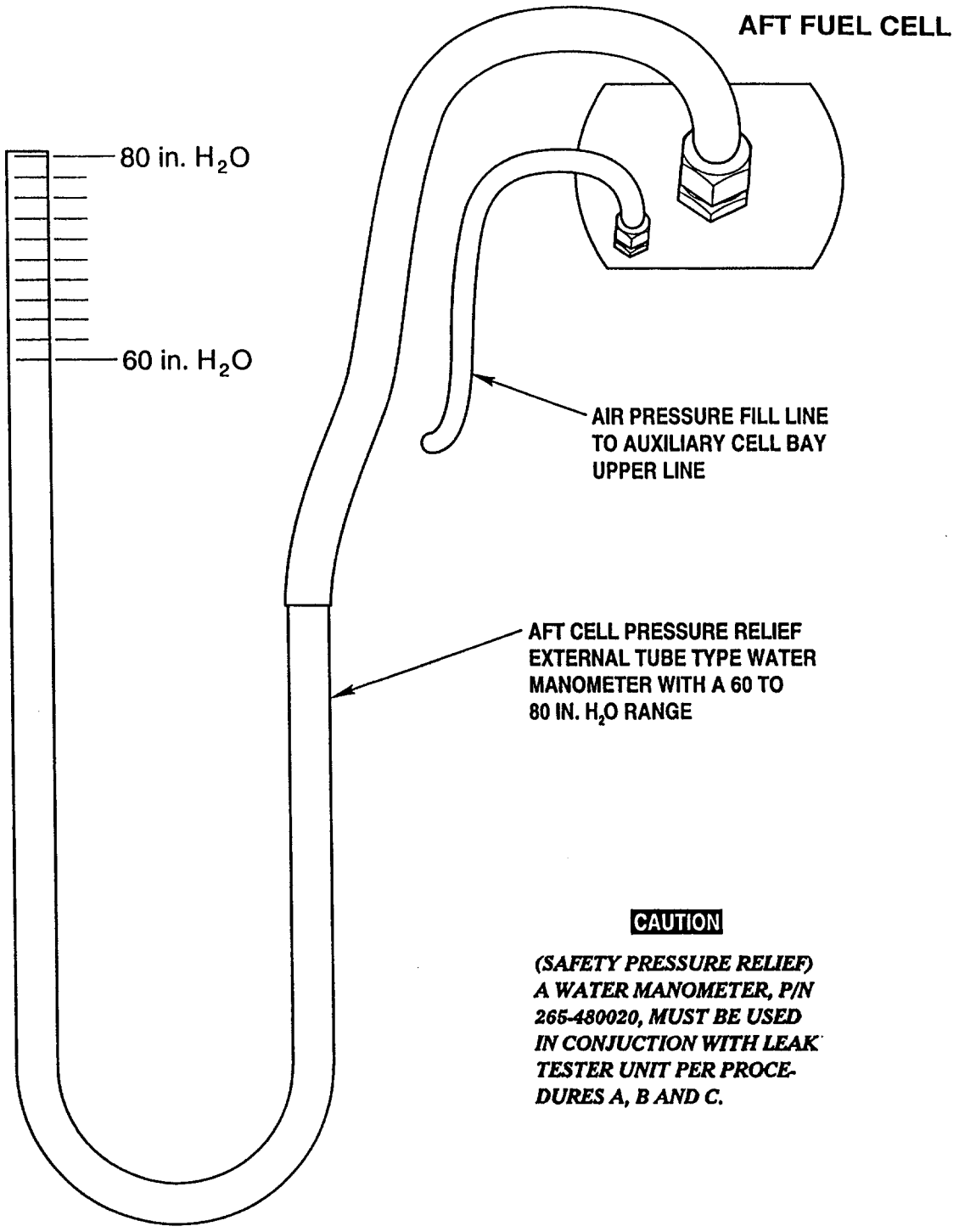
AFT FUEL CELL
80 in. H_2O
60 in. H_2O
AIR PRESSURE FILL LINE TO AUXILIARY CELL BAY UPPER LINE
AFT CELL PRESSURE RELIEF EXTERNAL TUBE TYPE WATER MANOMETER WITH A 60 TO 80 IN. H_2O RANGE
CAUTION
(SAFETY PRESSURE RELIEF) A WATER MANOMETER, P/N 265-480020, MUST BE USED IN CONJUCTION WITH LEAK TESTER UNIT PER PROCEDURES A, B AND C.

Figure 21-30-602 — External Tube Type Manometer

NEW

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LEAK TESTER
UNIT NO. TSC 10083

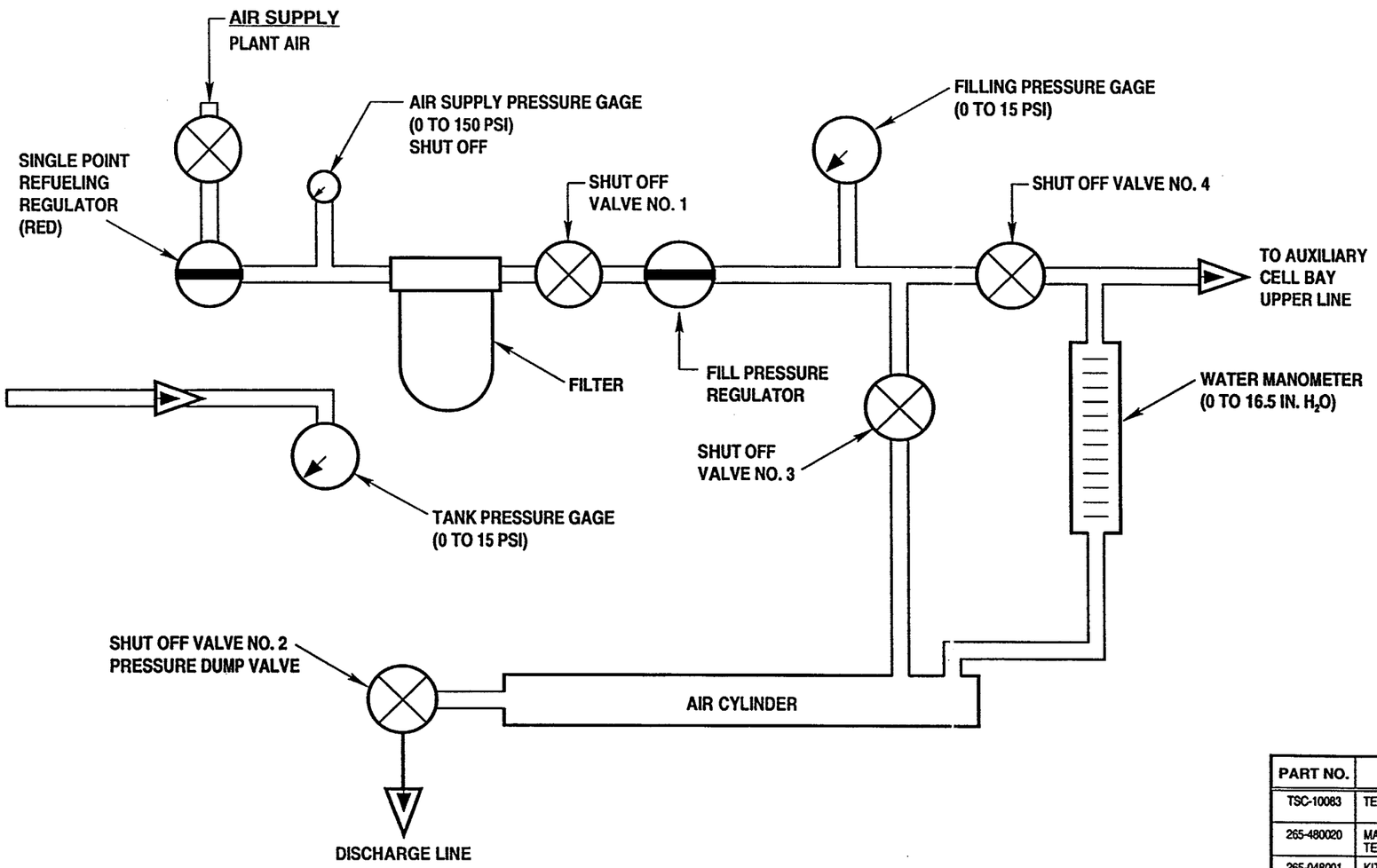

Figure 21-30-603 — Leak Tester Unit — Procedure A.

PARTS LIST

PART NO.	TITLE	VENDOR	MISC.
TSC-10083	TESTER, PANEL	SPRAGUE ENGINEERING NASHUA, NH 03060	
265-480020	MANOMETER — FUEL TANK TEST	SABRELINER CORP. CHESTERFIELD, MO 63005	
265-048001	KIT — COVER PLATE	SABRELINER CORP. CHESTERFIELD, MO 63005	

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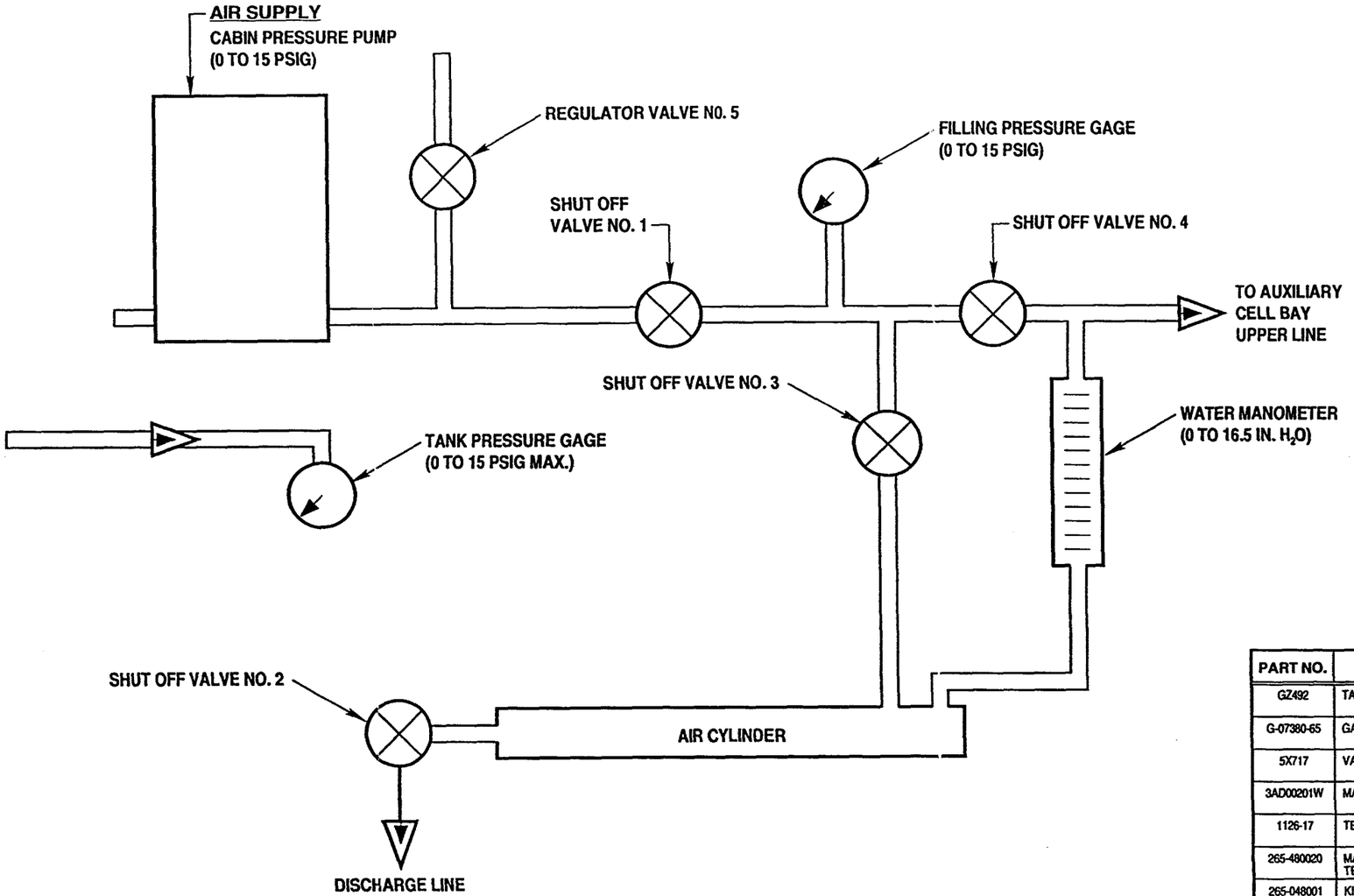
LEAK TESTER
CABIN PRESSURE PUMP



PARTS LIST

PART NO.	TITLE	VENDOR	MISC.
GZ492	TANK — PORTABLE AIR	W.W. GRANGER, INC. NORTH BROOKE, IL 60062	6 GAL
G-07380-65	GAUGE, PRESSURE	COLE-PARMER INTL. NILES, IL 60714	0 TO 15 PSI
5X717	VALVE, BRASS BALL	W.W. GRANGER, INC. NORTH BROOKE, IL 60062	1 INCH
3AD00201W	MANOMETER, ELECTRONIC	MITCHELL INSTRUMENT SAN MARCOS, CA 92069	0 INCH TO 20 INCH H_2O
1126-17	TESTER, PRESSURE	SPRAGUE ENGINEERING NASHUA, NH 03060	TYPE MB-3
265-480020	MANOMETER — FUEL TANK TEST	SABRELINER CORP. CHESTERFIELD, MO 63005	
265-048001	KIT — COVER PLATE	SABRELINER CORP. CHESTERFIELD, MO 63005	

Figure 21-30-604 — Leak Tester Unit — Procedure B.

NEW

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LEAK TESTER
SHOP VAC. FOR AIR SUPPLY

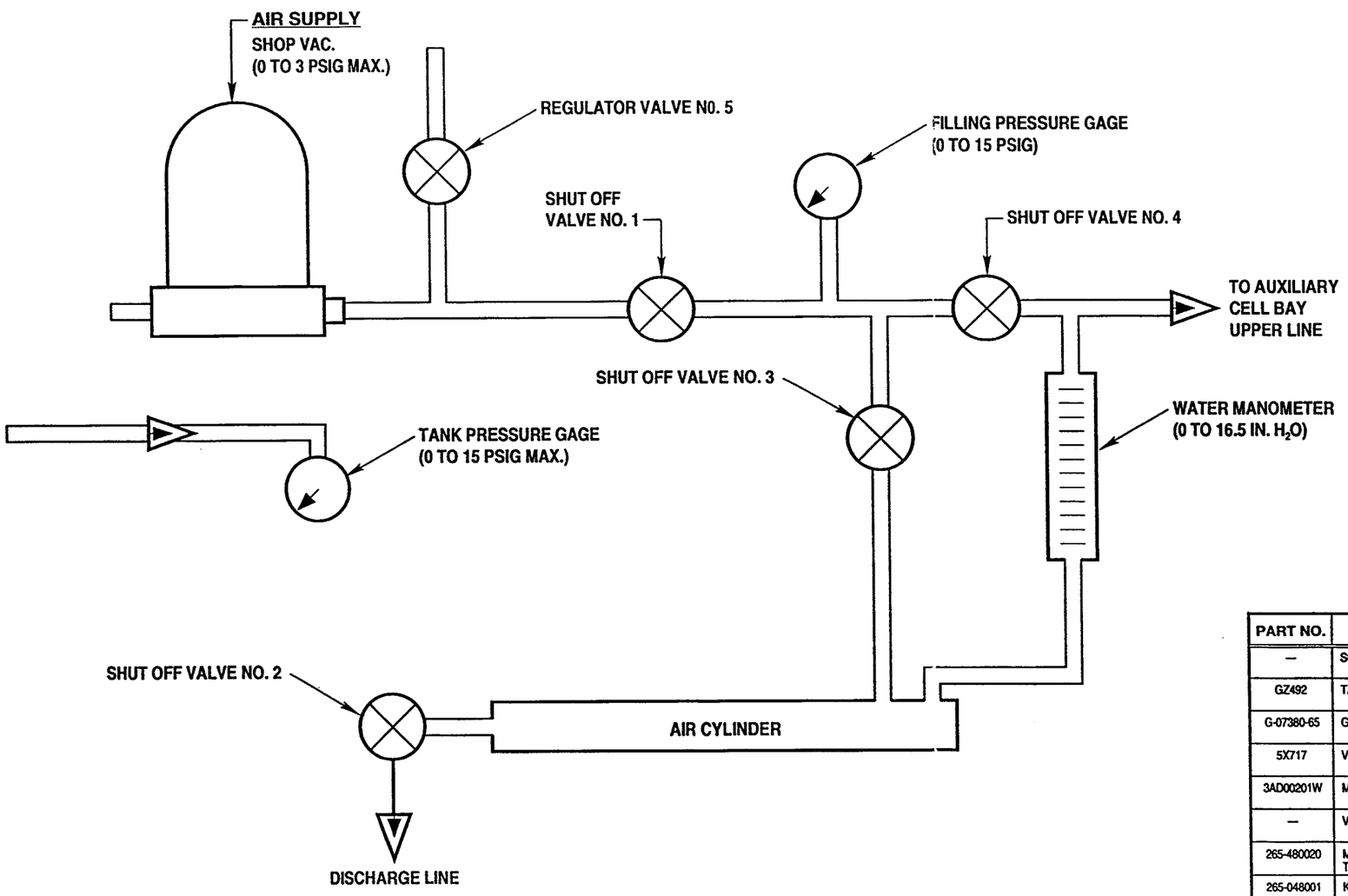

Figure 21-30-605 — Leak Tester Unit — Procedure C

PARTS LIST

PART NO.	TITLE	VENDOR	MISC.
—	SCREEN, FLAME ARRESTOR	—	200 MESH SCREEN TO BE INST'L IN OUTLET OF SHOP VACUUM
GZ492	TANK — PORTABLE AIR	W.W. GRANGER, INC. NORTH BROOKE, IL 60062	6 GAL
G-07380-65	GAUGE, PRESSURE	COLE-PARMER INTL. NILES, IL 60714	0 TO 15 PSI
5X717	VALVE, BRASS BALL	W.W. GRANGER, INC. NORTH BROOKE, IL 60062	1 INCH
3AD00201W	MANOMETER, ELECTRONIC	MITCHELL INSTRUMENT SAN MARCOS, CA 92069	0 INCH TO 20 INCH H_2O
—	VACUUM CLEANER	SEARS ROEBUCK CHICAGO, IL 60638	16 GAL, 3.0 PEAK HP
265-480020	MANOMETER — FUEL TANK TEST	SABRELINER CORP. CHESTERFIELD, MO 63005	
265-048001	KIT — COVER PLATE	SABRELINER CORP. CHESTERFIELD, MO 63005	

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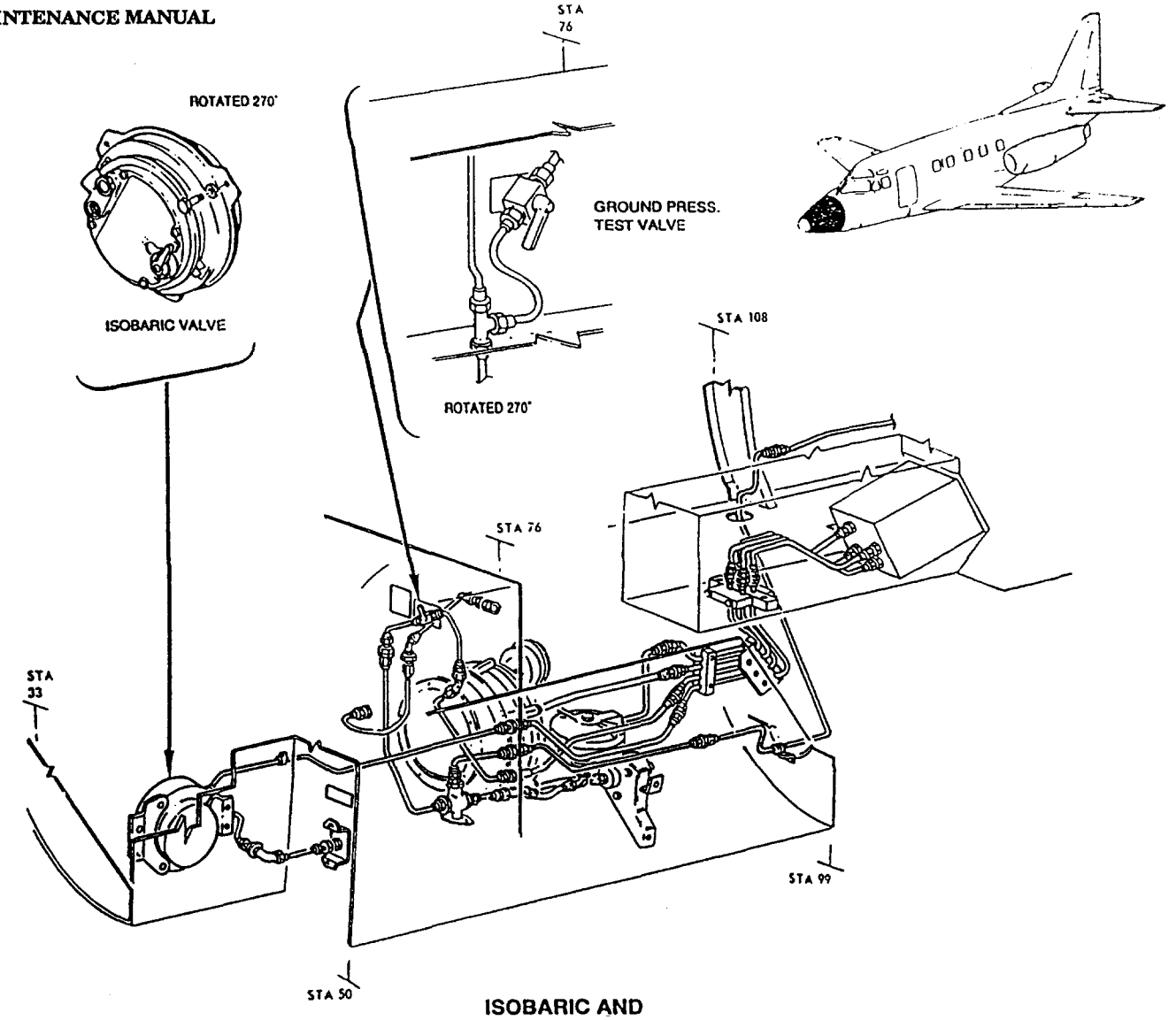
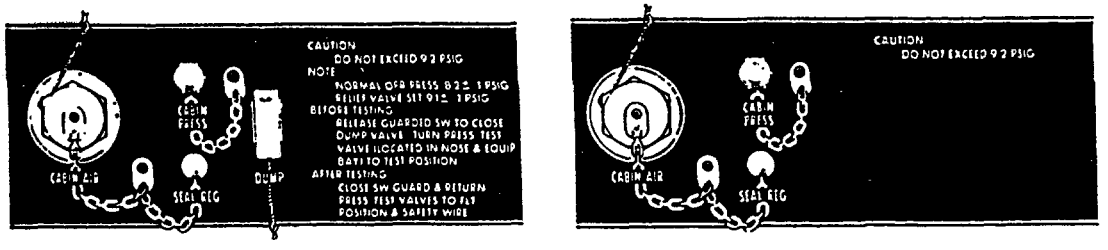


ISOBARIC AND
GROUND PRESSURIZATION TEST
VALVE LOCATIONS



GROUND PRESSURE TEST PANEL

Figure 21-30-606 — Cabin Pressurization Control System Leakage and Structural Integrity Check — Component Locations

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ELECTRONIC NOSE COMPARTMENT PRESSURIZATION

DESCRIPTION AND OPERATION

ELECTRONIC NOSE COMPARTMENT PRESSURIZATION SYSTEM — (S/N 370-1 thru -9 and 380-1 thru -23 Not Changed by S/L No. 7)

The nose compartment from station 50 forward is pressurized to maintain 29,000 (±1000) feet altitude, at altitudes above 29,000 feet. The cabin pressure regulator discharges air, plus airflow from a constant bleed duct from the cabin, supplies the necessary air to pressurize this area. When the airplane is on the ground, a blower provides cooling air to cool the electronic equipment within this area.

A controller and a regulator maintain the desired pressure within the nose compartment after the airplane climbs higher than 29,000 feet. The controller is an absolute-pressure control unit; the regulator functions as an outflow valve. When the area is pressurized, an overpressurization condition will cause a pressure relief valve to open. A low-pressure switch is provided to light the NOSE PRESS FAIL light on the caution-warning panel.

A sensing line, from the nose compartment absolute-pressure controller to the cockpit, provides pressure within the absolute-pressure controller to quickly close the nose compartment pressure (outflow) regulator when the airplane reaches 29,000 feet (2.43 psig air pressure). The nose compartment pressure regulator will continue to regulate at the 2.43 psig setting. If the area is pressurized and the pressure increases to 3.05 (±0.20) psig, the pressure relief valve will open. The pressure within the nose compartment will then drop below 2.43, actuating the pressure switch, causing the NOSE PRESS FAIL caution-warning light to light. The relief valve must be manually reset, when the airplane is on the ground, before the nose compartment will repressurize.

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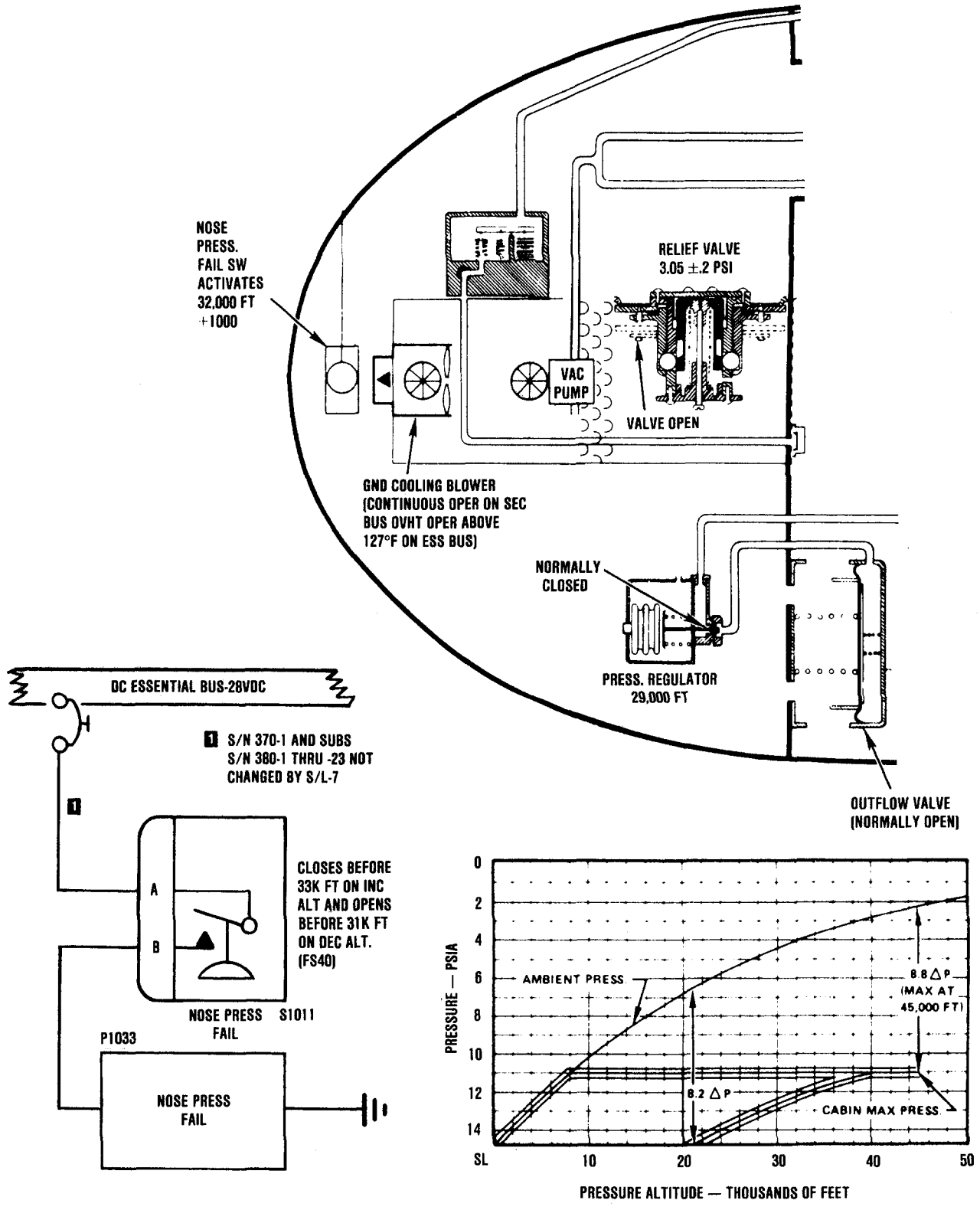

Figure 21-31-1 — Electronic Nose Compartment Pressurization System

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ELECTRONIC NOSE COMPARTMENT PRESSURIZATION - TROUBLE SHOOTING

1. Trouble Shooting Electronic Nose Compartment Pressurization System

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. ELECTRONIC COMPARTMENT WILL NOT PRESSURIZE.		
(1) Pressure relief valve open.	Manually close pressure relief valve.	
(2) Pressure relief valve damaged, causing leakage.	Check valve condition.	Repair or replace relief valve.
(3) Nose compartment pressure regulator will not close.	Check for regulator being stuck in open position. (Debris prevents regulator from closing.)	Repair or replace regulator.
	Check line from nose compartment pressure regulator to absolute pressure regulator for looseness or damage.	Tighten "B" nuts or replace line.
	Refer to procedure for checking cabin pressurization for proper operation of regulator.	
B. "NOSE PRESS FAIL" LIGHT COMES ON WHEN AIRPLANE IS BELOW 29,000 FEET.		
(1) Defective pressure switch.	See figure 21-00-2 or 21-00-2A and check continuity of electrical circuit.	Repair electrical circuit or replace pressure switch.

ELECTRONIC NOSE

COMPARTMENT PRESSURIZATION

MAINTENANCE PRACTICES

REPLACEMENT OF NOSE COMPARTMENT PRESSURE REGULATOR

Removing Nose Compartment Pressure Regulator

1. Remove access 7. (Refer to chapter 6, Figure 6-00-3.)
2. Remove all electronic equipment from nose compartment.
3. Remove sealant from head of regulator mounting screws.
4. Remove access 13. (Refer to chapter 6, Figure 6-00-3.)
5. Remove inverter.
6. Remove baffle plate aft of regulator. Remove lines from regulator.
7. Reach in nose compartment and remove all but one regulator mounting screw.
8. Have assistant hold regulator; then remove last mounting screw. Remove regulator.

Installing Nose Compartment Pressure Regulator

1. Have assistant hold regulator in place; then reach through nose compartment and install mounting screws.
2. Cover heads of mounting screws with sealant. Use general-purpose sealing compound (No. 1422GB). It may be purchased from Products Research Co., 3126 Los Feliz Blvd., Los Angeles, CA or Sabreliner Corp. A similar product may be used instead.
3. Install electronic equipment in nose compartment.
4. Install two lines on regulator.
5. Install baffle aft of regulator.
6. Install inverter.
7. Install accesses.

REPLACEMENT OF NOSE COMPARTMENT RELIEF VALVE

Removing Nose Compartment Relief Valve

1. Remove access 30. (Refer to chapter 6, Figure 6-00-3.)
2. Remove the eight attach bolts and washers.
3. Remove valve and gasket.

Installing Nose Compartment Relief Valve

1. Position valve and gasket; then install the eight bolts and washers.
2. Make sure relief valve is in closed position.
3. Install access.

Replacement of Nose Compartment Air Pressure Regulator Filter Cartridge

NOTE

The filter is installed forward of the pilot's instrument panel.

1. Gain access to filter.
2. Remove cartridge from filter housing.
3. Install new cartridge in filter housing. Ensure that cartridge is firmly seated in housing with small screened opening out.

REPLACEMENT — CABIN AIR PRESSURE REGULATOR FILTER CARTRIDGE

1. Open access door as shown in Figure 21-31-1.
2. Remove cartridge from filter housing.

NOTE

Ensure that cartridge is firmly seated in housing with small screened opening out.

3. Install new cartridge.
4. Reinstall access door removed in step 1.

ELECTRONIC NOSE COMPARTMENT PRESSURIZATION SYSTEM CHECKOUT PROCEDURE

Refer to 21-30-3, page 604.

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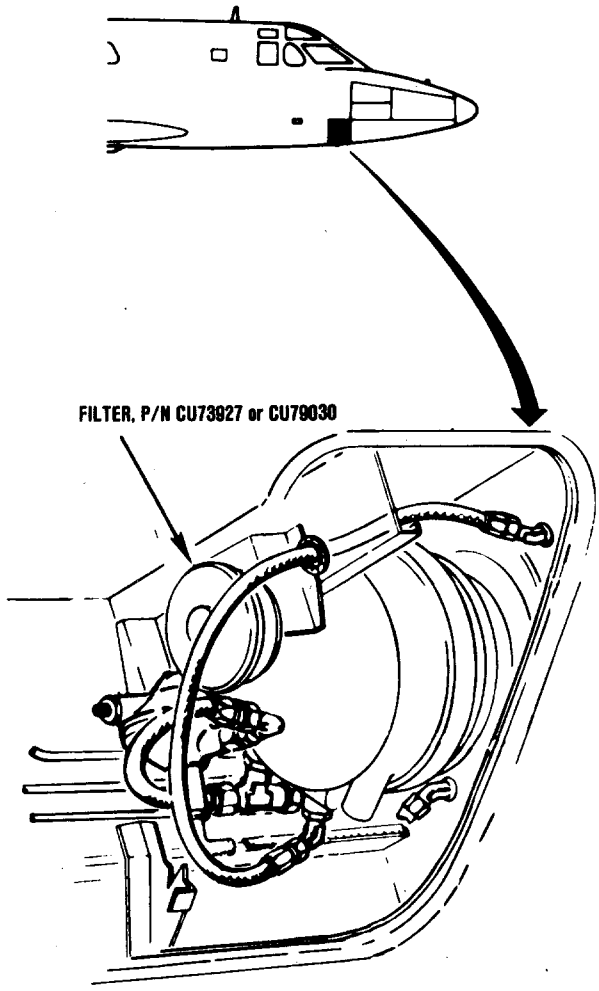

Figure 21-31-2 — Cabin Air Pressure Regulator Filter Cartridge

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INFLATABLE DOOR SEALS - DESCRIPTION AND OPERATION

1. Inflatable Door Seal System

To provide sealing of the cabin door, an inflatable rubber seal is used. Air for pressurizing this seal is obtained directly from each main engine bleed line downstream of the air extraction ports. A pressure regulator (figure 21-32-1 or 21-32-1A) and pressure reservoir are used in this system.

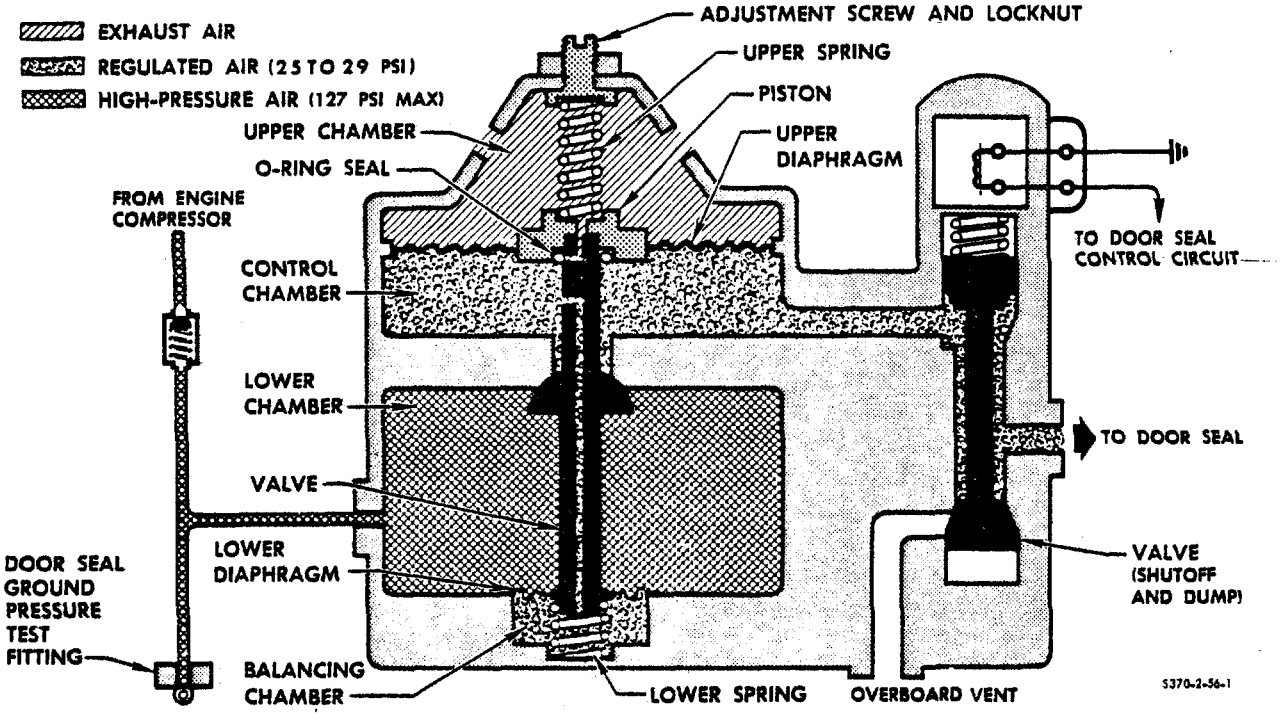
The air for this system is routed through pneumatic pressure check valves, where the two lines are joined together. The check valves prevent airflow back through the engine during ground pressurization of the seal, and prevent loss of pressure in the seal. The air then passes through another pneumatic pressure check valve to a door seal pressure regulator (figure 21-32-1 or 21-32-1A), which regulates the seal air pressure of 25-29 psig with the landing gear load switch in the air-borne position. The regulated air pressure is used to pressurize the door seal.

The door seal pressure reservoir stores the higher engine bleed-air pressure for use if engine pressure should decrease below the required 25 psig.

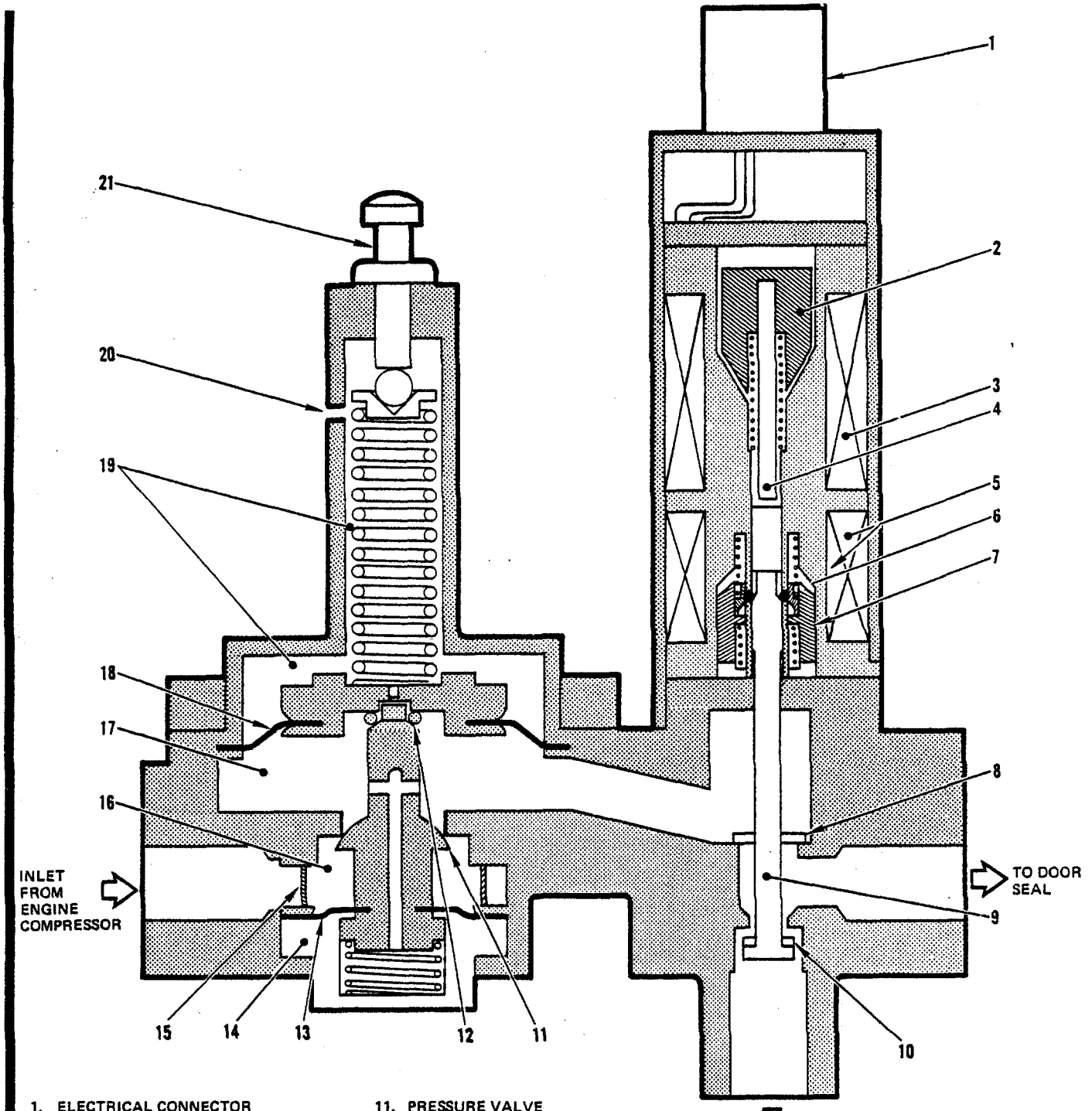
On NA-370 airplanes not changed by Optional Kit No. 5, the door seal pressure regulator incorporates a solenoid-actuated shutoff valve, which, when energized (figure 21-00-2 or 21-00-2A), allows airflow to the seal. The solenoid is energized only when the main cabin door is closed and locked and the main gear load switches are in flight position (struts extended). When the solenoid is de-energized, the shutoff valve is closed and the air is vented from the seal.

On NA-380 airplanes and NA-370 airplanes changed by Optional Kit No. 5, the door seal pressure regulator incorporates a latching type solenoid valve which, when energized, allows airflow to the seal. The solenoid is energized when the main cabin door is closed and locked and the main gear load switches are in flight position (struts extended). Electrical power is also required to close the valve which in turn vents the door seal. In the event of electrical power failure during flight, the door seal will not de-pressurize; however, after landing, the door seal may be de-pressurized by actuating the manually operated relief valve.

A manually operated relief valve, installed on the bulkhead just aft of the entrance door, relieves the door seal pressure in case of failure of the seal regulator to move to the vent position when on the ground.



Door Seal Pressure Regulator with Solenoid Valve -
Airplanes NA-370-2 thru -6, -8, -9 (Not Changed by Optional Kit No. 5)
Figure 21-32-1



1. ELECTRICAL CONNECTOR
2. VENTING PLUNGER
3. VENTING SOLENOID
4. POPPET SHAFT
5. PRESSURIZING SOLENOID
6. LATCH BALLS (4)
7. PRESSURIZING PLUNGER
8. SEAL PRESSURIZE VALVE
9. POPPET SHAFT
10. SEAL VENT VALVE
11. PRESSURE VALVE
12. RELIEF VALVE
13. BALANCE DIAPHRAM
14. BALANCING CHAMBER
15. STRAINER
16. PRESSURE CHAMBER
17. REGULATED CHAMBER
18. CONTROL DIAPHRAM
19. REFERENCE CHAMBER
20. REFERENCE AND RELIEF PORT
21. ADJUSTMENT SCREW AND LOCKNUT

S370-2-56-2

Door Seal Regulator With Latching Solenoid Valve - Airplanes NA-370-1, -7, and NA-380-1 and Later (and NA-370-2 thru -6, -8, and -9 That Are Changed by Optional Kit No. 5)
Figure 21-32-1A

INFLATABLE DOOR SEALS - TROUBLE SHOOTING

1. Trouble Shooting Inflatable Door Seal System

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. LOSS OF COMPARTMENT PRESSURIZATION WHILE IN FLIGHT.		
(1) Inflatable pressure seals deflated.	See figure 21-00-2 or 21-00-2A.	Adjust main door switch.
	Check seals for proper inflation. Refer to inflatable door seal system checkout procedure.	Replace seal. Refer to procedure for replacing inflatable door seal.
(2) Faulty door seal pressure regulator solenoid.*	Check electrical circuit. (See figure 21-00-2 or 21-00-2A.)	Repair or replace faulty components in electrical circuit.
		Repair or replace regulator.
(3) Faulty main door switch.	Check electrical circuit. (See figure 21-00-2 or 21-00-2A and refer to Wiring Diagram Manual.)	Repair or replace faulty components in electrical circuit.
		Replace switch.
		Adjust switch for proper actuation by main door.

*Airplanes NA-370-2 through -6, -8 and -9

INFLATABLE DOOR SEALS

MAINTENANCE PRACTICES

REPLACEMENT OF DOOR SEAL PRESSURE REGULATOR

Removal:

1. Open accesses 1 and 55 (refer to chapter 6).
2. Disconnect electrical wires and pressure control lines from pressure regulator.
3. Remove mounting bolts from pressure regulator and remove from mounting bracket.
4. Cover openings on pressure regulator and pressure control lines.

Installation:

1. Remove protective covering from pressure regulator and pressure lines.
2. Place pressure regulator in proper mounting position; then install mounting bolts.
3. Connect pressure lines to regulator; then torque B nuts to 40-65 inch-pounds.
4. Connect electrical wires leading to solenoid; then install rubber cap.
5. Close accesses.

ADJUSTMENT OF CABIN DOOR INFLATABLE SEAL SWITCH

1. Close and lock cabin door.
2. Make sure door locking pin is contacting switch; then adjust limit switch to allow 1/16-inch overtravel before switch is bottomed.

INFLATABLE DOOR SEAL SYSTEM CHECK-OUT PROCEDURE

Pressure and Leakage Check of Inflatable Door Seal

1. Open accesses 1 and 55 (refer to chapter 6).
2. Disconnect door seal pressure line at outlet port of door seal pressure regulator. Cap regulator outlet port.
3. Install test line to disconnected door seal pressure line and connect a source of dry pressurized air, a pressure regulator, a shutoff valve and a gauge (0-60 psig acurate to 1% of full scale, or equivalent). Install gauge between shutoff valve and door seal line.
4. Close and latch main cabin door.

CAUTION

Do not allow pressure to door seal to exceed 29 psig during this check.

5. Pressurize door seal line to 28 ±1 psig. Allow at least two minutes for air pressure to stabilize, then close shutoff valve. Check that gauge indication is 28 ±1 psig.
6. After a 20-minute period, check that gauge indication is not less than 21 psig. If less than 21 psig, check seal pressure line, manual relief valve, test equipment, and door seal for source of leak. Repair as necessary and repeat steps 5 and 6.
7. Relieve pressure and remove pressure test equipment.
8. Reconnect seal pressure line to outlet of door seal pressure regulator. Proceed to next portion of check.

Operational Check of Door Seal — S/N 370-2 Thru -6, -8 and -9

1. Connect external 28 VDC power to airplane.
2. Connect an air pressure source with pressure regulator and gauge (0-200 psig or equivalent) to SEAL REG fitting of cabin ground pressure test panel.

NOTE

On airplanes equipped with an APU but not incorporating SOK No. 7, it is necessary to disconnect the APU sensing line and cap the tee fitting on line leading from SEAL REG fitting.

3. Pressurize system to 82-115 psig. Do not exceed 115 psig.
4. Check that LANDING GEAR - POS and DOOR SEAL & LOCK IND circuit breakers are engaged.
5. Position both main landing gear switches (on gear scissors) at an unloaded (gear extended) condition.
6. Open main cabin door. Check that DOOR OPEN amber caution light comes on, and that the door seal is deflated.
7. Manually depress and hold cabin door locked (seal pressurizing) switch. Check that door seal inflates and that DOOR OPEN caution light goes out.

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MAINTENANCE PRACTICES (Continued)

INFLATABLE DOOR SEAL SYSTEM CHECK-OUT PROCEDURE (Cont'd)

8. Release, then depress and hold door switch. Seal should deflate then reinflate.
9. Release switch then close and latch door. Check that seal inflates and that DOOR OPEN caution light goes off.
10. In aft fuselage, check the following for air leakage: all airplane lines connected to SEAL REG fitting of cabin ground pressure test panel, related check valves and fittings, door seal regulator, water separator delta P sensor (disregard air flowing into cold air duct), cabin flood flow valve, emergency bleed air overpressure switch, door seal and hydraulic air reservoir tanks, hydraulic system air pressure regulator and filter, dehydrator, and air hydraulic reservoir tank drain valve. (Disregard air flowing from regulator vents or air flowing into deicer boot system.) Repair air leaks as necessary.
11. On main landing gear, place both load switches at loaded position and install cotter pin. Check that main cabin door seal has deflated. (DOOR OPEN caution light should remain out.)
12. Disconnect test equipment and reconnect APU sensing line (if applicable).
13. Open main cabin door and close accesses 1 and 55.

Operational Check of Door Seal — S/N 370-1, -7, and 380-1 and Subs

1. Connect external 28 VDC power to airplane and disengage (pull out) LH and RH PITOT HTR, STALL WARNING INBOARD and OUTBOARD VANE HEATER, and EPR PITOT & STALL WARN HTR CONTROL circuit breakers.
2. Connect an air pressure source with pressure regulator and gauge (0-200 psig or equivalent) to SEAL REG fitting of cabin ground pressure test panel.

NOTE

On S/N 370-1, -7, and 380-1 thru -38 that do not incorporate SOK No. 7, it is necessary to disconnect the APU sensing line and cap the tee fitting on line leading from SEAL REG fitting.

3. Pressurize system to 82-115 psig. Do not exceed 115 psig.
4. Check that LANDING GEAR - POS and DOOR SEAL & LOCK IND circuit breakers are engaged.
5. Position both main landing gear switches (on gear scissors) at an unloaded (gear extended) condition.
6. Open main cabin door. Check that DOOR OPEN amber caution light comes on, and that door seal is deflated.
7. Manually depress and hold cabin door locked (seal pressurizing) switch. Check that door seal inflates and that DOOR OPEN light goes out.
8. Release door switch. Seal should deflate.
9. Depress and hold door switch. Seal should reinflate.
10. While holding switch, disengage DOOR SEAL & LOCK IND circuit breaker (LH outboard overhead circuit breaker panel).
11. Release door switch. Check that door seal does not deflate.
12. Engage DOOR SEAL & LOCK IND circuit breaker. Check that door seal deflates (DOOR OPEN light comes on.)
13. Close and latch main cabin door. Check that seal inflates and that DOOR OPEN caution light goes out.
14. In aft fuselage, check the following for air leakage: all airplane lines connected to SEAL REG fitting of ground pressure test panel, related check valves and fittings, door seal regulator, cabin flood flow valve, emergency bleed air overpressure switch, door seal and hydraulic air reservoir tanks, hydraulic system air pressure regulation and filter, dehydrator, and hydraulic air reservoir tank drain valve. (Disregard air flowing from regulator vents.) Repair air leaks as necessary.
15. On main landing gear, place both load switches at loaded position and install cotter pin.
16. *Check that main cabin door seal has deflated. (DOOR* OPEN caution light should still be out.)
17. Disconnect test equipment and reconnect APU sensing line (if applicable.)
18. Remove external electrical power and engage circuit breakers (refer to step 15.)
19. Open main cabin door and close accesses 1 and 55.

REPLACING INFLATABLE DOOR SEAL

Refer to chapter 52 for inflatable door seal replacement.

REPLACEMENT OF ENTRY DOOR SEAL AND SENSING LINES FILTER

1. Open aft fuselage access doors on aft fuselage lower centerline.
2. Disconnect dehydrator lines from each end of filter and remove filter from airplane.
3. Remove safety wire and unscrew filter head (outlet end) from filter body.

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MAINTENANCE PRACTICES (Continued)

REPLACEMENT OF ENTRY DOOR SEAL AND SENSING LINES FILTER (Cont'd)

4. Remove filter element from filter case. Discard filter element.
5. Install new filter element (Purolator P/N 37820) in filter body.
6. Install new gasket (P/N 54694-Purolator) and coat threads of filter head with a solid film lubricant (MIL-L-8937C). Install filter head to filter body and torque 45-55 ft-lbs; then safety with lockwire.
7. Install filter in dehydrator lines with flow arrow on filter body pointing downstream from dehydrator unit.
8. Close aft fuselage access doors.

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COOLING

DESCRIPTION AND OPERATION

COOLING-AIR SYSTEM — (S/N 370-2 thru -6, -8 and -9)

Bleed-air from either or both engines passes through the flow limiter valve and is then divided and routed in two directions. A portion of the hot bleed-air is ducted to the hot-air mixing valve; the remainder is ducted to the refrigeration unit for cooling.

The refrigeration unit consists of a heat exchanger and an expansion cooling turbine. Initial cooling is accomplished in the heat exchanger; final cooling is accomplished within the expansion cooling turbine. Ram air-cooled tubes within the heat exchanger provide a path for the initial cooling of the hot bleed-air. The ram air comes from the ram air inlet duct at the leading edge of the dorsal fin. To increase the ram air flow, an axial-flow fan (driven by the cooling turbine) accelerates ram air flow through the heat exchanger to an overboard duct.

Engine bleed-air, partially cooled by the heat exchanger, is normally routed to the expansion turbine for final cooling. However, if air pressure within the duct downstream from the turbine drops to 15 psi or lower, the cabin flood flow valve will open. When the cabin flood flow valve is pneumatically actuated to the open position, the partially cooled air from the heat exchanger will bypass the expansion turbine and the water separator (high-pressure drop areas) and automatically supply an adequate supply of air to the cabin.

Partially cooled air is ducted to the expansion cooling turbine for final cooling. This is accomplished when partially cooled air drives the cooling turbine, expends heat energy, and expands to its final cooled condition. The cooled air is then ducted to a water separator, where the excess moisture is extracted. The resultant dry, cool air is then ducted to the air distribution system.

The water separator delta P sensor prevents freezing conditions within the water separator. The sensor senses the increase in pressure drop across the water separator when ice forms within the separator. When this condition exists, the sensor pneumatically actuates the water separator anti-ice valve to an open condition. The water separator anti-ice valve is positioned in a small branch of the hot bleed-air duct upstream from the heat exchanger. Opening this anti-ice valve provides sufficient hot air to melt the ice within the separator.

WATER SEPARATOR — (S/N 370-2 thru -6, -8 and -9)

The water separator is installed downstream from the cooling turbine to remove moisture from the system air supply. Moisture is collected by the separator filter and drained overboard through a drain line attached to the bottom of the separator. If icing occurs on the filter, a delta P (pressure differential) sensor detects the pressure drop across the separator and automatically opens the water separator (pneumatic) anti-ice valve, allowing hot engine compressor bleed-air to flow through the water separator until the ice is cleared from the filter. The water separator may be reached through the lower fuselage rear section equipment access doors.

WATER SEPARATOR DELTA P (PRESSURE DIFFERENTIAL) SENSOR — (S/N 370-2 thru -6, -8 and -9)

The water separator delta P sensor is mounted above the water separator and has pneumatic sensing lines connected across the water separator. These sensing lines and delta P sensor operate the water separator anti-ice valve to allow hot engine compressor bleed-air to flow through the water separator if ice has collected on the water separator filter. When icing condition has been cleared from the water separator filter, the water separator anti-ice valve closes.

An in-line filter is installed in the pressure supply line to the water separator delta P sensor. This filter will reduce the possibility of contamination entering and restricting the delta P sensor orifice. Restriction of the sensor orifice could result in insufficient pressure to actuate the water separator anti-ice valve in case of water separator icing and could cause rapid oscillation and surging of cabin pressurization.

WATER SEPARATOR ANTI-ICE VALVE — (S/N 370-2 thru -6, -8 and -9)

The water separator anti-ice valve hot-air line is connected to the turbine cooled-air duct on upstream side of the water separator. The water separator anti-ice valve is opened to allow hot engine compressor bleed-air to flow through the water separator when icing conditions have clogged the water separator filter. The water separator anti-ice valve is pneumatically opened or closed by operation of the water separator delta P sensor, depending upon possible pressure drop across the water separator.

COOLING-AIR SYSTEM — (S/N 370-1 and -7, 380-1 and Subs)

Bleed-air from either or both engines passes through the flow limiter valve and is then divided and routed in three directions. A portion of the hot bleed-air is ducted to the cockpit and the cabin hot-air mixing valves; the remainder is ducted to the primary section of the refrigeration unit for cooling.

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DESCRIPTION AND OPERATION (Continued)

COOLING-AIR SYSTEM — (S/N 370-1 and -7, 380-1 and Subs) (Cont'd)

The refrigeration unit consists of a heat exchanger, three-wheel cooling turbine, 35°F temperature control valve, 35°F temperature control/sensor, and a bleed air pressure regulator. The cooling turbine consists of a vane-axial fan, a centrifugal compressor, and a radial expansion turbine. These three units are mounted on a common shaft.

The heat exchanger is a single core unit in which the core is split into two sections to obtain a compact primary and secondary heat exchanger. Approximately one-third of the ambient cooling air flows through the primary heat exchanger section and two-thirds through the secondary heat exchanger section.

During normal operation, bleed-air from either engine passes through the bleed-air shutoff valve, through the flow control valve, and to the refrigeration unit. Bleed-air entering the refrigeration unit passes through the primary section of the heat exchanger where it is cooled and then through the pressure regulator which controls bleed-air pressure to a nominal value of 25 psig. The cooled air then enters the compressor component of the three-wheel cooling turbine and undergoes an increase of pressure and temperature in the compression process. Upon leaving the compressor the air flows through the secondary section of the heat exchanger where it is cooled to nearly the temperature of the ambient cooling air. This air is then expanded through the turbine component of the three-wheel turbine. This expansion process not only reduces the bleed-air temperature but produces shaft power to drive the compressor and the fan. About 70 percent of the power produced by the turbine is absorbed by the compressor and the remainder by the fan, the main function of which is to provide ambient cooling air for the heat exchanger during ground operation. A door in the ram air inlet duct is opened during ground operation to provide a lower resistance path than the ram air scoop inlet.

A segmented, circumferential flapper type check valve installed between the ram air outlet connection of the refrigeration package and the fan serves to prevent fan air recirculation during ground operation. Also, during certain flight conditions when the fan tends to impede ram airflow, the check valve allows a lower resistance flow path around the fan, thereby preventing undesirable decrease of ram airflow.

If the bleed-air is cooled below the dew point during the expansion process in the turbine, water vapor in the air will condense out as small droplets (fog), or, if the temperature is below 32°F, ice crystals will form in the airstream. The cold turbine discharge air from the refrigeration unit is passed through a water separator for removal of entrained moisture. Protection against ice formation in the water separator and downstream ducting during all modes of operation is achieved by the action or the 35°F temperature control/sensor and the 35°F temperature control valve.

The latent heat of the condensed water collected in the water separator is utilized to increase the heat sink capability of the heat exchanger cooling air. An aspirator which uses a small quantity of bleed-air from the heat exchanger outlet pan functions to spray this water into the secondary heat exchanger section of the heat exchanger core, thereby enhancing the heat sink capacity of the cooling air by evaporation.

The 35°F temperature control/sensor functions to modulate the electro-pneumatic 35°F temperature control valve to control the flow of compressor inlet air that is mixed with turbine discharge air to maintain a minimum temperature of 35°F into the water separator.

35°F TEMPERATURE CONTROL VALVE — (S/N 370-1, -7, 380-1 and Subs)

The 35°F temperature control valve, is a normally closed, electro-pneumatically actuated right angle poppet type modulating and shutoff valve. The valve controls the refrigeration unit cold air outlet to a minimum value of 35°F (nominal) by modulating the flow of compressor inlet air to the anti-ice muff at the turbine discharge. Valve position is controlled by a torque motor which varies the valve actuator pressure in response to electrical signals from the 35°F temperature control/sensor.

PRESSURE REGULATOR — (S/N 370-1, -7, 380-1 and Subs)

The pressure regulator is a pneumatically-actuated spring-loaded open butterfly type valve which functions to control the compressor inlet pressure to 25 psig.

35°F TEMPERATURE CONTROL/SENSOR — (S/N 370-1, -7, 380-1 and Subs)

This solid state controller, which in conjunction with the 35°F temperature control valve, consists of a temperature sensor and an electronic control packaged as an integral assembly. The unit is boss mounted to the water separator outlet duct with the sensor probe immersed in the airflow.

The sensor probe contains two glass-bead type thermistor sensing elements. The thermistors are mounted in the stainless steel probe using a ceramic potting compound. Windows in the end of the probe expose the tips of the thermistors to the duct airflow.

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DESCRIPTION AND OPERATION (Continued)

35°F TEMPERATURE CONTROL/SENSOR — (S/N 370-1, -7, 380-1 and Subs) (Cont'd)

In the control circuit, the sensing elements are connected into a voltage divider network whose ouptut is fed to an integrated-circuit differential amplifier. The amplifier output is proportional to the difference between 35°F and sensed temperatures.

The temperature error signal is fed through a dynamic compensation network to drive the output stage. Since the torque motor of the 35°F temperature control valve is a current-operated device, the control circuit employs a current output stage. Output current is proportional to temperature error, so that the 35°F temperature control valve is modulated from closed to full open over a proportional band of sensed temperature about the 35°F control point.

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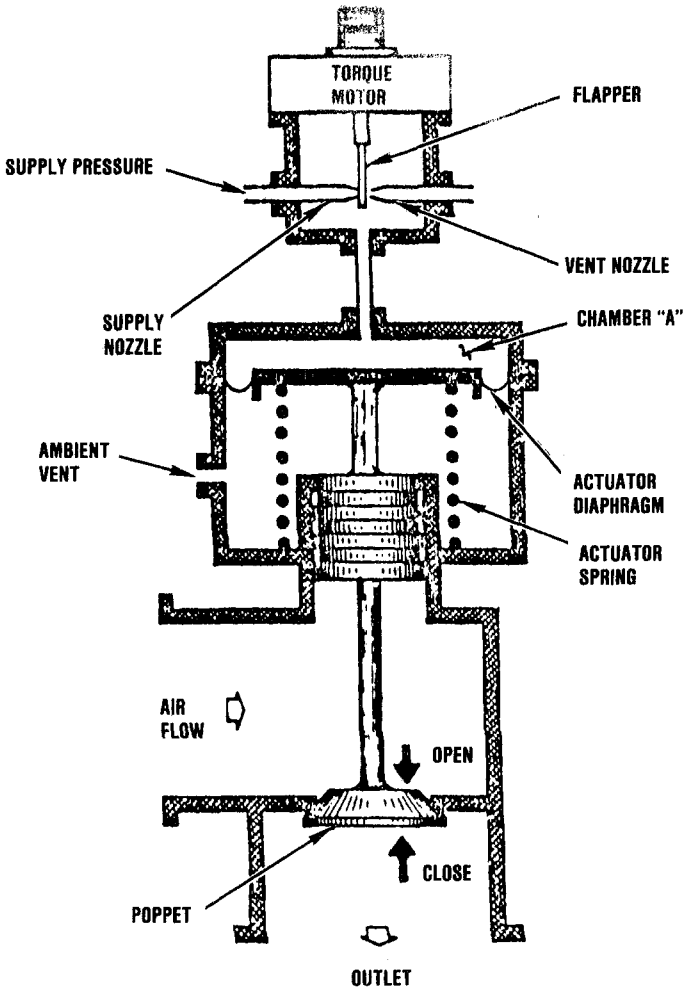

Description:

This unit is a normally closed electro-pneumatically actuated shutoff and modulating valve. The poppet is inlet pressure balanced and spring loaded in the closed position. The valve is opened by applying pressure to Chamber "A" and overcoming the spring load.

Supply pressure is ported to the torque motor supply nozzle and with zero electrical signal applied, this nozzle is closed. Chamber "A" is then vented through the torque motor vent nozzle. As the electrical signal to the torque motor is increased, the flapper moves to open the supply nozzle and close the vent nozzle raising the pressure in Chamber "A" and opening the valve. For a given value of input signal there is a predetermined valve position.

Figure 21-50-1 — 35°F Temperature Control Valve — S/N 370-1, -7, 380-1 and Subs

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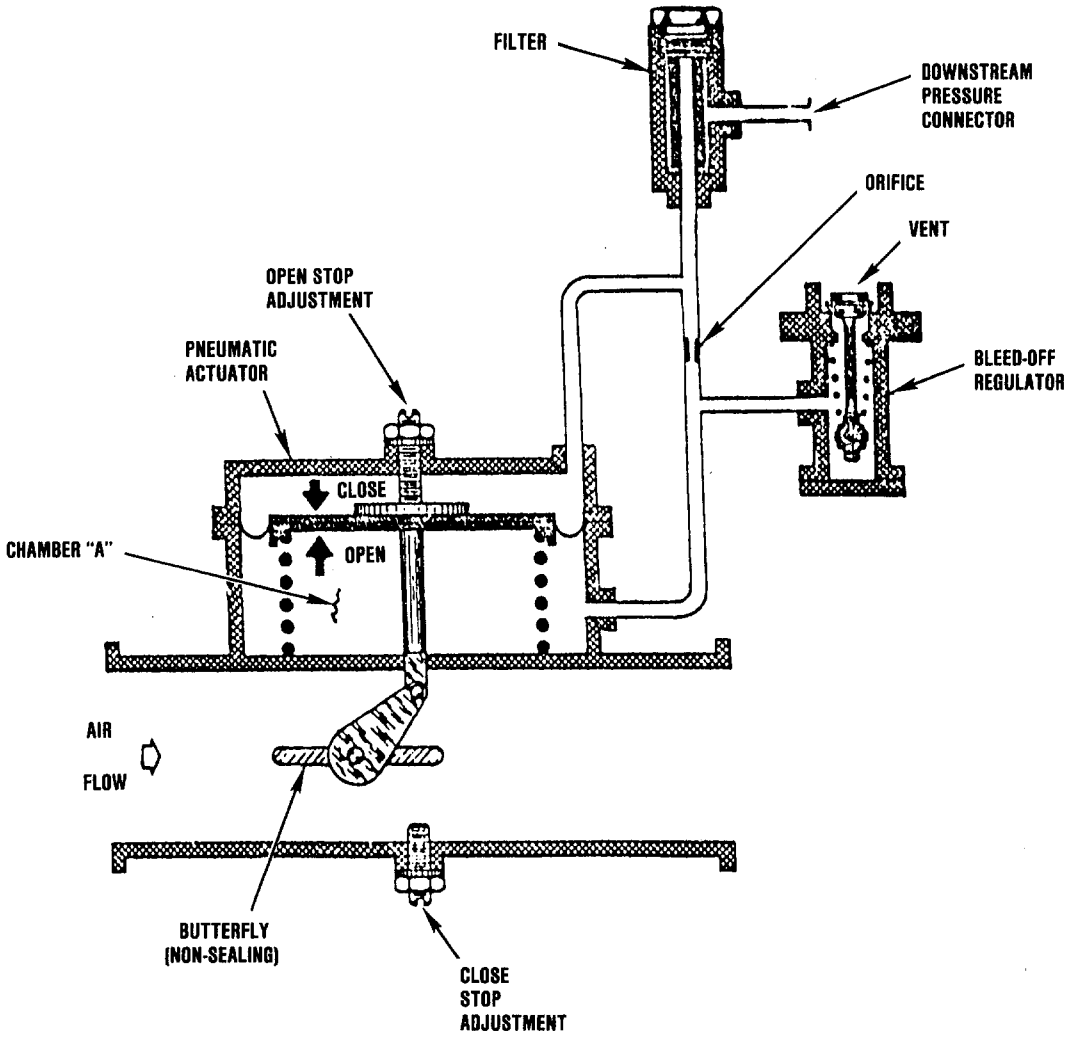

Description:

This unit is a spring loaded open, butterfly type pressure regulating valve.

The bleed off regulator maintains a constant pressure downstream of the orifice. This pressure is applied to Chamber "A" and serves to reduce the required spring load.

When the downstream pressure increases and reaches the predetermined valve, it overcomes the forces which are the sum of the spring force and the Chamber "A" pressure force. The butterfly closes to maintain the desired downstream regulated pressure.

Figure 21-50-2 — Pressure Regulator — S/N 370-1, -7, 380-1 and Subs

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COOLING - TROUBLE SHOOTING

1. Trouble Shooting Cooling Air System

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. NO COOLED AIRFLOW TO CABIN.		
(1) Faulty cooling turbine or clogged water separator; air not bypassing through separator.	Check for abnormal noise from refrigeration unit cooling turbine, or check for clogged water separator filter. Also, check for plugged separator overboard drain line.	If cooling turbine failure is evident, replace turbine or refrigeration unit.
(2) Faulty water separator anti-ice valve* or 35°F temperature control valve† not clearing ice from water separator.	Check for loose connections on pressure-sensing lines at water separator anti-ice valve and delta P sensor.	
(3) Cooling turbine frozen because of lack of oil to bearings.	Check for oil mist in cooling turbine exhaust duct denoting turbine seal damage.	Replace cooling turbine or refrigeration unit.
(4) 35°F temperature control/sensor† inoperative.	Remove sensor electrical plug and check for 28 VDC at pin "D" of electrical connector.	If 28 VDC is present at electrical connector, replace sensor.
B. HUMIDITY TOO HIGH (HIGH MOISTURE CONTENT) IN COOLED AIRFLOW TO CABIN.		
(1) Water separator filter clogged, allowing cooled airflow to bypass filter, or plugged separator drain line.	Remove separator, and check condition of filter. Check separator overboard drain line for obstruction.	Clean or replace separator filter and clear overboard drain line.

*Aircraft NA-370-2 through -6, -8 and -9
†Aircraft NA-370-1, -7 and NA-380-1 and later

COOLING

MAINTENANCE PRACTICES

REPLACEMENT OF REFRIGERATION UNIT

Removing Refrigeration Unit

NOTE

The refrigeration unit consists of an air-to-air heat exchanger and cooling turbine. The cooling turbine may be removed separately by removing the turbine attaching bolts and disconnecting the turbine hot-engine bleed-air duct expansion joint and cooled-air outlet duct.

1. Open fuselage accesses 1 and 51 for S/N 370 airplanes, and 1 and 59 for S/N 380 airplanes. (Refer to Chapter 6.)

NOTE

The refrigeration unit is self-supported by the attaching bracket, and does not require support while ducts are being disconnected.

2. At refrigeration unit, disconnect all high-pressure duct clamps, and slide clamps back on ducts.
3. Disconnect all intermediate pressure and ram air ducting clamps, and slide back on ducting; then, to prepare for removal of refrigeration unit, work the seal off duct joints.
4. Disconnect water drain line from bottom of refrigeration unit.
5. On airplanes S/N 380-1 and Subs, disconnect line from water separator.
6. From outboard side of refrigeration unit lower attach bracket, loosen and remove two securing bolts. Then lift refrigeration unit upward to free the upper attaching hooks. Do not damage ducting flanges when displacing the refrigeration unit.
7. Cover all refrigeration unit openings, ducting, and drain line, to prevent entry of foreign material.

Installing Refrigeration Unit

1. Remove protective covering from refrigeration unit, ducting, and drain line.
2. Carefully lift refrigeration unit into proper mounting position, taking care not to damage attaching ducting flanges. Engage upper mounting hooks only. This leaves the unit free to be shifted while the unit ducting is being attached.

CAUTION

Do not disengage refrigeration unit upper support hooks while shifting unit to align the ducting. The refrigeration unit securing bolts on lower support bracket may be loosely installed if desired.

3. After ducting is properly aligned, install all unit duct connectors and ground straps.
4. Place gaskets between flanges, and install Marman clamps. Torque clamp bolts as follows:

Clamp P/N	Steel Flanges	Aluminum Flanges
51394H100	35-40 in-lbs	—
51394H175	50-55 in-lbs	—
51394H200	55-60 in-lbs	45-50 in-lbs
51394H250	65-70 in-lbs	50-55 in-lbs

5. Torque hose type clamps on fiberglas ducts to 25 inch-pounds.
6. Tighten bolts on lower mounting bracket, and safety with lockwire.
7. Connect water drain line and, on S/N 380-1 and Subs, connect water separator drain line to water injector.
8. Close accesses.

REPLACEMENT OF REFRIGERATION UNIT COOLING TURBINE — (S/N 370-2 thru -6, -8 and -9)

Removing Refrigeration Unit Cooling Turbine

1. Disconnect hot-air line to turbine by removing clamp from expansion joint; then work clamp spring up on duct flange, away from turbine inlet duct. This allows duct clearance when turbine is pulled forward out of the heat exchanger unit.
2. After disconnecting clamp and seal from turbine cooled-air outlet duct, disconnect two lines attached to this large cooled-air outlet duct; then loosen clamp on this same duct (second clamp from turbine), allowing duct to be rotated around elbow part of duct. The turbine can then be pulled forward later in this removal procedure.
3. Remove lockwire and turbine attaching bolts around periphery of turbine.
4. Remove lower refrigeration unit support bracket, allowing turbine to be pulled forward out of heat exchanger.
5. Pull turbine forward and out of heat exchanger. Retain sealing O-ring gasket for reinstallation of turbine.

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MAINTENANCE PRACTICES (Continued)

REPLACEMENT OF REFRIGERATION UNIT COOLING TURBINE — (S/N 370-2 thru -6, -8 and -9) (Cont'd)

Installing Refrigeration Unit Cooling Turbine

1. Place O-ring seal in groove on heat exchanger. A small amount of petrolatum may be used to retain O-ring during installation.
2. Lubricate O-ring on air guide and lift turbine into heat exchanger housing. Align turbine inlet fitting with duct.
3. Install bracket to bottom of heat exchanger. Install three screws through turbine flange and install two AN5-5A bolts from airplane bracket into turbine bracket.
4. Install remainder of turbine attach screws, and safety all screws with lockwire. Install turbine hot-air inlet duct expansion coupling. This is done by working expansion joint spring down from duct flange. (The expansion joint spring was worked up on duct flange during step 1. of the removal procedure.) Install the two screws securing expansion joint clamp.
5. Align turbine cooled-air ducting; then install silicone couplings, ground straps, and Marman clamps. Torque clamps to 25-30 inch-pounds. Reconnect line and duct disconnected during removal procedure. Torque clamps and "B" nut.

LUBRICATION OF REFRIGERATION UNIT COOLING TURBINE — (S/N 370-2 thru -6, -8 and -9)

NOTE

At intervals (refer to Chapter 5), it is necessary to drain and replace the oil in the refrigeration unit cooling turbine. Removal of the cooling turbine is not necessary.

1. Provide a small receptacle to catch oil.
2. Remove drain plug at bottom of turbine housing and filler plug on top of housing to allow any water and any other residue to drain. Reinstall drain plug finger-tight.
3. Fill turbine with approximately 100 cubic centimeters (3.4 fluid ounces) of aircraft instrument lubricating oil. (Refer to Chapter 12 for "Table of Approved Refrigeration Unit Cooling Turbine Oils".)
4. Replace filler plug finger-tight and permit oil to remain in turbine for approximately 15 minutes.
5. Remove drain plug and insert a clean piece of wire (0.040- to 0.060- inch diameter) approximately 1/2 inch into sump cotton to facilitate drainage. Remove filler plug. Allow approximately 20 minutes to properly drain off excess oil. The cotton will retain the proper amount of oil.
6. Using S8316A2 gaskets, replace and tighten drain and filler plugs. Safety both drain and filler plugs with lockwire. (AiResearch gasket S8316A2 is equivalent to AN920-2 or MS29512-2.)

LUBRICATION OF REFRIGERATION UNIT COOLING TURBINE — (S/N 370-1, -7, 380-1 and Subs)

NOTES

- ***Refer to Chapter 5 for intervals of lubrication.***
- ***The oil level may be checked by using the filler plug dip stick.***

1. If turbine oil is to be drained, provide a small receptacle under drain plug to catch oil.
2. Remove drain plug from turbine and allow oil to drain.
3. Using gasket (NAS1595-2), install and tighten drain plug and safety wire.
4. Remove filler plug.
5. Fill turbine with approximately 100 cubic centimeters (3.4 fluid ounces) MIL-L-23699 oil (Mobil Jet II, type II) or equivalent (see Chapter 12).

NOTE

Although locking pin cannot fall out when the dipstick plug is installed in the turbine, it can fall out of the plug during oil servicing while it hangs from the retention chain.

6. Check oil level using dipstick. Add additional oil if oil level is below line on dipstick.

CAUTION

The dipstick plug can be inserted into the turbine with the locking pin missing. This condition can allow the dipstick to come loose, during turbine operation, and will cause a loss of oil resulting in turbine failure.

NOTE

If dipstick plug lockpin is loose or missing, P/N MS9015-08 plug, and P/N NAS1595-08 packing may be installed. Thereafter, at each oil service interval, the oil level should be maintained to the top of the casting hole.

7. Using gasket (NAS1593-012), replace oil filler plug.
8. Check assembly for oil leaks.

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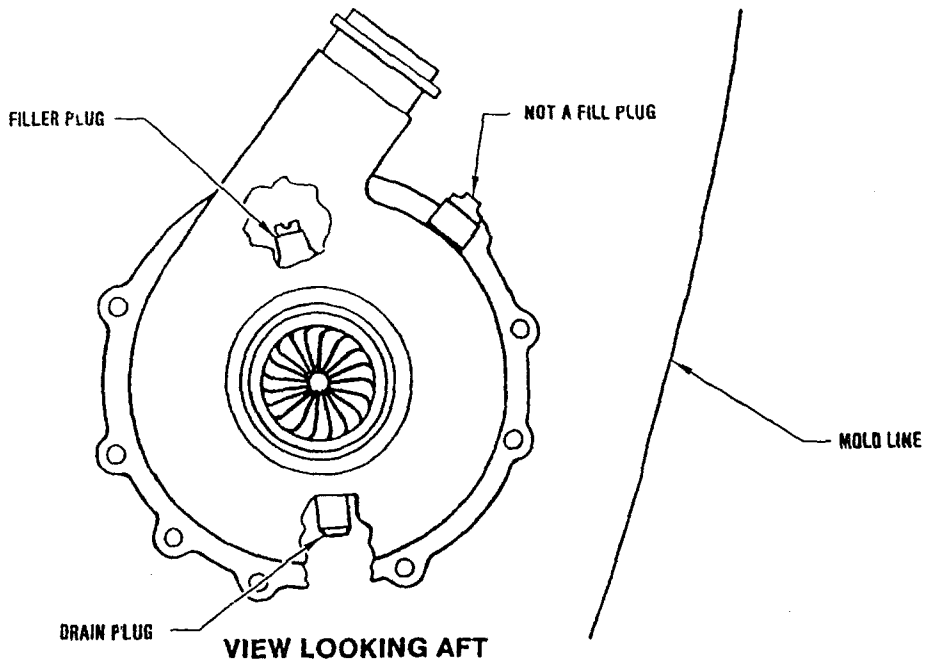
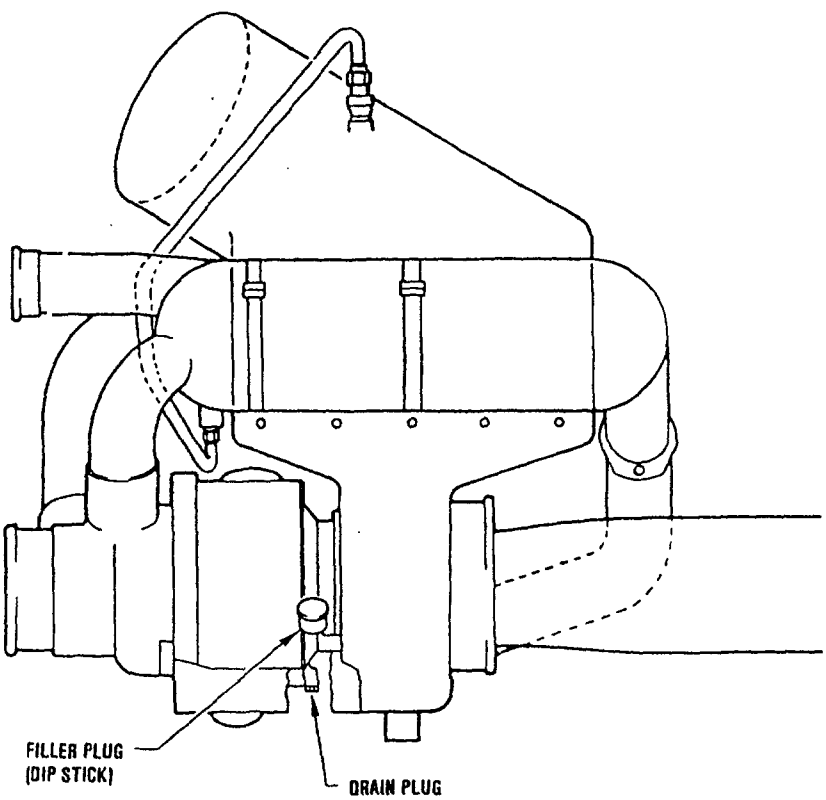


REFRIGERATION TURBINE — AIRPLANES S/N 370-2 THRU -6, -8 AND -9



REFRIGERATION TURBINE — AIRPLANES S/N 370-1, -7, S/N 380-1 AND SUBS

S370-2-53-9C

Figure 21-50-3 — Lubrication of Refrigeration Unit Cooling Turbine

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MAINTENANCE PRACTICES (Continued)

REPLACEMENT OF WATER SEPARATOR

Removing Water Separator

1. Open accesses 1 and 55. (Refer to chapter 6, figure 6-00-3 or 6-00-3A.)
2. Loosen clamps at both ends of separators and slide clamps back on ducting. Then, to allow removal of separator, work the seal off duct joint.
3. Disconnect water drain line from bottom of separator.
4. While supporting separator, loosen large clamps around body of separator and slide clamps off over ends of separator supporting bracket. Then lift separator from mounting position.
5. To prevent entry of foreign material, cover separator, duct, and drain line openings.

Installing Water Separator

1. Remove protective coverings from separator and duct openings.
2. Place separator in position by engaging right coupling first and then left lower duct coupling.
3. Install support clamps loosely around separator body and support brackets.
4. Shift separator until it is in alignment with ducts. Install ground straps and connector clamps. Torque clamps to 25-30 inch-pounds.
5. Tighten clamps around body of separator to 25-30 inch-pounds.
6. Connect water drain line. Install hose clamps. Torque nut on swivel fitting to 15-25 inch-pounds.
7. Close accesses.

REPLACEMENT OF WATER SEPARATOR ANTI-ICE VALVE — (S/N 370-2 thru -6, -8 and -9)

Removing Water Separator Anti-Ice Valve

1. Open fuselage accesses 1 and 51. (Refer to chapter 6, figure 6-00-3.)
2. Disconnect sensing line from anti-ice valve and loosen clamps at valve inlet and outlet; then slide clamps back on ducting and remove anti-ice valve.
3. Cover anti-ice valve, sensing line and duct openings.

Installing Water Separator Anti-Ice Valve

1. Remove protective covering from anti-ice valve, sensing line and duct openings.
2. Place Conoseal gasket (51134-100A) in flange of duct coupling and while supporting valve in position, loosely install Marman clamp.
3. Place gasket (265-533365) on mounting flange of valve; then bolt valve in position. Torque bolts to 25-35 inch-pounds.
4. Torque Marman clamp bolt to 35-40 inch-pounds.
5. Install line support clamp, spacer and bolt.
6. Connect sensing line. Torque "B" nut to 40-65 inch-pounds.
7. Close accesses.

REPLACEMENT OF WATER SEPARATOR DELTA P SENSOR FILTER ELEMENT — (S/N 370-2 thru -6, -8 and -9)

1. Open access doors 1 and 51.
2. Disconnect lines from each end of filter case and remove filter from airplane.
3. Disassemble filter case and remove filter.
4. Discard filter and replace with Purolator P/N 37820 and gasket P/N 54694.
5. Assemble and torque filter head in filter case to 45-55 inch-pounds. Install lockwire between filter head and filter case.
6. Install filter in sensor line with flow arrow on filter body pointing up. Tighten "B" nuts at both ends of filter.
7. Close access doors 1 and 51.

REPLACEMENT OF 35°F TEMPERATURE CONTROL/SENSOR — (S/N 370-1, -7 and 380-1 and Subs)

1. Open access doors 1 and 51.
2. Disconnect electrical plug from sensor.
3. Remove sensor by using hex portion of switch.
4. Install and tighten new sensor in duct.
5. Connect electrical plug to sensor.
6. Close access doors 1 and 51.

REPLACEMENT OF REFRIGERATION UNIT COOLING TURBINE — (S/N 370-1, -7, 380-1 and Subs)

Removing Refrigeration Unit Cooling Turbine

1. Disconnect hot air line at turbine by removing 3 screws around flange securing hot air line to turbine case.

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MAINTENANCE PRACTICES (Continued)

REPLACEMENT OF REFRIGERATION UNIT COOLING TURBINE — (S/N 370-1, -7, 380-1 and Subs) (Cont'd)

2. Loosen clamps from the four turbine air outlet ducts, then slide hose up on ducts. Remove the turbine forward duct.
3. Remove the forward attaching bolts (turbine to heat exchanger) around periphery of turbine.
4. Loosen turbine support bracket (bracket to fuselage bolts) bolts to permit bracket to be moved.
5. Remove the eight exhaust duct screws.
6. Remove exhaust duct.
7. Remove flapper valve dampener.
8. Remove flapper check valve.
9. Remove flapper valve retainer halves.
10. Pull turbine forward away from heat exchanger and remove from airplane.

Installing Refrigeration Unit Cooling Turbine.

1. Position turbine in heat exchanger.
2. Install bolts around periphery of turbine. Tighten bolts.
3. Tighten turbine support bracket bolts.
4. Lockwire applicable plugs from cooling turbine to bolts around periphery of turbine.
5. Install flapper valve retainer halves.
6. Install flapper check valve.
7. Install flapper valve dampener.
8. Install exhaust duct and tighten screws.
9. Install turbine hot air and cooling air ducts. Tighten clamps and bolts.
10. Check turbine oil level. Service as required.
11. Inspect complete installation for security and applicable safeties.
12. Perform operational check of cooling turbine.
13. Close access doors.

CLEANING WATER SEPARATOR CONDENSER

1. Open access doors 1 and 55.
2. Remove water separator from airplane.
3. Remove clamp from end of separator, then remove end of separator.
4. Remove the eight screws securing condenser support to separator, then remove condensor support from separator.
5. Remove springs securing condenser (sock) to support.
6. Remove sock and wash in a solution of warm water and mild soap. Rinse thoroughly.
7. Inspect poppet and poppet seating surface of by-pass valve for corrosion, nicks, and freedom of operation.
8. Install sock and springs on support.
9. Install support in separator housing and install 8 screws. Tighten screws.
10. Assemble separator housing and install clamp. Tighten clamp.
11. Disassemble water drain line swivel fitting and clean finger screen. Blow out fitting. Reassemble and install drain fitting using new "O" rings MS29513-111 as required, (see figure 21-50-5).
12. Install separator in airplane.
13. Check complete installation for security. Check installation for air leaks during engine run.

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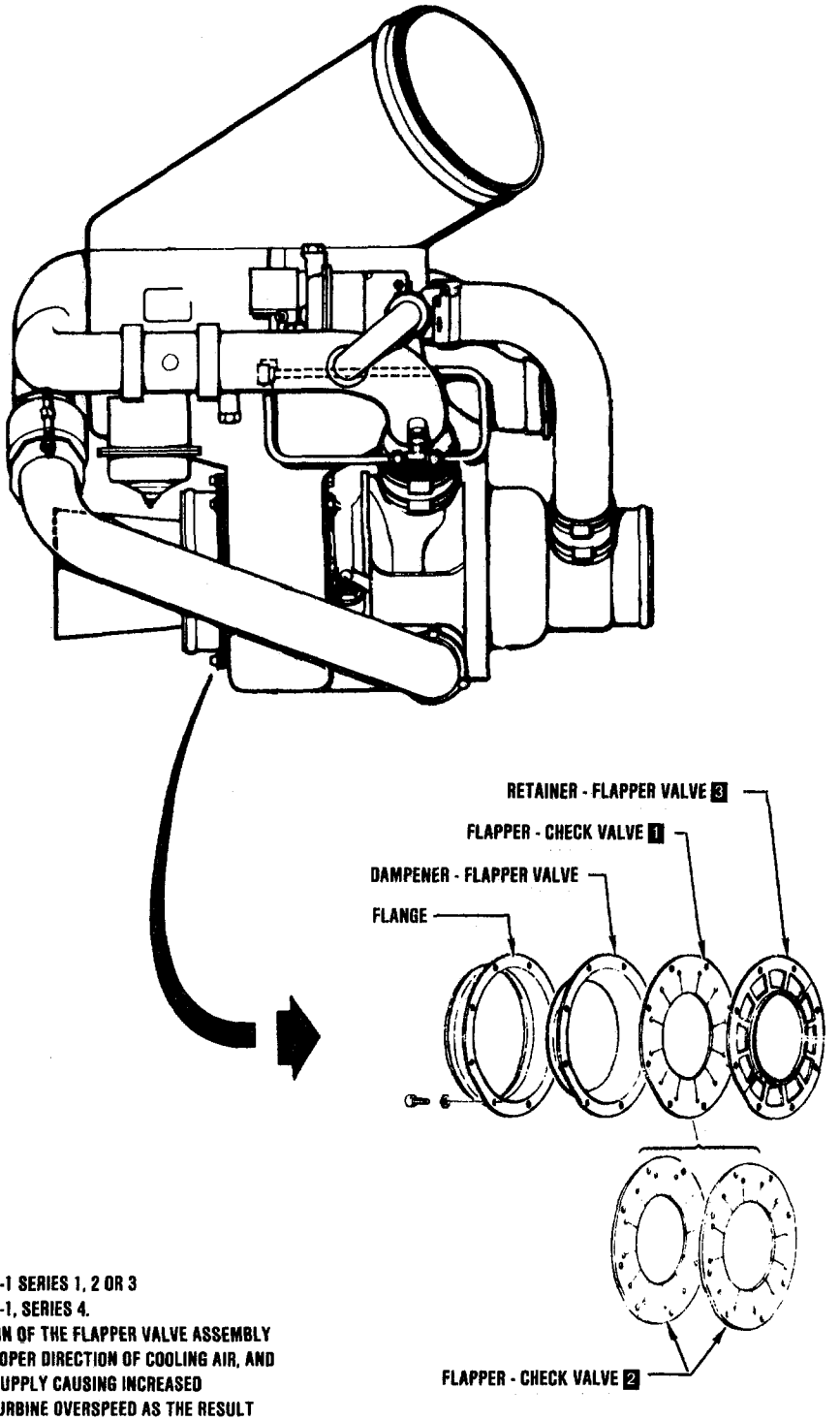

1 USED ON P/N 2200150-1 SERIES 1, 2 OR 3
2 USED ON P/N 2200150-1, SERIES 4.
3 REVERSE INSTALLATION OF THE FLAPPER VALVE ASSEMBLY WILL RESULT IN IMPROPER DIRECTION OF COOLING AIR, AND INCREASED RAM AIR SUPPLY CAUSING INCREASED TEMPERATURE AND TURBINE OVERSPEED AS THE RESULT OF FAN UNLOADING.

Figure 21-50-4 — Cooling Turbine Installation — S/N 370-1, -7, 380-1 and Subs

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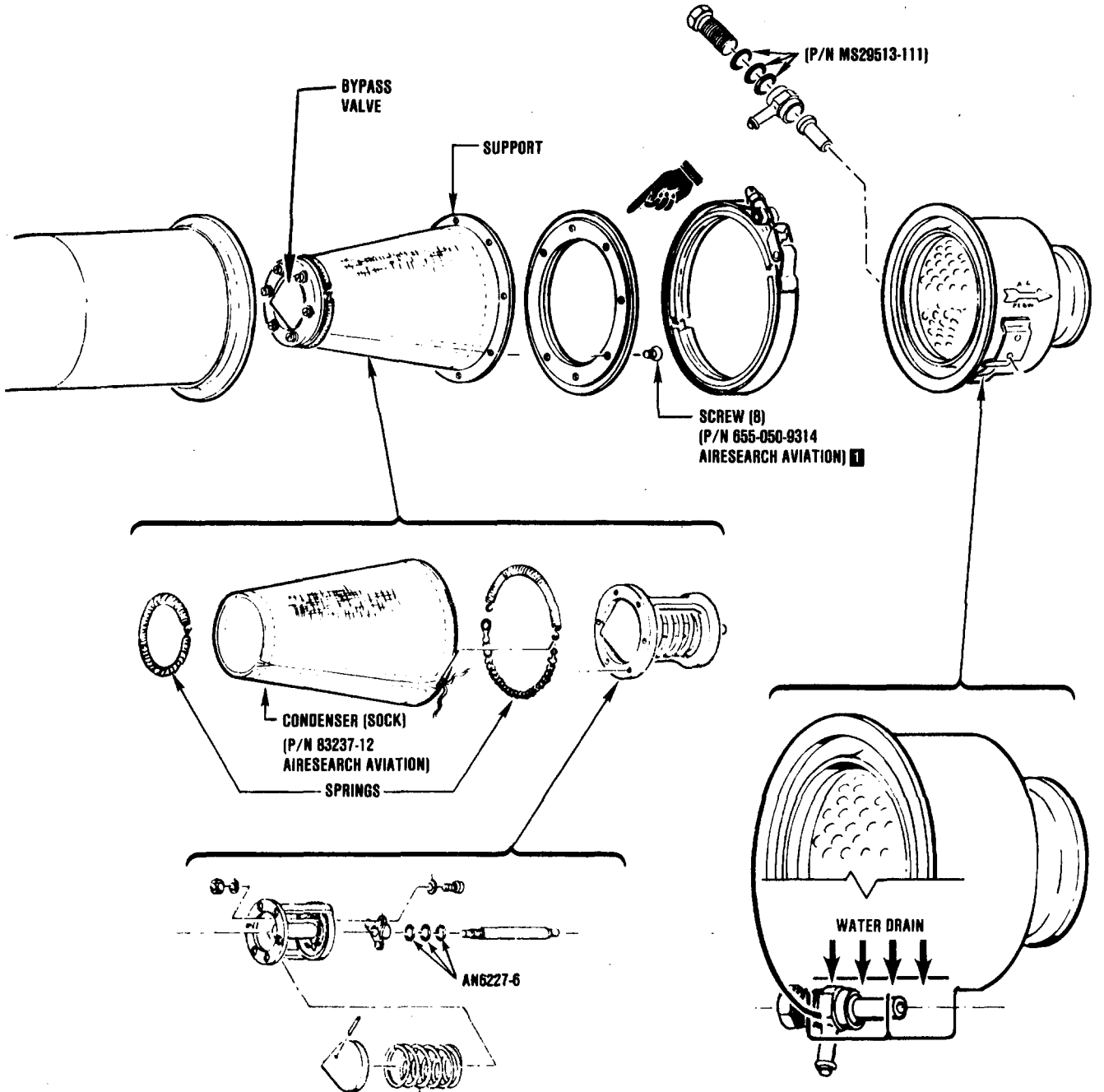

1 INTERCHANGEABLE WITH 1000-008-7B, INDUSTRIAL SCREW PRODUCTS, LOS ANGELES, CA.

2 SPRING FREE LENGTH 6.46 IN. LOAD REQUIRED TO COMPRESS SPRING 2.50 INCHES MUST BE 9.5 (±0.50) POUNDS.

NOTE

It is essential to install the housing in such a way that the water drain is parallel with the ground. Failure to install this way will not allow moisture to drain properly.

Figure 21-50-5 — Water Separator Condenser

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MAINTENANCE PRACTICES (Continued)

CHECK REGULATOR SETTING — PRESSURE REGULATING VALVE — 3 WHEEL COOLING TURBINE (AiResearch)

1. Install a tee in the downstream pressure sensing line and install a pressure gauge (0 to 100 psi) at tee location (see Figure 21-50-6).
2. Operate one engine at approximately 90 percent power and check that downstream pressure is regulated from 22 to 26 psi.
3. Vary engine RPM from 78 to 95 percent and check that downstream pressure remains regulated from 22 to 26 psi.
4. If the regulated pressure is not within limits specified, regulating valve must be replaced.

ALTERNATE METHOD

1. Disconnect regulating pressure supply line from pressure regulating valve.
2. Attach pressure gauge (0 - 50 psi) and air supply to pressure supply line.
3. Valve indexer should move to closed position when air pressure of 22 to 26 psi is applied to valve. See Figure 21-50-6.

NOTE

Indexer is located on left side of valve. A mirror is required to view indexer.

4. If valve does not meet the requirements specified in step 3, in either the opening or closing position, replace valve.

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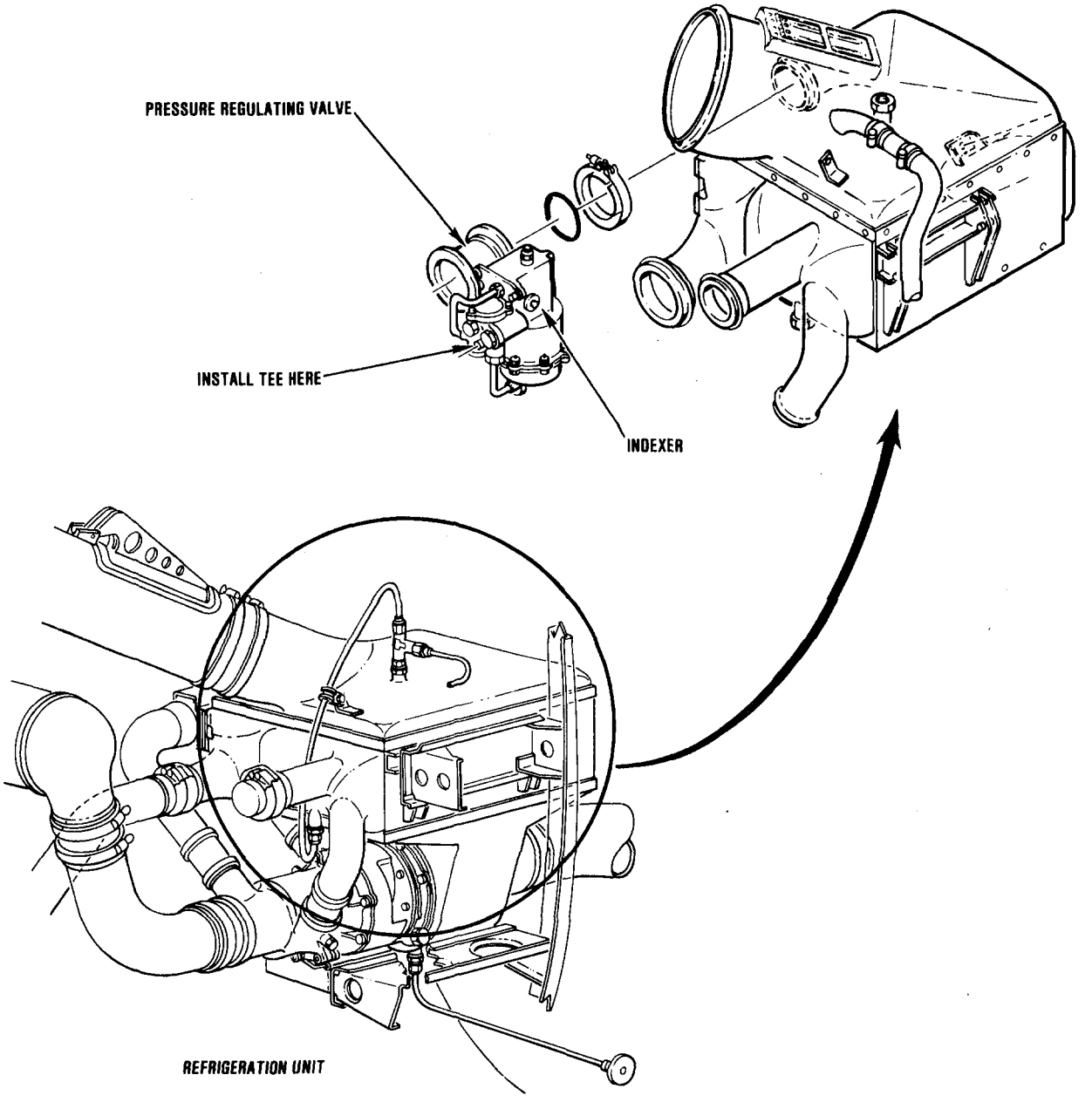

Figure 21-50-6 — Pressure Regulating Valve — 3 Wheel Cooling Turbine

NEW

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NOSE AND ELECTRONIC EQUIPMENT COMPARTMENT COOLING - DESCRIPTION AND OPERATION

1. Nose Compartment Cooling in Flight

The nose compartment, forward of station 50, is cooled in flight by cabin air discharged through the cabin pressure regulator. In airplanes with a pressurized nose compartment (NA-370 airplanes and NA-380 airplanes -1 through -23 not changed by Service Letter No. 7) the cabin air circulates through the nose compartment and is discharged from the nose compartment through the nose compartment pressure regulator (outflow valve). In airplanes with unpressurized nose compartment (NA-380 airplanes -1 through -23 changed by Service Letter No. 7, NA-380-24 and later), the cabin air circulates through the nose compartment and flows unrestricted into the electronic equipment compartment.

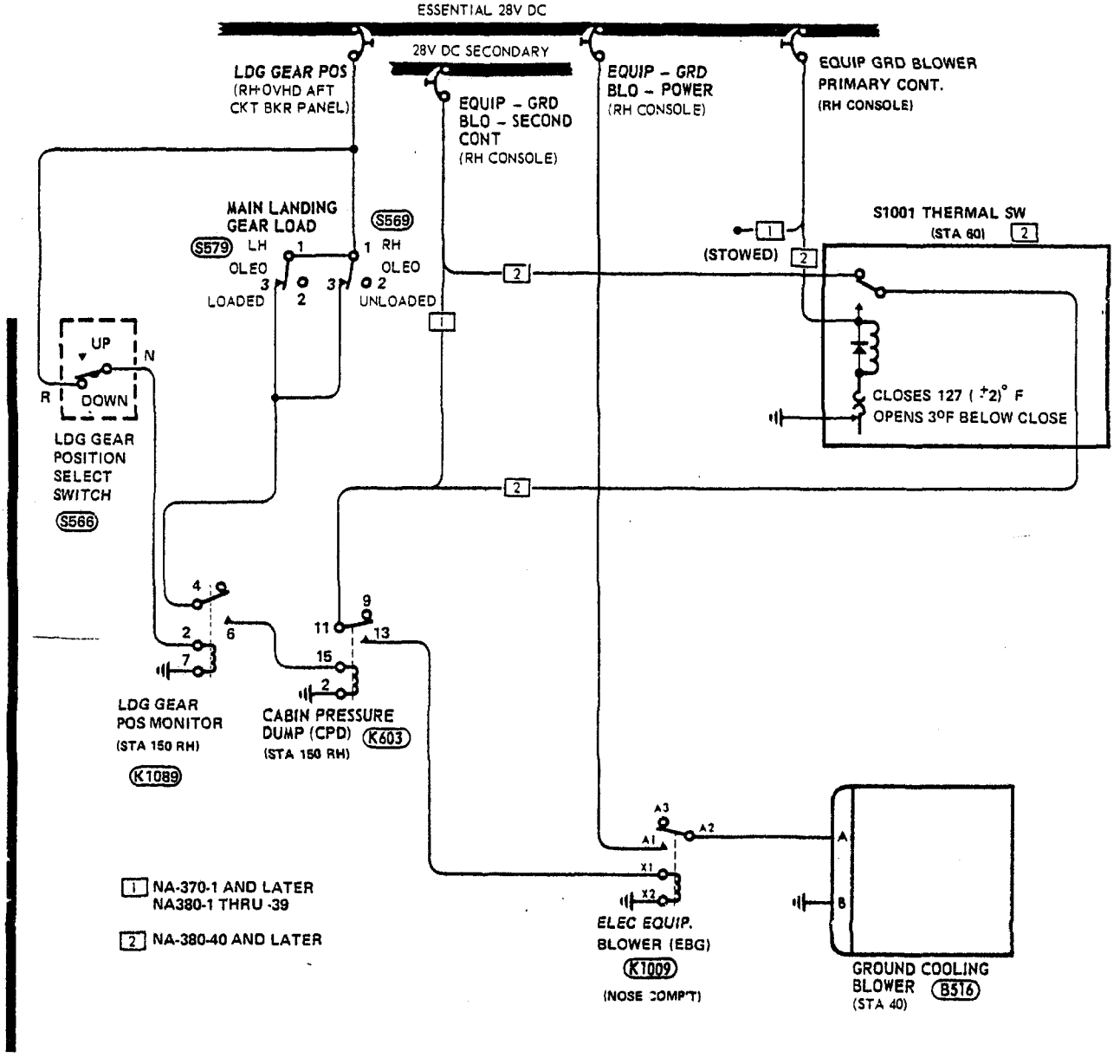
2. Nose Compartment Cooling on the Ground

When the airplane is on the ground (main landing gear loaded), a blower located in the lower part of the nose compartment draws air from beneath the fuselage and supplies it through a check valve to the nose compartment. The blower will not operate unless both the essential and secondary dc busses are energized, except that on airplanes NA-380-40 and later a thermoswitch S1001 (see figure 21-51-1) permits operation on essential bus power alone when the nose compartment temperature rises above 127° $\pm 2^{\circ}$F.

Relay K1009 receives power normally from the secondary bus; however, when thermal switch S1001 closes (127° $\pm 2^{\circ}$F) the relay is powered from the essential bus. (See figure 21-51-1.)

3. Electronic Equipment Compartment Cooling

The electronic equipment compartment, located between fuselage stations 50 and 76, is not pressurized but must be cooled. Cabin air, after circulating through the nose compartment, enters the left side at station 50. Ram air (in flight) enters the right side at station 50 from a ram air duct leading from an inlet on the lower right hand side of the fuselage forward of station 50. Both the ram air and the cabin (nose compartment) air flow through openings between sections of the compartment, and are exhausted through outlet vents, located on either side of the fuselage. On NA-370 and NA-380-1 thru -37 airplanes not changed by Service Letter No. 10, a ram air duct, leading air from an inlet on the lower left hand side of the fuselage forward of station 50, provides cooling air for the left hand inverter. When the airplane is on the ground, blower air from the nose compartment also circulates through the electronic equipment compartment.



S370-2-52-10B

Electronic Equipment Compartment Cooling System Electrical Schematic
Figure 21-51-1

NOSE AND ELECTRONIC EQUIPMENT COMPARTMENT COOLING - TROUBLE SHOOTING

1. Trouble Shooting Electronic Equipment Compartment Cooling System

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. EQUIPMENT IN ELECTRONIC EQUIPMENT COMPARTMENT OVERHEATING DURING FLIGHT.		
(1) Ram-cooling-air ducting and/or outlet vents clogged.	Make visual check of ducting and ram-air outlet vents.	Remove foreign matter.
B. EQUIPMENT IN ELECTRONIC EQUIPMENT COMPARTMENT OVERHEATING DURING GROUND OPERATIONS.		
(1) Defective cooling-air blower.	With external dc electrical power on dc essential bus (figure 21-51-1), engage proper circuit breakers and check for electrical power at blower disconnect plug. If power is present, blower motor is defective. If power is not present at unit, check circuitry.	Remove and replace cooling-air blower.
(2) Defective relay (K603 and K1009).	See figure 21-51-1 and check for proper relay operation.	Remove and replace defective relay.
(3) Defective Main gear load switches, or switches improperly adjusted.	Cabin pressure dump relay (K603) not energized.	Remove and replace defective switches, or make proper switch adjustments.
(4) Ram-cooling-air ducting and/or outlet vents clogged.	Make visual check of ducting and ram-air outlet vents for obstruction.	Remove obstructions.
(5) Check valve forward of cooling-air blower will not open.	Make visual check of check valve for malfunction.	Repair or replace check valve.
*(6) Nose pressure regulator defective.	Check air exhaust from regulator.	Replace regulator.

*Airplanes NA-370 and NA-380-1 thru -23 Not Changed by Service Letter No. 7

NOSE AND ELECTRONIC EQUIPMENT COMPARTMENT COOLING - MAINTENANCE PRACTICES

1. Replacement of Ground Cooling Blower

A. Removing Ground Cooling Blower

(1) Remove accesses 7 and 30 (Chapter 6, figure 6-00-3).

(2) Reach in through access 30 (Chapter 6, figure 6-00-3), and remove sealing compound from eight mounting screws.

(3) Remove all but one top screw.

(4) Remove electrical connector from blower.

(5) Hold blower with one hand; then remove remaining mounting screw. Remove blower and gasket. Cover electrical plug, blower receptacle, and blower opening.

B. Installing Ground Cooling Blower

(1) Remove covering or plug from blower receptacle, electrical plug, and blower opening.

(2) Position blower and gasket in blower opening. Install top mounting screw, but do not tighten.

(3) Install remaining screws; then tighten all screws.

(4) Seal screwheads. Use any of the following sealers, or equivalent, for internal sealing:

- PR-1222 — Products Research Co, 3126 Los Feliz Blvd, Los Angeles, California
- Compound 707 — Coast Pro-Seal & Mfg Co, 2235 Beverly Blvd, Los Angeles, California
- EC-1528 and EC-1941 — Minnesota Mining & Mfg Co, Los Angeles, Calif.

(5) Install electrical connector.

(6) Perform electronic equipment compartment cooling system check-out procedure, then install applicable accesses.

2. Electronic Equipment Compartment Cooling System Check-out Procedures

A. Checking Electronic Equipment Compartment Cooling System

(1) Make sure the following circuit breakers are engaged:

- "LDG GEAR - POS"
- "EQUIP GRD BLOW - SECOND CONT"
- "EQUIP GRD BLO - POWER"

(2) Connect external 28-volt dc power.

(3) Check that cooling-air blower is operating in the electronic equipment compartment.

(4) Position both main landing gear switches, installed on main landing gear scissors, at an unload (gear extended) condition. Check that cooling-air blower stops operating.

(5) Alternately position right, then left, landing gear switch at loaded (gear compressed) condition. Cooling-air blower should operate with either switch or both switches in the loaded position.

(6) Remove electrical power from airplane bus system. Disengage circuit breakers used in this test.

(7) Position both landing gear load switches at loaded position and install cotter pin.

TEMPERATURE CONTROL - DESCRIPTION AND OPERATION

1. Cabin Temperature Control System

The cabin temperature controls (figure 21-00-2 or 21-00-2A) consist of a switch and rheostat on the electrical panel on the aft face of the power control lever pedestal in the cockpit. The "CABIN TEMP" switch has four settings: AUTO, OFF, MAN COLD, and MAN HOT. The "AUTO CONT" rheostat has COLD and HOT settings as well as a number of intermediate positions between COLD and HOT.

For normal operation, the "CABIN TEMP" switch is positioned at AUTO and the "AUTO CONT" rheostat toward HOT or COLD to maintain the cabin temperature between 60°F and 80°F, as desired by the pilot. This temperature is automatically controlled by a cabin air temperature sensor and blower and a duct air temperature sensor. These sensors, in conjunction with the cabin air temperature control box, position the hot-air mixing valve, thereby proportioning the proper amounts of heating and cooling air to be supplied.

If the automatic temperature control fails, the hot-air mixing valve can be opened or closed by positioning the "CABIN TEMP" control switch at MAN HOT or MAN COLD respectively, or may be stopped at any point between open and closed by positioning the switch at OFF. A cabin air supply overheat thermoswitch in the conditioning air duct downstream of the hot- and cool-air mixing point, causes a light to come on if an overheat condition (over 265°F) occurs, alerting the pilot to take appropriate action. A control system or a hot-air mixing valve failure is indicated if the light is on. The cabin air temperature control box is mounted on the forward side of the aft cabin pressure bulkhead.

2. Cabin Air Temperature Control Network

The cabin air temperature control network consists of the cabin air temperature control, cabin air temperature sensor, sensor blower, duct air temperature sensor, and cabin air overheat thermoswitch.

The cabin air temperature control unit is a foam compound potted, transistorized, two-stage, push-pull amplifier, which is used with the cabin air temperature sensor and dual-element duct air temperature sensor to modulate the air conditioning system hot-air mixing valve for automatic temperature control in the cabin. The cabin air temperature control is operated by a 115-volt, 400-cycle ac electrical power, and the hot-air mixing valve is operated by ac power output from the cabin air temperature control.

NOTE: The cabin air temperature sensor blower keeps cabin air circulating around the sensor when the temperature control system is in operation.

The cabin air temperature control, "AUTO CONT" selector, cabin air temperature sensor, and anticipator element (part of the dual-element duct air temperature sensor) automatically control positioning of the system hot-air mixing valve during lower temperature requirements in the cabin. A rate feed-back potentiometer, in the hot-air mixing valve, supplies signals

to a bridge compensating network in the cabin air temperature control, preventing hot-air mixing valve overtravel for system stabilization.

When cabin temperature requirement exceeds 220°F, a diode switching network in the cabin air temperature control cuts the cabin air temperature sensor out of the control network and allows the high-limit element (part of the dual-element duct air temperature sensor) to supply temperature requirement signals to the cabin air temperature control. This in turn positions the system hot-air mixing valve as required. The cabin air overheat thermoswitch, installed in ducting directly above duct air temperature sensor, operates the overheat caution lights if an overheat condition (265°F) develops in the air conditioning system ducting.

3. Cabin Temperature Hot-air Mixing Valve

The hot-air mixing valve is installed in the engine bleed-air ducting, downstream from the flow limiter valve. It regulates the flow of hot engine compressor air that is mixed with cooled air coming from the refrigeration unit. The valve is electrically operated and is controlled by the cabin air temperature control. (See figure 21-00-2.)

4. Cockpit Temperature Control System - Aircraft NA-370-1, -7, NA-380-1 and Later

The cockpit temperature controls (figure 21-00-2A) consist of a switch and rheostat on the electrical panel on the aft face of the power control lever pedestal in the cockpit. The "COCKPIT TEMP" switch as four settings: AUTO, OFF, MAN COLD, and MAN HOT. The "AUTO CONT" rheostat has COLD and HOT settings as well as a number of intermediate positions between COLD and HOT.

For normal operation, the "COCKPIT TEMP" switch is positioned at AUTO and the "AUTO CONT" rheostat toward HOT or COLD to maintain the cabin temperature between 60°F and 80°F, as desired by the pilot. This temperature is automatically maintained by a cockpit air temperature sensor installed in the cockpit air supply ducting. This sensor in conjunction with the cockpit air temperature control box, positions the cockpit hot-air mixing valve, thereby proportioning the proper amounts of heating and cooling air to be supplied to the cockpit.

If the automatic temperature control fails, the hot-air mixing valve can be opened or closed by positioning the "COCKPIT TEMP" control switch at MAN HOT or MAN COLD respectively, or may be stopped at any point between open and closed by positioning the switch at OFF. A cockpit air supply overheat thermoswitch in the conditioning air duct downstream of the hot- and cool-air mixing point, causes a light to come on if an overheat condition occurs, alerting the pilot to take appropriate action. A control system or a hot-air mixing valve failure is indicated if the light is on. The cockpit air temperature controller unit is located at fuselage station 194, right side.

5. Cockpit Air Temperature Control Network - Aircraft NA-370-1, -7, NA-380-1 and Later

The cockpit air temperature control network consists of the cockpit air temperature control, cockpit duct air temperature sensor, and cockpit air overheat thermoswitch.

The cockpit air temperature control unit is a foam compound potted, transistorized, two-stage, push-pull amplifier, which is used with the cockpit air temperature duct sensor to modulate the cockpit system hot-air mixing valve for automatic temperature control in the cockpit. The cockpit air temperature control is operated by 115-volt, 400-cycle ac electrical power, and the hot-air mixing valve is operated by ac power output from the cockpit air temperature control.

The cockpit air temperature control, "AUTO CONT" selector, cockpit air temperature duct sensor, and anticipator element (part of the dual-element duct air temperature sensor) automatically control positioning of the cockpit hot-air mixing valve during lower temperature requirements in the cockpit. A rate feed-back potentiometer, in the hot-air mixing valve, supplies signals to a bridge compensating network in the cockpit air temperature control, preventing hot-air mixing valve overtravel for system stabilization.

When cockpit temperature requirement exceeds 220°F, a diode switching network in the cockpit air temperature control cuts the cockpit air temperature sensor out of the control network and allows the high-limit element (part of the dual-element duct air temperature sensor) to supply temperature requirement signals to the cockpit air temperature control. This in turn positions the cockpit hot-air mixing valve as required. The cockpit air overheat thermo-switch, installed in the cockpit ducting operates the overheat caution light if an overheat condition (265°F) develops in the cockpit system ducting.

6. Cockpit Temperature Hot-air Mixing Valve - Aircraft NA-370-1, -7, NA-380-1 and Later

The hot-air mixing valve is installed in the engine bleed-air ducting, downstream from the flow limiter valve. It regulates the flow of hot engine compressor air that is mixed with cooled air coming from the refrigerator unit. The valve is electrically operated and is controlled by the cockpit air temperature control. (See figure 21-00-2A.)

TEMPERATURE CONTROL - TROUBLE SHOOTING

1. Trouble Shooting Cabin Temperature Control System

PROBABLE CAUSE	ISOLATION PROCEDURES	REMEDY
A. AIRFLOW TO CABIN OR COCKPIT DOES NOT AGREE WITH "TEMP" SELECTION.		
(1) Faulty hot-air mixing valve, cabin or cockpit air temperature control, duct temperature sensor, "AUTO CONT" selector, "CABIN or COCKPIT TEMP" selector, or "CABIN AIR" selector.	See figure 21-00-2 or 21-00-2A and, with external dc electrical power on airplane bus system, refer to cabin or cockpit temperature control system check-out procedure. If hot-air mixing valve does not respond when the "AUTO CONT" selector is moved from hot to cold, or from cold to hot, check for presence of ac voltage at mixing valve disconnect plug. Also, check for mixing valve operation with "CABIN or COCKPIT TEMP" switch at AUTO during engine operation. If ac power is at disconnect plug of hot-air mixing valve, and valve does not respond, replace mixing valve. If no electrical power is at plug, check for power at cabin or cockpit air temperature control disconnect plug. Make sure control selector switches are properly positioned.	Replace any faulty units and repeat check-out procedure.
(2) Hot-air mixing valve stuck in closed position.	Make sure "CABIN or COCKPIT TEMP" control switch was not previously operated at MAN COLD and positioned at OFF. With "CABIN or COCKPIT TEMP" switch at AUTO, check system as indicated in isolation procedure of probable cause (1).	If mechanical failure of hot-air mixing valve is indicated, replace valve.
B. CABIN AIR PRESSURE FLUCTUATION WHEN CABIN AIR TEMPERATURE SELECTOR IS ON AUTO.		
(1) Faulty feedback potentiometer in hot-air mixing valve.	Check that valve resistance at pins "B" and "D" is 900 to 1100, with valve open. Resistance must steadily increase on closing to between 8.5 and 12.0 kilohms.	Remove and replace hot-air mixing valve.

1. Replacement of Cabin Hot-air Mixing Valve

A. Removing Cabin Hot-air Mixing Valve

(1) Open fuselage accesses 1 and 55. (Refer to Chapter 6, figure 6-00-3.)

(2) Disconnect electrical wiring from mixing valve.

(3) Loosen clamps at both sides of mixing valve and remove mixing valve from ducting.

(4) Remove gaskets, and check for re-use.

(5) Cover mixing valve and ducting openings to prevent entry of foreign material.

B. Installing Cabin Hot-air Mixing Valve

(1) Remove protective covering from mixing valve and ducting openings.

(2) Insert gaskets in flanges of connections on valve; place mixing valve in mounting position between ducting. Install mounting bolts.

(3) Install Marman clamps. Torque to 45-50 inch-pounds.

(4) Connect mixing valve electrical wires.

(5) Perform cabin temperature control system check-out - engines not operating (refer to paragraph 2, B), then close accesses.

1A. Replacement of Cockpit Hot-air Mixing Valve - Airplanes NA-370-1, -7, NA-380-1 and Later

A. Removing Cockpit Hot-air Mixing Valve

(1) Open fuselage accesses 1 and 55. (Refer to Chapter 6.)

(2) Disconnect electrical wiring from mixing valve.

(3) Support curved outlet duct and remove four bolts (with washers) from outlet flange of valve. Separate duct from valve and remove gasket at flange. Discard gasket. Retain bolts and washers for reinstallation.

(4) Support valve and remove four bolts (with washers) from inlet flange of valve. Remove valve from airplane. Discard gasket. Retain bolts and washers for reinstallation.

(5) Cover mixing valve and ducting openings to prevent entry of foreign material.

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B. Installing Cockpit Hot-air Mixing Valve

(1) Remove protective covering from mixing valve and ducting.

(2) Using a new gasket (P/N 380-533323-3) at inlet of valve, install valve in upright position on end of adapter. Loosely secure valve inlet flange to adapter ducting with four bolts and washers.

(3) Using a new gasket (P/N 380-533323-3) at outlet of valve, attach curved ducting to outlet flange. Loosely secure with four bolts and washers.

(4) Tighten all eight bolts and install lockwire.

(5) Connect electrical plug and install lockwire.

(6) Perform cockpit temperature control system check-out, as applicable (refer to paragraph 3), then close accesses.

1B. Cleaning of Cabin Temperature Sensor/Blower

NOTE: On all NA-370 and NA-380 airplanes, the cabin temperature sensor/blower is installed on the left rear side of the passenger compartment forward bulkhead.

A. Removing Cabin Temperature Sensor/Blower

(1) Remove electrical power from airplane.

(2) Disconnect electrical leads from sensor/blower assembly.

(3) Remove screws securing sensor/blower assembly to mounting surface and remove sensor/blower assembly.

(4) Remove fasteners securing sensor/blower assembly to cover and remove sensor/blower. Remove blower cover.

(5) Clean blower blades and blower cover with a suitable cleaning solvent and thoroughly dry using low pressure compressed air.

B. Installing Cabin Temperature Sensor/Blower

(1) Install blower cover.

(2) Secure sensor/blower assembly to cover.

(3) Install sensor/blower assembly to mounting surface using previously removed fasteners.

(4) Install electrical leads to sensor/blower assembly.

(5) Check complete installation for security.

(6) Perform cabin temperature control system check-out with engines running (refer to paragraph 2, A).

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2. Cabin Temperature Control System Check-out Procedures

A. Checking Cabin Temperature Control System With Engines Operating

NOTE: If any of the following checks fail to produce proper results, perform check outlined in paragraph B.

- These checks may be performed using the APU (if installed) instead of engines.

(1) With engines operating, 115-volt ac system operating, and CABIN TEMP CONT circuit breaker engaged (left hand aft overhead circuit breaker panel), place PRESSURIZATION (or CABIN AIR) selector switch at BOTH ENG position. (If only one engine is operating place selector switch at that engine position.)

(2) On center pedestal, lift switch cover and move toggle of CABIN TEMP switch toward MAN HOT. Check that air from cabin air outlets becomes warmer.

(3) Move CABIN TEMP switch to MAN COLD. Check that air from cabin air outlets becomes colder.

(4) Move CABIN TEMP switch to AUTO and close switch cover.

(5) Rotate CABIN TEMP (or AUTO CONTROL - CABIN) rheostat fully clockwise, then (after a pause) fully counterclockwise. Check that air from a cabin air outlet becomes first warmer then cooler.

(6) Select a comfortable cabin temperature. Check that control system maintains temperature selected.

NOTE: If desired, procedures for checking cockpit system may be performed at this time. Refer to paragraph 3.

(7) Shut down engines (as required).

B. Checking Cabin Temperature Control System - Engines Not Operating

NOTE: Each component of the cabin temperature control system should be properly installed and connected for this test. The system must be electrically energized with 115-volt, 400 Hz, ac power.

- If any of these checks fail to produce proper results, perform the non-operating resistance checks outlined in paragraph C.

(1) Check that circuit breaker CABIN TEMP CONT (left hand aft overhead circuit breaker panel) is engaged.

(2) Place PRESSURIZATION (or CABIN AIR) selector switch at BOTH ENG position.

(3) On center pedestal, lift switch cover and move toggle of CABIN TEMP switch toward MAN HOT. Check that cabin hot-air mixing valve position indicator indicates that valve is fully open. (The valve, located at fuselage station 360, is accessible through fuselage accesses 1 and 55, refer to Chapter 6).

(4) Move CABIN TEMP switch to MAN COLD. Check that cabin hot-air mixing valve operates toward fully closed position. (Valve may be heard to operate and indicator on valve moves toward S (shut or closed).

(5) Rotate CABIN TEMP (or AUTO CONTROL - CABIN) rheostat fully clockwise (toward HOT or WARMER), then move CABIN TEMP switch to AUTO and close switch cover. Check that cabin hot-air mixing valve goes open (to 0).

NOTE: On extremely hot or cold days, the automatic system may not permit full valve travel. This condition is acceptable providing sufficient movement is observed to assure that valve is operating properly.

(6) Rotate rheostat fully counterclockwise. Check that valve goes closed.

C. Non-operating Resistance Check of Cabin Temperature System Components and Wiring

A multimeter (Simpson 260, or equivalent capable of 20,000 ohms per volt sensitivity) and a thermometer (N3A, or equivalent) are required for these checks. These checks should be made with the system non-energized and the cabin air temperature control unit removed. All resistance checks are made from the rear (outboard) of the control unit connector block. If any of these checks fails to give proper results, the appropriate check of the related component should be made as outlined in paragraph D.

CAUTION: IN ALL RESISTANCE MEASUREMENTS OF TEMPERATURE SENSORS, ENOUGH TIME MUST BE GIVEN THE SENSOR, ESPECIALLY IN STILL AIR, TO STABILIZE AT THE TEMPERATURE OF THE SURROUNDING AMBIENT AIR. ALSO; TO AVOID SELF-HEATING, THE RESISTANCE MEASURING DEVICE USED MUST LIMIT THE CURRENT THROUGH THE SENSOR. IN THE TEMPERATURE RANGES SHOWN IN FIGURE 21-60-1, THE CURRENT SHOULD BE LIMITED TO 100 MICROAMPERES. THE CURRENT SHOULD BE LIMITED TO 10 MICROAMPERES AT TEMPERATURES BELOW 30°F.

(1) For cabin ("AUTO CONT") temperature rheostat (RDI R507), measure dc resistance between sockets "F" and "G" of airplane connector. With the "AUTO CONT" temperature selector in the full clockwise (HOT) position, the resistance must be less than 150 ohms. With the "AUTO CONT" temperature selector in the full counterclockwise (COLD) position, the resistance must be between 30 and 40 kilohms.

(2) For cabin temperature sensor, measure dc resistance between sockets "S" and "T" of airplane connector. The resistance should fall within the limits of curve I. (See figure 21-60-1.) (Place thermometer in vicinity of cabin air temperature sensor/blower.)

(3) For duct air temperature sensor, measure dc resistance between sockets "P" and "R" and between sockets "d" and "e" of airplane connector (P605). The resistance between "d" and "e" should fall within the limits of curve I (figure 21-60-1), and the resistance between "P" and "R" should fall within the limits of curve II (figure 21-60-2). (Place thermometer as close to duct sensor as practical. If results are not within limited curves, it may be necessary to remove sensor from duct (with electrical wires connected) and repeat this check with thermometer adjacent to sensor probe.)

(4) For hot-air mixing valve, measure dc resistance between sockets indicated, and check that measured values are within the following specified limits:

PIN SOCKETS	RESISTANCE VALUES
Between "M" and "E"	600 to 900 ohms
Between "K" and "E"	600 to 900 ohms
Between "N" and "E"	9 to 14 kilohms
Between "H" and "E"	0.5 to 1.5 kilohms when valve is in full-open position, or 7 to 14 kilohms when valve is in full-closed position

(5) The cabin air temperature control unit (RDI CN500) cannot be checked without additional equipment. If system did not operate properly during operation checks (paragraphs A or B), and if results of non-operating checks of every other component were satisfactory, replace temperature control unit and repeat paragraphs B and A.

D. Checking Individual Cabin Air Temperature System Components

These checks are made on individual components. If proper results are obtained here, when results of the corresponding check at the control unit connector block were not satisfactory, the airplane wiring is faulty.

(1) On cabin ("AUTO CONT") temperature selector, with system wiring disconnected from at least one terminal, check that dc resistance between terminals "A" and "B" of selector is less than 150 ohms when rheostat is at full clockwise position, and between 30 and 40 kilohms when rheostat is at full counterclockwise position. If unit does not meet this requirement, a new unit should be installed.

(2) On cabin air temperature sensor, with system wiring disconnected, check that dc resistance between terminals "A" and "B" is within limits of curve I. (See figure 21-60-1.) If the unit does not meet this requirement, a new unit should be installed.

(3) On duct temperature sensor, with system wiring disconnected, check that dc resistance between terminals "A" and "B" is within limits of curve II (figure 21-60-2), and resistance between terminals "C" and "D" is within limits of curve I (figure 21-60-1). If the unit does not meet this requirement, a new unit should be installed.

(4) On hot-air mixing valve, with system wiring disconnected, make approximate dc resistance measurements from terminals of actuator connector as follows:

TERMINALS	RESISTANCE VALUES
Between "C" and "B"	600 to 900 ohms
Between "A" and "B"	600 to 900 ohms
Between "E" and "B"	9 to 14 kilohms
Between "D" and "B" (VALVE OPEN)	Valve fully open, the resistance measured must be 900 to 1100 ohms and must steadily increase as valve goes toward closed.
Between "D" and "B" (VALVE CLOSED)	The resistance in the fully closed valve position must be between 8,500 and 12,000 ohms.

If the unit does not meet these requirements, a new unit should be installed in the airplane, and the check must be repeated.

3. Cockpit Temperature Control System Check-out Procedures - Airplanes NA-370-1, -7, NA-380-1 and Later

A. Checking Cockpit Temperature Control System with Engines Operating

NOTE: If any of the following checks fail to produce proper results, perform checks outlined in paragraph B.

- These checks may be performed using the APU (if installed) instead of engines.

(1) With engines operating, 115-volt ac system operating and CABIN TEMP CONT circuit breaker engaged (left hand aft overhead circuit breaker panel), place PRESSURIZATION (or CABIN AIR) selector switch at BOTH ENG position. (If only one engine is operating, place selector switch at that engine position.)

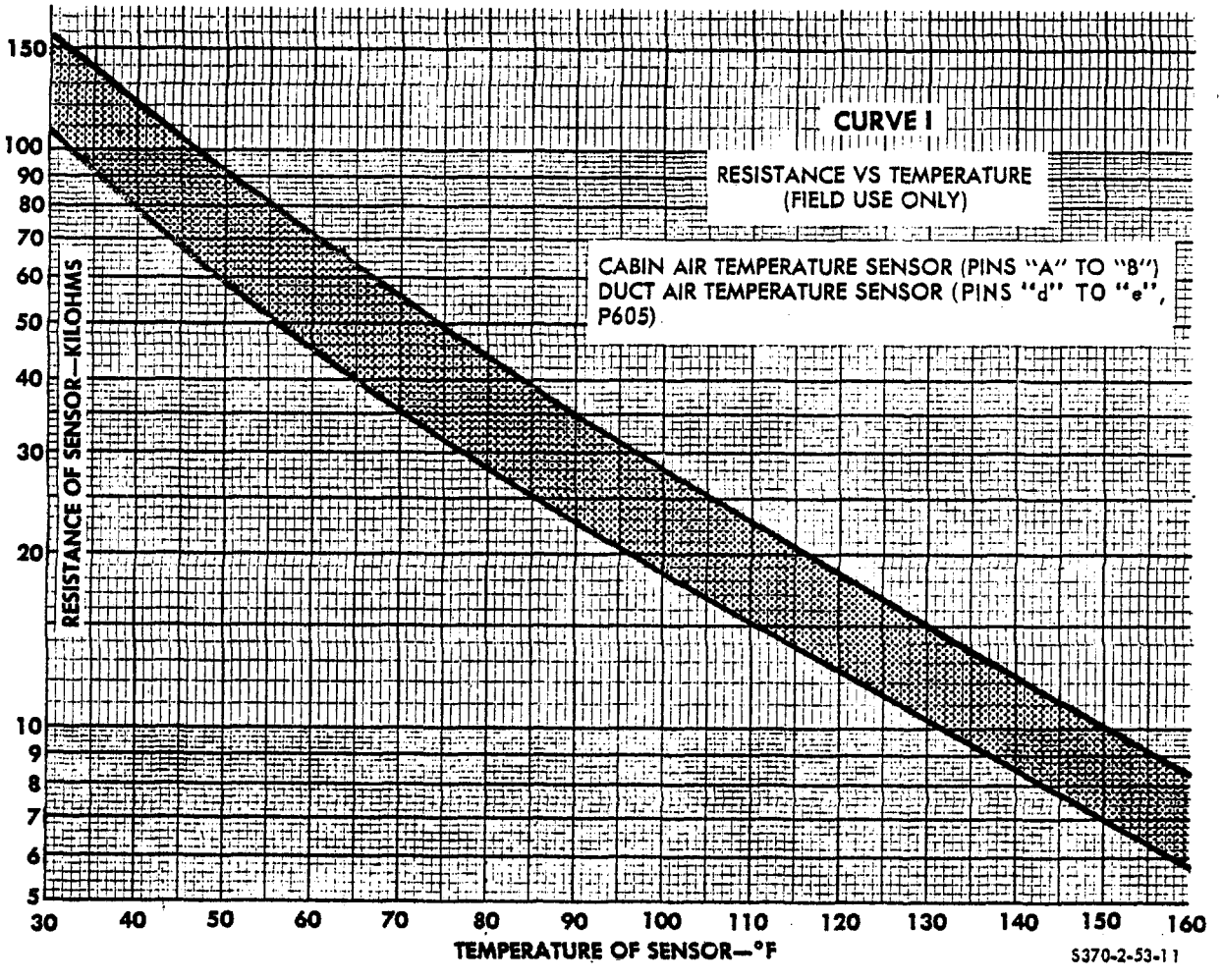
(2) On center pedestal, lift switch cover and move toggle of COCKPIT TEMP switch toward MAN HOT. Check that air from cockpit air outlets becomes warmer.

(3) Move COCKPIT TEMP switch towards MAN COLD, check that air from cockpit air outlets becomes colder.

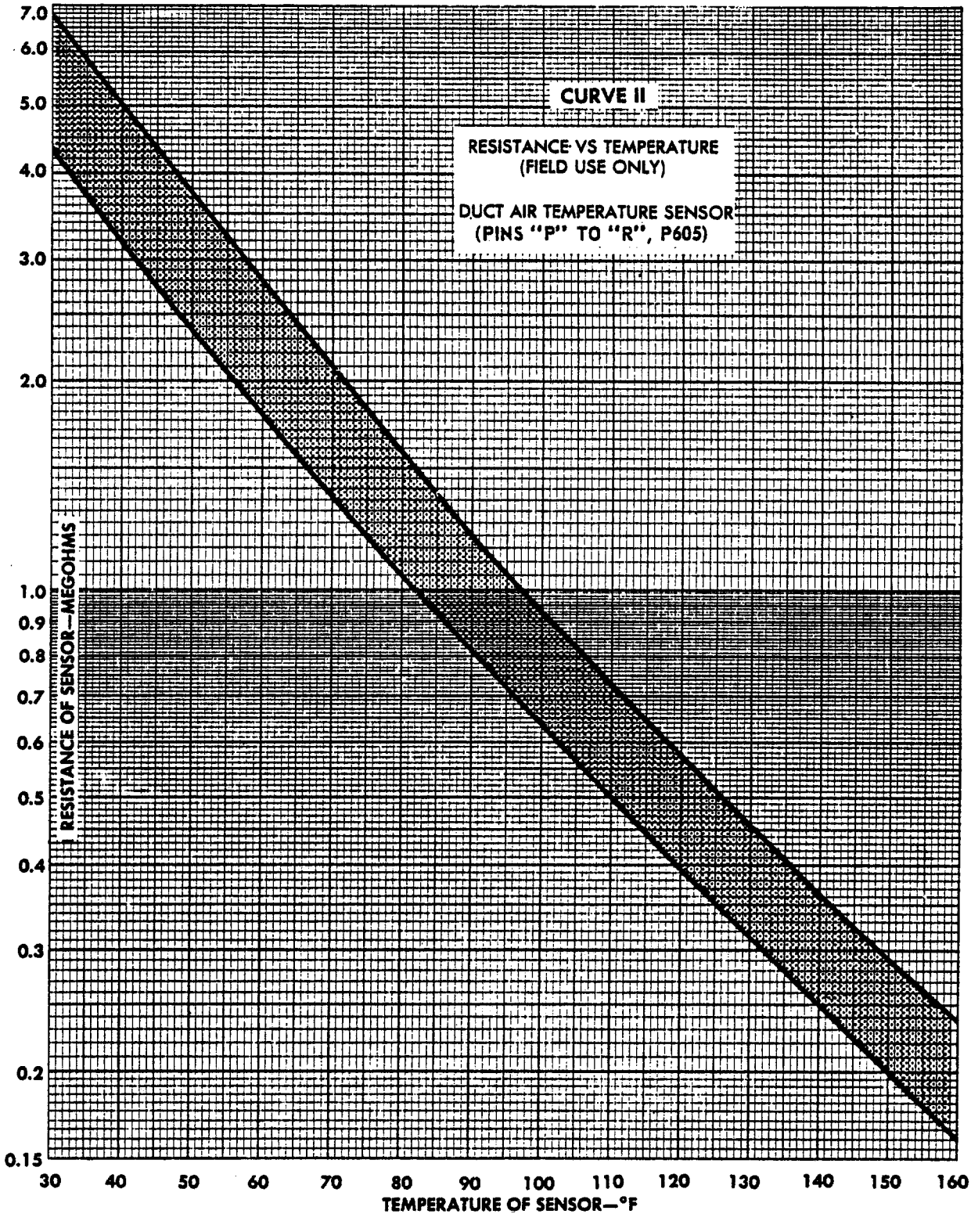
(4) Move COCKPIT TEMP switch to AUTO and close switch cover.

(5) Rotate COCKPIT TEMP (or AUTO CONTROL - COCKPIT) rheostat fully clockwise, then (after a pause) fully counterclockwise, check that air from a cockpit air outlet becomes first warmer, then cooler.

(6) Select a comfortable cockpit temperature. Check that control system maintains temperature selected.



Temperature Sensor Resistance Value Chart - Curve I
Figure 21-60-1



Temperature Sensor Resistance Value Chart - Curve II
Figure 21-60-2

(7) Shut down engines (as required).

B. Checking Cockpit Temperature Control System - Engines Not Operating

NOTE: Each component of the cockpit temperature control system should be properly installed and connected for this test. The system must be electrically energized with 115-volt, 400 Hz, ac power.

- If any of these checks fail to produce proper results, perform the non-operating resistance checks outlined in paragraph C.
- The cockpit hot-air mixing valve, located at fuselage station 357, is not readily visible; however, its operation may be checked by touch or by listening for the sound of its response to control signals.

(1) Check that CABIN TEMP CONT circuit breaker is engaged (left hand, aft, overhead circuit breaker panel).

(2) Place PRESSURIZATION (or CABIN AIR) selector switch at BOTH ENG position.

(3) Lift switch cover and move COCKPIT TEMP switch toggle toward MAN HOT position. Check that cockpit hot-air mixing valve goes open.

(4) Move COCKPIT TEMP switch toward MAN COLD position. Check that cockpit hot-air mixing valve goes closed.

(5) Rotate COCKPIT TEMP (or AUTO CONTROL - COCKPIT) rheostat fully clockwise, then move COCKPIT TEMP switch to AUTO (close cover). Check that cockpit hot-air mixing valve goes open.

(6) Rotate rheostat fully counterclockwise. Check that valve goes closed.

C. Non-operating Resistance Check of Cockpit Temperature System Components and Wiring

A multimeter (Simpson 260, or equivalent capable of 20,000 ohms per volt sensitivity) and a thermometer (N3A, or equivalent) are required for these checks. These checks should be made with the system non-energized and the cockpit air temperature control unit removed. All resistance checks are made from the rear (outboard) side of the control unit connector block. If any of these checks fails to give proper results, the appropriate check of the related component should be made as outlined in paragraph D. Remove and replace terminal cover as required.

CAUTION: IN ALL RESISTANCE MEASUREMENTS OF TEMPERATURE SENSOR, ENOUGH TIME MUST BE GIVEN THE SENSOR, ESPECIALLY IN STILL AIR, TO STABILIZE AT THE TEMPERATURE OF THE SURROUNDING AMBIENT AIR. ALSO, TO AVOID SELF-HEATING THE RESISTANCE MEASURING DEVICE USED MUST LIMIT THE CURRENT THROUGH THE SENSOR. IN THE TEMPERATURE RANGES SHOWN IN FIGURE 21-60-1 THE CURRENT SHOULD BE LIMITED TO 100 MICROAMPERES. THE CURRENT SHOULD BE LIMITED TO 10 MICROAMPERES AT TEMPERATURES BELOW 30°F.

(1) Check cockpit temperature rheostat (RDI R518) by measuring dc resistance between terminals "F" and "G" of connector block. With the COCKPIT TEMP (AUTO CONTROL - COCKPIT) temperature selector rheostat (center pedestal) in full clockwise position (HOT or WARMER) the measured resistance must be 2,250 to 2,750 ohms. With the rheostat in full counterclockwise (COLD) position, the measured resistance must be less than 150 ohms.

(2) Check cockpit air duct temperature sensor (RDI SS502) by measuring dc resistance between terminals "P" and "R" and between terminals "d" and "e" of connector block. The measured resistance between "d" and "e" should fall within the limits of curve I (figure 21-60-1). The measured resistance between "P" and "R" should fall within the limits of curve II (figure 21-60-2). Place thermometer as close to duct sensor as practical. If results are not within limits of curves, it may be necessary to remove the sensor from duct (with electrical wires connected) and repeat this check with thermometer adjacent to sensor probe.

(3) The cockpit air temperature control unit (RDI CN502) cannot be checked without additional equipment and is not interchangeable with the cabin unit (RDI CN500). If system did not operate properly during operation checks (paragraphs A or B) and results of step (1) and (2) of this paragraph are proper, replace temperature control unit or replace cockpit hot-air mixing valve and repeat paragraphs B and A.

(4) For hot-air mixing valve, measure dc resistance between sockets indicated, and check that measured values are within the following specified limits:

PIN SOCKETS	RESISTANCE VALUES
Between "M" and "E"	600 to 900 ohms
Between "K" and "E"	600 to 900 ohms
Between "N" and "E"	9 to 14 kilohms
Between "H" and "E"	0.5 to 1.5 kilohms when valve is in full-open position, or 7 to 14 kilohms when valve is in full-closed position

D. Checking Individual Cockpit Air Temperature System Components

These checks are made on individual components. If proper results are obtained here when results of the corresponding check at the control unit connector block were not satisfactory, the airplane wiring is faulty.

(1) On cockpit temperature selector rheostat (RDI R518), with system wiring disconnected (from at least one terminal) check that dc resistance between terminals "A" and "B" of rheostat is 2,250 to 2,750 ohms with the rheostat turned fully clockwise (HOT or WARMER) and is less than 150 ohms with the rheostat turned fully counterclockwise. If resistance measurements are not as required a new unit should be installed.

(2) On cockpit air duct temperature sensor (RDI SS502) with system wiring disconnected, check that dc resistance between terminals "A" and "B" is within limits of curve II (figure 21-60-2), and that resistance between terminals "C" and "D" is within limits of curve I (figure 21-60-1). If unit does not meet these requirements a new unit should be installed.

(3) On hot-air mixing valve, with system wiring disconnected, make approximate dc resistance measurements from terminals of actuator connector as follows:

TERMINALS	RESISTANCE VALUES
Between "C" and "B"	600 to 900 ohms
Between "A" and "B"	600 to 900 ohms
Between "E" and "B"	9 to 14 kilohms
Between "D" and "B" (VALVE OPEN)	Valve fully open, the resistance measured must be 900 to 1100 ohms and must steadily increase as valve goes toward closed.
Between "D" and "B" (VALVE CLOSED)	The resistance in the fully closed valve position must be between 8,500 and 12,000 ohms

CHAPTER

22

AUTO FLIGHT

NOTICE

The following information applies to NA 265-70 airplanes only.

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AUTOPILOT

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CHAPTER 22

AUTOPILOT

LIST OF EFFECTIVE PAGES

THIS ENTIRE CHAPTER HAS BEEN REVISED

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22-00-00	1*	Oct 31/86
	2*	Oct 31/86
22-00-01	201/202*	Oct 31/86

* The asterisk indicates pages revised, added, or deleted by the current revision.

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AUTOPILOT

GENERAL

A Collins AP-104 autopilot system is installed in the airplane and is integrated with a Collins FD-109H flight director providing visual displays on the pilot's instrument panel. The Collins autopilot system uses electrically operated servos, mechanically linked to the flight control system. Location of the autopilot components is shown in Figure 22-00-1.

For information concerning installation, operation, maintenance, and trouble shooting of the equipment, refer to the manuals listed in paragraph 2. (Instruction manuals combine installation, maintenance, overhaul, and illustrated parts breakdown manuals.)

AUTOPILOT EQUIPMENT MANUALS

The following manuals are available for the AP-104 autopilot system:

1. Pilot's Guide, 523-0759326
2. Maintenance and Installation Manual, 523-0759319
   - A. 161A-1B trim coupler
     - Unit Instruction Manual, 523-0756021
   - B. 161F-1 air data coupler
     - Overhaul and Illustrated Parts Breakdown Manual, 523-0759318
   - C. 332D-11 vertical gyro (dual)
     - Maintenance Manual, 523-0757988
     - Overhaul and Illustrated Parts Breakdown Manual, 523-0757989
   - D. 332G-2 yaw rate gyro
     - Instruction and Overhaul Manual, 523-0755240
     - Illustrated Parts Breakdown Manual, 523-0756052
   - E. 334C-3B primary servo and 351B-3 servo mount
     - Illustrated Parts Breakdown Manual, 523-0756054
     - Overhaul Manual, 523-0755470
   - F. 345A-4B two axis rate sensing unit
     - Instruction and Overhaul Manual, 523-0755240
     - Illustrated Parts Breakdown Manual, 523-0756054
   - G. 345C-2A acceleration sensor
     - Overhaul Manual, 523-0760121
   - H. 562C-4A autopilot amplifier
     - Overhaul and Illustrated Parts Breakdown Manual, 523-0759312
   - I. 590-D1 air data computer
     - Overhaul and Illustrated Parts Breakdown Manual, 523-0761478
   - J. 614E-4 autopilot controller
     - Overhaul and Illustrated Parts Breakdown Manual, 523-0759315

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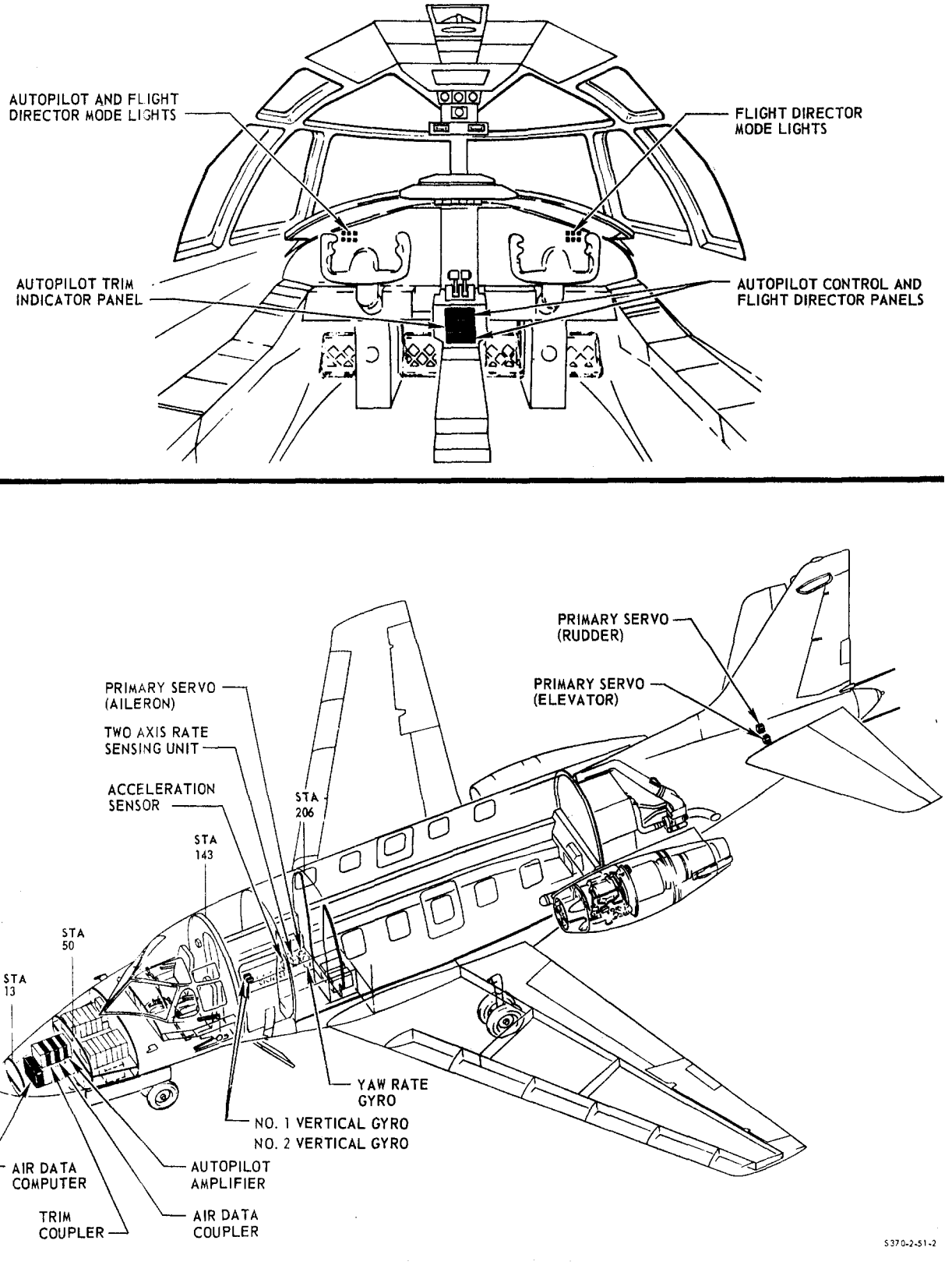

Figure 22-00-1 — Autopilot System Unit Locations — S/N 370-1 and Subs

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AUTOPILOT

MAINTENANCE PRACTICES

CHECKING AUTOPILOT SERVO TORQUE ADJUSTMENTS

The autopilot primary servos are set at the proper torque values when initially installed. Should it become necessary to replace any of the servo units, the torque values will have to be adjusted.

Adjust primary servo torque settings as follows:

1. Apply 28 VDC power to pin "A," and ground pin "B" of servo electrical connector (MS3102A-10SL-3P).
2. Lock gears by inserting screwdriver into hole provided in servo.
3. Using a calibrated torque wrench, certified to 1-1/2 inch-pound accuracy, apply a very slow pull on torque wrench, and check that clutch breakout is at the following values:
   A. Aileron servo 65 (±2) inch-pounds
   B. Elevator servo 130 (±2) inch-pounds
   C. Rudder servo 175 (±2) inch-pounds
4. If value is not within preceding limits, adjust breakout torque in accordance with Collins Autopilot Maintenance and Installation Manual, 523-0759319.
5. After adjustment is completed, identify channel ("AIL," "ELE," or "RUD") and record torque setting on name plate of unit.
6. Refer to chapter 27 for Autopilot System cable tensions.

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NOTICE

The following information applies to NA 265-80 airplanes only.

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AUTOPILOT

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	2*	Oct 31/86
22-00-01	201/202*	Oct 31/86

* The asterisk indicates pages revised, added, or deleted by the current revision.

NEW

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AUTOPILOT

GENERAL

A Collins FCS-105 autopilot system is installed in the airplane and is integrated with a Collins FD-109Z flight director providing visual displays on the pilot's instrument panel. The Collins autopilot system uses electrically operated servos, mechanically linked to the flight control system. Location of the autopilot components is shown in Figure 22-00-1.

For information concerning installation, operation, maintenance, and trouble shooting of the equipment, refer to the manuals listed in paragraph 2. (Instruction manuals combine installation, maintenance, overhaul, and illustrated parts breakdown manuals.)

AUTOPILOT EQUIPMENT MANUALS

The following manuals are available for the FCS-105 autopilot system:

1. Pilot's Guide, 523-0763450
2. Instruction Book, 523-0762755
   - A. 161E-2A mode coupler
     - Maintenance and Illustrated Parts Breakdown Manual, 523-0762683
   - B. 332D-11A vertical gyro (dual)
     - Overhaul and Illustrated Parts Breakdown Manual, 523-0757989
   - C. 562C-8A yaw damper
     - Overhaul and Illustrated Parts Breakdown Manual, 523-0762918
   - D. 334C-6B primary servo and 351B-6B/7C servo mount
     - Overhaul and Illustrated Parts Breakdown Manual, 523-0672690
   - E. 562A-5M5 flight computer
     - Maintenance and Illustrated Parts Breakdown Manual, 523-0763110
   - F. 562C-5 autopilot amplifier
     - Overhaul Manual, 523-0764051 (Volume 1)
     - Illustrated Parts Breakdown Manual, 527-0770027 (Volume 2)
   - G. 590A-3H air data control
     - Overhaul and Illustrated Parts Breakdown Manual, 523-0764343
   - H. 614E-5A autopilot controller
     - Overhaul and Illustrated Parts Breakdown Manual, 523-0762796
   - I. 590B-2 airspeed sensor
     - Maintenance and Illustrated Parts Breakdown Manual, 523-0758589

NEW

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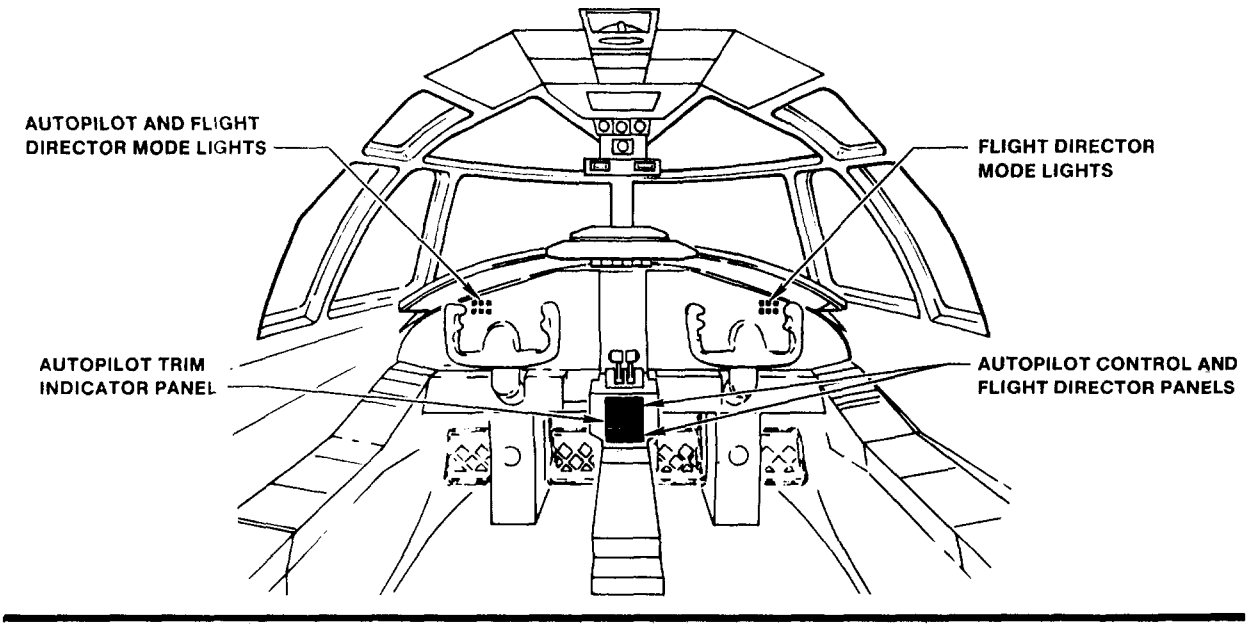
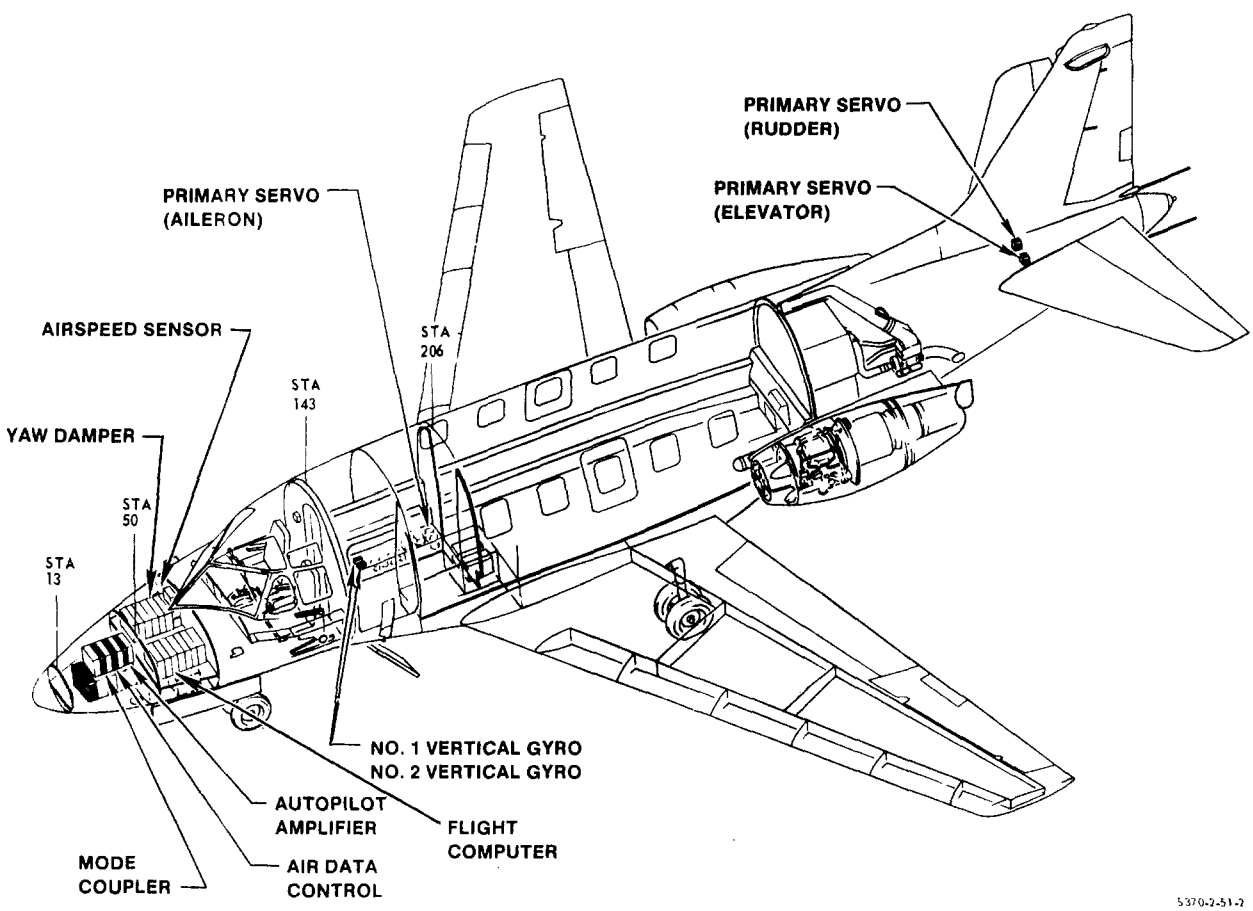

AUTOPILOT AND FLIGHT DIRECTOR MODE LIGHTS
FLIGHT DIRECTOR MODE LIGHTS
AUTOPILOT TRIM INDICATOR PANEL
AUTOPILOT CONTROL AND FLIGHT DIRECTOR PANELS



NEW Figure 22-00-1 — Autopilot System Unit Locations — S/N 380-1 and Subs

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AUTOPILOT

MAINTENANCE PRACTICES

CHECKING AUTOPILOT SERVO TORQUE ADJUSTMENTS

Refer to Collins Service Information Letter 3-75,R2 (11/15/76) for any service or adjustments to the autopilot.

To order, contact: Collins General Aviation Avionics Division
Attn: Service Parts Order Administration
1100 Hibiscus, m/s 310-100
PO Box 1060
Melbourne, FL 32901

TELEPHONE: (800) 327-0854
or
(305) 768-7230 (West Coast)
(305) 768-7236 (Midwest)
(305) 768-7232 (East Coast)

TELEX: 671-3069

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CHAPTER 23

COMMUNICATIONS

CHAPTER 23

COMMUNICATIONS

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COMMUNICTIONS

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23-60-01		401/402		Oct 5/92
		403		Mar 15/83
		404		Mar 15/83
		405/406		Mar 15/83
		601		Mar 1/70
	Added	602	*	Nov 12/93
	Added	603/604	*	Nov 12/93

* The asterisk indicates pages revised, added, or deleted by the current revision.

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COMMUNICATIONS - DESCRIPTION AND OPERATION

1. General

The communication system consists of dual Collins Airborne Master Audio Control Centers and Collins VHF dual transceivers.

The intercommunication system provides a means of transceiver selection and communication between the pilot and copilot and the passenger compartment.

The VHF system provides airplane-to-ground and airplane-to-airplane communication. For information concerning the maintenance and operation of these systems, the following manuals are available:

2. Communications Equipment Manuals

A. 618M-2D VHF Transceiver

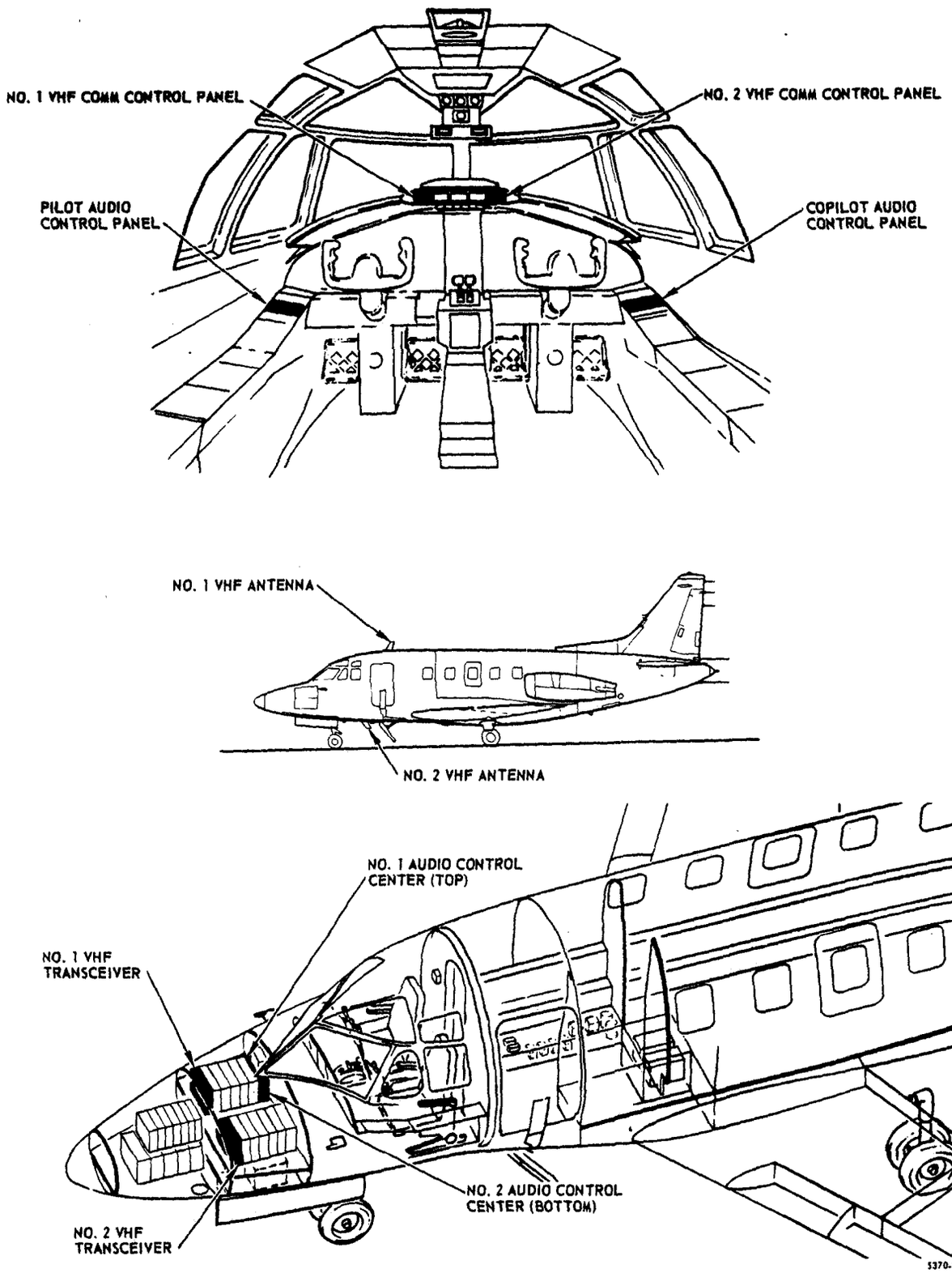
Maintenance and Installation Manual 523-0758 989

Overhaul and Illustrated Parts Breakdown Manual 523-0758 990

B. 346B-3 Airborne Master Audio Control Center

Maintenance and Installation Manual 523-0756 760

Overhaul and Illustrated Parts Breakdown Manual 523-0756 882


NO. 1 VHF COMM CONTROL PANEL
NO. 2 VHF COMM CONTROL PANEL
PILOT AUDIO CONTROL PANEL
COPILOT AUDIO CONTROL PANEL
NO. 1 VHF ANTENNA
NO. 2 VHF ANTENNA
NO. 1 AUDIO CONTROL CENTER (TOP)
NO. 1 VHF TRANSCEIVER
NO. 2 AUDIO CONTROL CENTER (BOTTOM)
NO. 2 VHF TRANSCEIVER
5370-2-71-1

Communication System Unit Location
Figure 23-00-1

STATIC DISCHARGER - DESCRIPTION

1. General

Precipitation static is dissipated by static dischargers. Static dischargers are installed on the aft end of the two lightning diverter rods attached to the aft end of the fuselage rear section, one above and one below the fuel vent outlet. Static dischargers are also installed on the horizontal stabilizers, elevators, vertical stabilizer, rudder, and wing tips.

STATIC DISCHARGER

MAINTENANCE PRACTICES

REMOVING STATIC DISCHARGER

Procedure

1. Remove AN565B615 setscrew from end of retainer.
2. Lift static discharger from retaining recess.

INSTALLING STATIC DISCHARGER

Procedure

1. Position static discharger in recess.
2. Install setscrew in threaded hole.
3. Check that static discharger is firmly mounted and rigid.

INSTALLING STATIC DISCHARGER RETAINER(S)

Procedure

1. Mark location of discharger retainer(s) on airplane surface. Do not exceed 0.09 inch between trailing edge discharger retainers and trailing edges.
2. Remove all foreign material, including paint, from airplane surface and bottom of discharger retainer(s).
3. Clean airplane surface and bottom of discharger retainer(s) with MEK, Acetone, or equivalent.
4. Mix resin and hardener thoroughly until mixture is smooth. Use Dayton-Granger, Inc. adhesive kit (P/N 15348).

NOTE

Pot life of adhesive after mixing is one to two hours at 70°F.

5. Clean both areas to be bonded with 400 to 600 grit silicone carbide paper. Clean until all oxide is removed.
6. Apply a coat of adhesive cement to cleaned surfaces within five minutes after completion of step 5.
7. Position retainer and conductive gasket, P/N 16181, on surface and twist slightly while applying moderate pressure.
8. Remove excess adhesive, except for a fillet, around the entire surface of mounting base with a clean cloth dampened with trichloroethylene.

NOTE

No cracks should be visible if airplane surface does not exactly conform to mounting base.

9. Allow adhesive to cure at one of the following intervals:
   a. 75°F for 24 hours
   b. 120°F for 4 hours
   c. 150°F for 1 hour
   d. 200°F for 1/2 hour
10. Apply a fillet of sealing compound around entire edge of mounting base. Use Product Research and Chemical Corporation (Burbank, CA) sealing compound (PRC-341).

CAUTION

Rudder and elevator may need rebalancing after reinstallation of static discharger retainer(s).

NOTES

- ***Rivets must be installed after sealant has cured to insure electrical bond between retainer and trailing edge skin thru rivets.***
- ***Use applicable portion of Figure 23-60-1 for rivet installation.***

11. Drill rivet holes using an 1/8 in. (0.125) drill.
12. Install flush head rivets (NAS 1399B4-2) as required.
13. Refinish surface per procedures in the Corrosion Control Manual, Report No. SR-76-023.

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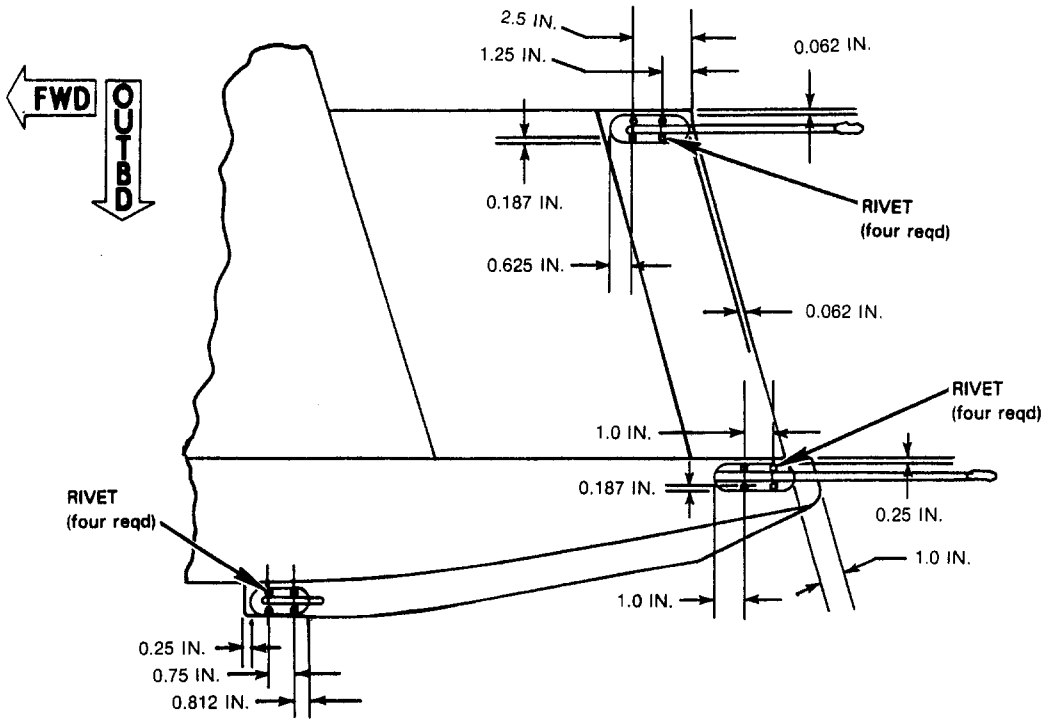

WING TIP
(LH SHOWN)

MAJOR CHANGE

Installation of Static Discharger Retainers
Figure 23-60-1 (Sheet 1 of 3)

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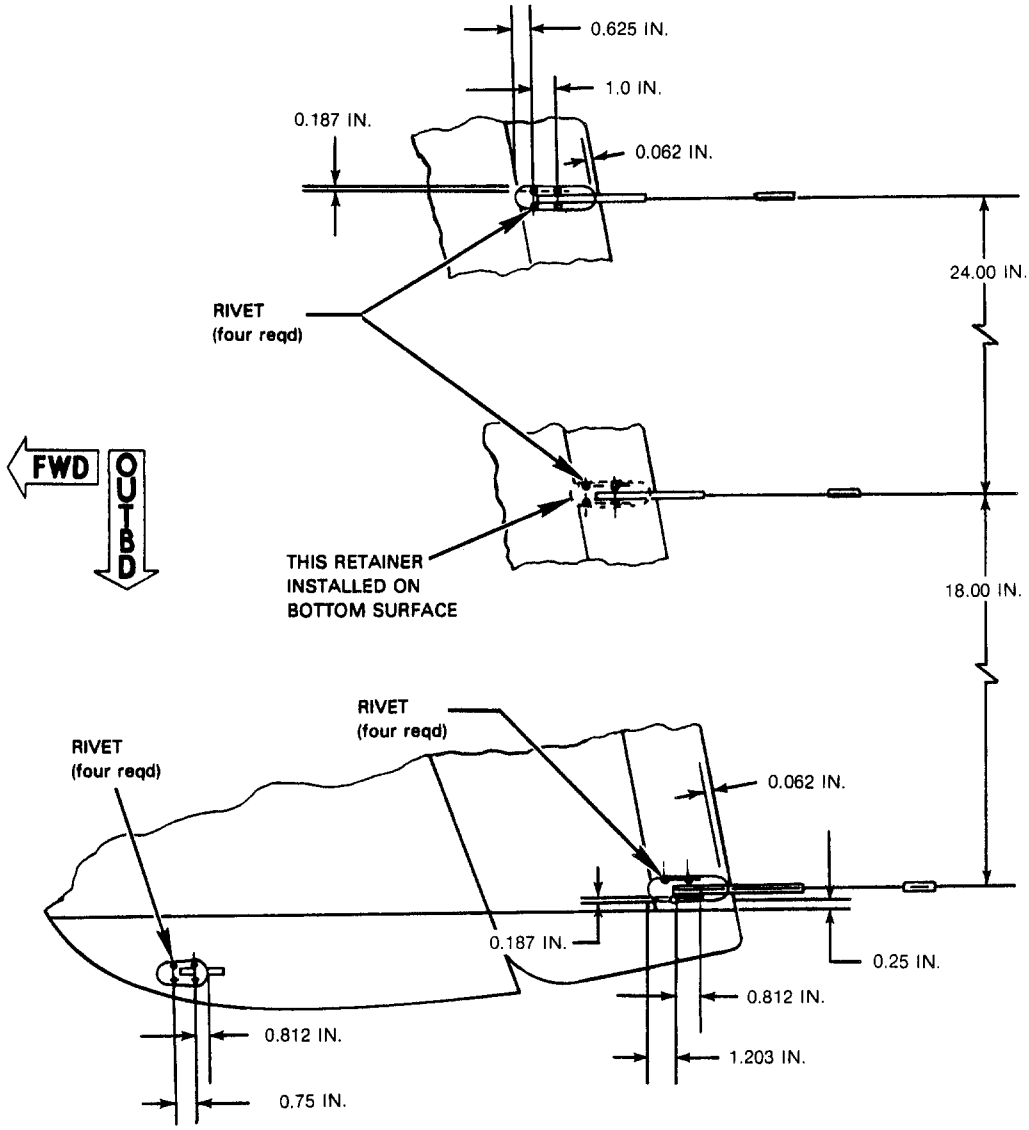

ELEVATOR
(LH SHOWN)

MAJOR CHANGE

Installation of Static Discharger Retainers
Figure 23-60-1 (Sheet 2 of 3)

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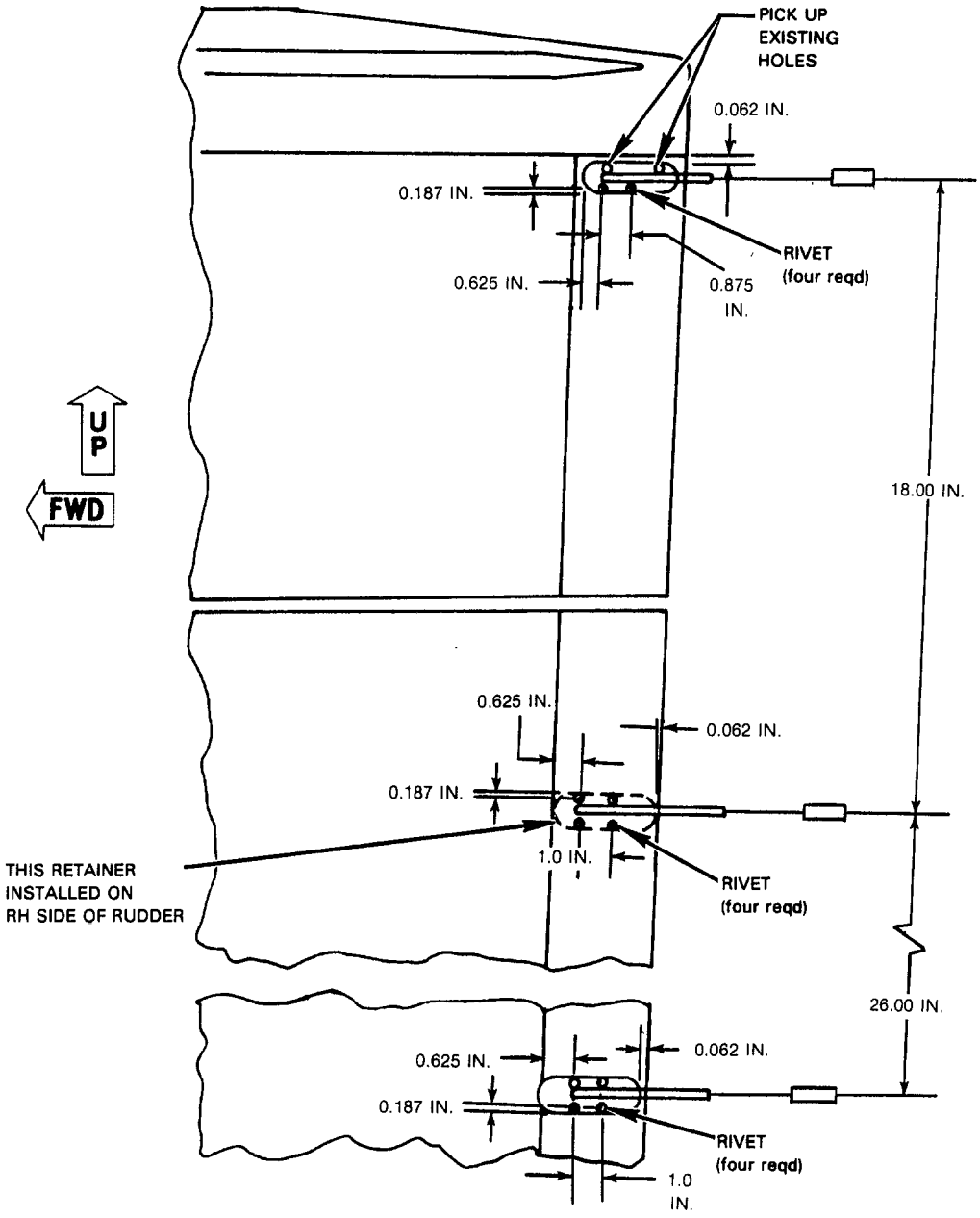

**VERTICAL STABILIZER
AND RUDDER
(LH SIDE SHOWN)**

MAJOR CHANGE

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1. Checking Static Dischargers for Resistance

A. With a megohmmeter, measure resistance of static dischargers between discharger pin and airplane structure; and replace the ones that are not within tolerance as shown in the following schedule:

LENGTH	DISCHARGER	TOLERANCE
Long	NFD-610-1011 (trailing)	8 to 100 megohms
Short	NFD-611-1008 (tip)	5 to 60 megohms

CAUTION: USE TEST CLIPS WITH CARE, AS DISCHARGER POINTS MAY BE EASILY BENT OR DAMAGED.

B. On any discharger retainers that show any evidence of cracking or burning, or which have been reinstalled, check that resistance of adhesive bond between retainer and airplane structure is less than 0.5 ohm. Check condition of sealant covering adhesive bond lines. Repair as necessary and reseal if any exposed areas are noted.

2. Checking Static Dischargers for Possible Replacement

A. The following discrepancies are cause for replacement of the static dischargers:

(1) Broken pins (broken plastic guards do not affect performance).

(2) Blunted or bent pins. Check as if pin were a phonograph needle.

(3) Lightning damage - burning and roughening of the discharger black conductive coating and/or pitting of the discharger retainer. Resistance tolerances must be lower than the tolerances noted in paragraph 1.

(4) Excessive erosion or peeling of the tip cap. Leading edge erosion of the airfoil tip dischargers should not extend back over more than one-third the width of the discharger.

MAINTENANCE PRACTICES (Continued)

REPAIR — H.F. COMM. ANTENNA (WING TO NACELLE)

Equipment/Tools

Torch-appropriate for Silver Solder application
Kearney Type I portable Swaging Tool (or equivalent)

Consumables

1 ea. — MS20664C4 Ball

1 ea. — MS25036-12 Terminal Lug

48 in. — 1/8" Diameter, 6 x 19, Phosper Bronze-Fiber Core Cable

1 ea. — .032 Phosphor Bronze Sheet Mat'l: Comp A, Cond: Soft QQ-B-750 (See Figure 23-60-1).

Solder — MIL-S-45743.

Procedure

NOTE

This repair is to be made only if a minimum 6 inches of un-frayed antenna cable remains: measured from the wing mast head to the point of splice.

1. Silver solder broken end of cable and mating end of replacement cable to prevent fraying and unwinding when cut.
2. Trim cable ends to accept splice.

CAUTION

During soldering, care should be taken to avoid excessive heating that may damage fiber core of cable.

3. Mate cable ends within splice and solder in place. Check splice with Milliohmeter for resistance.

   Load test cable (See 2 on Figure 23-60-1)
4. Thread loose cable end thru hole in nacelle door assy. Pull cable as tight as possible and mark where it enters nacelle door with a turn or two of narrow masking tape.
5. Remove nacelle door from nacelle. Determine length of cable, from the taped mark, required to accommodate the tensioner assy (60M-144-2797-035) See Figure 23-60-1.
6. Subtract .65 inch from this total antenna measurement and mark cable for swage ball position.
7. Remove tape marking place where cable enters nacelle door. Apply silver solder to an area of the cable extending 1 inch to either side of the mark (See Figure 23-60-1)
8. Thread cable thru door, and thru tensioner assy.
9. Swage ball into position (See Figure 23-60-1) using Kearney Type I portable Swaging Tool (or equivalent.)
10. Crimp MS25036 Terminal Lug on cable end, after correct cable length has been determined.
11. Install Terminal Lug on nut plate provided in nacelle.
12. Install Nacelle Door.

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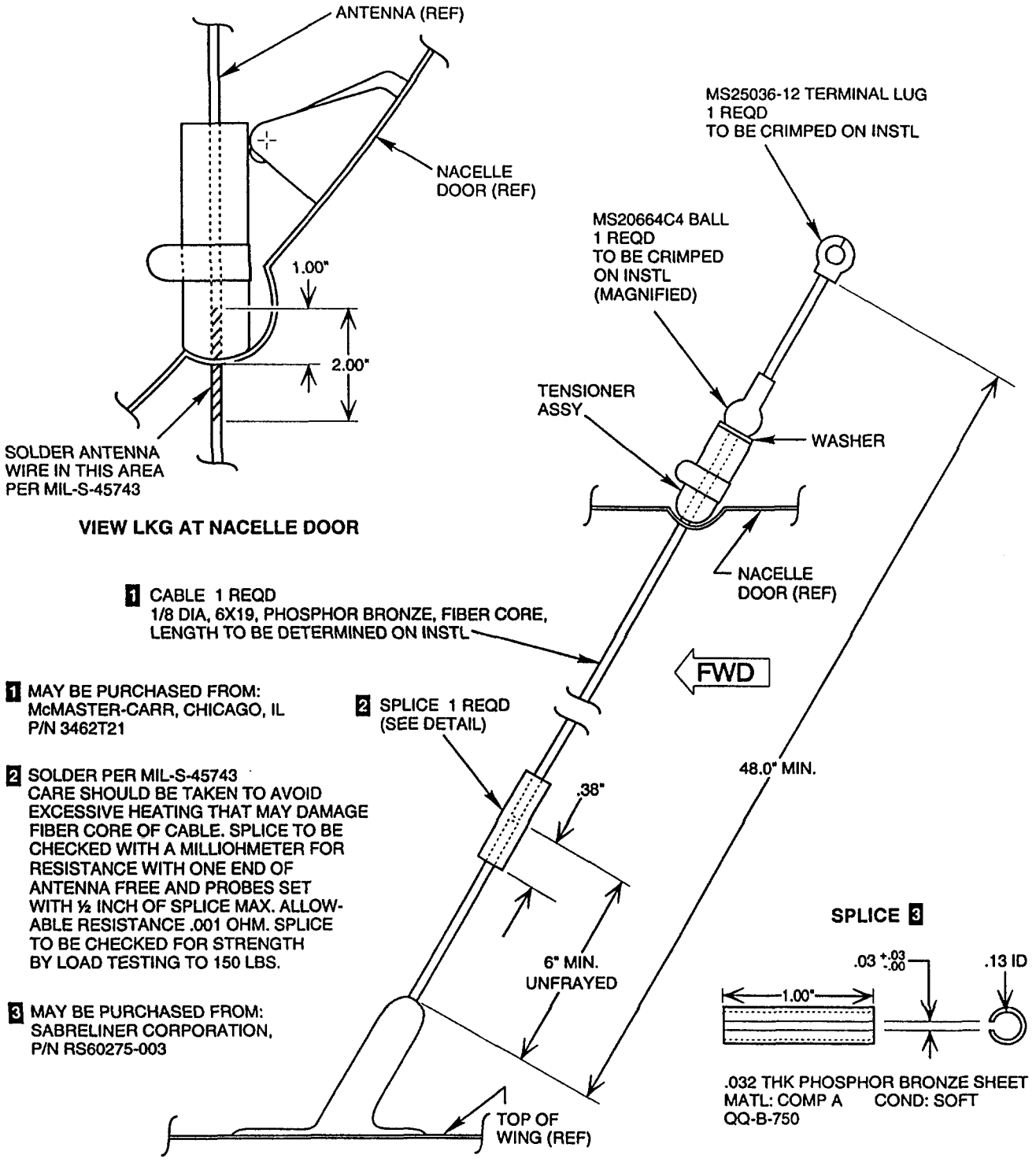
ANTENNA (REF)
NACELLE DOOR (REF)
1.00"
2.00"
SOLDER ANTENNA WIRE IN THIS AREA PER MIL-S-45743
VIEW LKG AT NACELLE DOOR
MS25036-12 TERMINAL LUG 1 REQD TO BE CRIMPED ON INSTL
MS20664C4 BALL 1 REQD TO BE CRIMPED ON INSTL (MAGNIFIED)
TENSIONER ASSY
WASHER
NACELLE DOOR (REF)
1 CABLE 1 REQD 1/8 DIA, 6X19, PHOSPHOR BRONZE, FIBER CORE, LENGTH TO BE DETERMINED ON INSTL
FWD
1 MAY BE PURCHASED FROM: McMASTER-CARR, CHICAGO, IL P/N 3462T21
2 SPLICE 1 REQD (SEE DETAIL)
2 SOLDER PER MIL-S-45743 CARE SHOULD BE TAKEN TO AVOID EXCESSIVE HEATING THAT MAY DAMAGE FIBER CORE OF CABLE. SPLICE TO BE CHECKED WITH A MILLIOHMETER FOR RESISTANCE WITH ONE END OF ANTENNA FREE AND PROBES SET WITH ½ INCH OF SPLICE MAX. ALLOWABLE RESISTANCE .001 OHM. SPLICE TO BE CHECKED FOR STRENGTH BY LOAD TESTING TO 150 LBS.
3 MAY BE PURCHASED FROM: SABRELINER CORPORATION, P/N RS60275-003
.38"
6" MIN. UNFRAYED
48.0" MIN.
TOP OF WING (REF)
SPLICE 3
.03 +.03 -.00
.13 ID
1.00"
.032 THK PHOSPHOR BRONZE SHEET
MATL: COMP A COND: SOFT
QQ-B-750

Figure 23-60-1 — Repair of H.F. Comm. Antenna

NEW

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CHAPTER 24

ELECTRICAL POWER

CHAPTER 24

ELECTRICAL POWER

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CHAPTER 24

ELECTRIC POWER

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ELECTRICAL POWER - DESCRIPTION

1. General

The electrical power systems comprise the ac generation, dc generation, external power, and electrical load distribution (ac and dc).

The ac generation system is made up of a No. 1 inverter, a No. 2 inverter, an instrument inverter, and the necessary relays and controls to provide regulated ac power for use in various systems. (For more detailed information, refer to text covering description and operation of this system.)

The dc generation system supplies dc power for use throughout the airplane. During normal operation, this power is furnished by two dc generator systems. During an emergency, a limited amount of dc power is supplied by the battery system. (For more detailed information, refer to text covering description and operation of this system.)

The external power system consists of an engine start external power receptacle, a utility external power receptacle, an external power cutout relay, and necessary relays and control circuitry. No provision is made for the application of ac external power. (For more detailed information, refer to text covering description and operation of this system.)

The electrical load distribution systems (ac and dc) are made up of busses, tie-in and control relays, switches, and circuit breakers to provide control of the distribution of power to the using systems. (For more detailed information, refer to text covering description and operation of these systems.)

AC GENERATION - DESCRIPTION AND OPERATION

1. AC Power Generation System - NA-370 Airplanes

The ac power electrical system consists of a 2500-volt-ampere No. 1 inverter, a 2500-volt-ampere No. 2 inverter, a 750-volt-ampere instrument inverter, an inverter change-over control relay, an ac transfer relay, and associated failure relays and control circuitry. (See figures 24-20-1 and 24-20-2.) During normal operation ("INVERTER" switch at NO. 1), the No. 1 inverter supplies delta-connected ac power to all of the ac-operated equipment in the airplane. Should the No. 1 inverter fail during normal operation, the inverter change-over control relay automatically selects the No. 2 inverter. Manual selection of the No. 2 inverter is done by placing the "INVERTER" switch at NO. 2. When selected (either manually or automatically), the No. 2 inverter also supplies delta-connected ac power to all of the ac-operated equipment in the airplane. Should the No. 2 inverter then fail, the instrument inverter will be automatically started to supply delta-connected ac power to the primary (instrument) and secondary ac busses.

NOTE: The 2500-volt-ampere inverters may have been replaced with 3000-volt-ampere inverters.

The No. 2 inverter and instrument inverter also provide ac power to the windshield anti-icing system. During normal operation, when the No. 1 inverter is powering the airplane ac distribution system, the No. 2 inverter powers the windshield anti-icing system. If the No. 1 inverter should fail and the No. 2 inverter is required to power the airplane systems, the automatic change-over occurs, causing the No. 2 inverter to power the airplane systems and the instrument inverter to power the windshield anti-icing system to one windshield only, depending upon the position of the selector switch. Also any time the No. 2 inverter is selected by the "INVERTER" switch, while the windshield anti-icing system is on, the instrument inverter will start automatically to power the windshield anti-icing system. The No. 2 inverter can be turned on either by the "INVERTER" switch or by the "FWD WSHIELD" switch.

During normal operation, the "INVERTER" switch is at NO. 1. (See figure 24-20-2.) This connects dc power to the "SW" terminal on the inverter change-over control relay. The "SW" terminal routes the dc power to the No. 1 inverter power relay coil, through a set of normally closed contacts on the K3 relay inside the inverter change-over control relay. (The control relay contains three relays. They are referred to as K1, K2 and K3 for identification purposes.) The energized No. 1 inverter power relay connects essential bus power to the No. 1 inverter, which becomes operational and produces delta-connected ac power to the essential A-phase bus and the instrument bus, through normally closed contacts on the ac transfer relay.

When a No. 1 inverter power failure occurs, or when the A-phase output voltage of the No. 1 inverter drops below 90 volts ac, K1 relay inside the inverter change-over control relay becomes de-energized. When de-energized, K1 relay connects dc power to the heater element through normally closed contacts of the K3 relay. Heat from the heater element actuates the thermoswitch after a delay of about 5 seconds. When actuated, the thermoswitch connects dc power to the coil of the K3 relay, and to the coil of the ac

transfer control relay. The energized K3 relay locks itself in the energized position and at the same time locks in the ac transfer control relay. When locked, the K3 relay also disconnects power from the heater element and from No. 1 inverter power relay. (After about 10 seconds delay time, the thermoswitch contacts open; however, relay K3 remains locked and thereby keeps the ac transfer control relay energized.) The energized ac transfer control relay connects power to the coil of the ac transfer relay and the No. 2 inverter power relay, causing these relays to energize.

The energized No. 2 inverter power relay connects dc power to the No. 2 inverter, which becomes operational. The energized ac transfer relay transfers the primary (instrument) and secondary bus ac loads to the output of the No. 2 inverter.

The No. 1 inverter fail relay becomes de-energized when the No. 1 inverter fails, connecting dc power to energize the inverter fail light control relay and cause the "#1 INV OFF" light to come on.

Should the No. 2 inverter also fail, the instrument inverter control relay becomes de-energized and connects dc power to energize the instrument ac bus control relay and instrument inverter power relay. The instrument inverter becomes operational and its output is directed to the ac instrument bus.

When the No. 2 inverter fails, the No. 2 inverter fail relay is de-energized, connecting dc power to the "#2 INV OFF" light, causing it to come on.

The instrument power fail relay becomes de-energized when the No. 1 or No. 2 inverter fails, and connects dc power to the "AC INST PWR OFF" light, illuminating the light.

When the No. 2 or instrument inverter becomes operational, the instrument power fail relay becomes energized and disconnects dc power from the indicator light. The light goes out, indicating that ac power is available on the ac instrument bus.

When the "INVERTER" switch is placed at NO. 2, dc power is connected directly to the coil of the ac transfer control relay. The ac transfer control relay, when energized, connects power to the coils of the ac transfer relay and the No. 2 inverter power control relay. This causes the circuit to operate in the same manner as previously outlined for the automatic function of the No. 2 inverter power relay, the ac transfer relay, and the instrument power fail relay. Placing the "INVERTER" switch at OFF disconnects dc power from the control circuitry and shuts down all inverters. The de-energized No. 1 inverter fail, No. 2 inverter fail, and instrument power fail relays will connect dc power to the "#1 INV OFF", "#2 INV OFF", and "AC INST PWR OFF" lights, causing them to come on.

2. AC Power Generation System - NA-380 Airplanes

The ac power electrical system consists of a 3000-volt-ampere No. 1 inverter, a 3000-volt-ampere No. 2 inverter, a 750-volt-ampere instrument inverter, an inverter change-over control relay, an ac transfer relay, and associated failure relays and control circuitry. (Earlier airplanes may have 2500-volt-ampere No. 1 and No. 2 inverters.) (See figures 24-20-3, 24-20-4 or 24-20-5.)

During normal operation (with either engine operating or external electrical power applied to airplane and the "INVERTER" switch at NO. 1), the No. 1 inverter supplies (delta-connected*) ac power to all of the ac-operated equipment in the airplane. Should the No. 1 inverter fail during normal operation, the inverter change-over control relay automatically selects the No. 2 inverter. Manual selection of the No. 2 inverter is done by placing the "INVERTER" switch at NO. 2. When selected (either manually or automatically), the No. 2 inverter also supplies (delta-connected*) ac power to all of the ac-operated equipment in the airplane. Should the No. 2 inverter fail, the instrument inverter will be automatically started to supply delta-connected ac power to the ac primary bus.

The No. 2 inverter and instrument inverter also provide ac power to the windshield anti-icing system. During normal operation, when the No. 1 inverter is powering the airplane ac distribution system, the No. 2 inverter powers the windshield anti-icing system. If the No. 1 inverter should fail and the No. 2 inverter is required to power the airplane systems, the automatic change-over occurs, causing the No. 2 inverter to power the airplane systems and the instrument inverter to power the windshield anti-icing system to one windshield only, depending upon the position of the selector switch. Also, any time the No. 2 inverter is selected by the "INVERTER" switch while the windshield anti-icing system is on, the instrument inverter will start automatically to power the windshield anti-icing system. The No. 2 inverter can be turned on either by the "INVERTER" switch or by the "FWD WSHIELD" switch.

With external electrical power applied to the airplane or either dc generator operating (see figure 24-20-2), when the "INVERTER" switch is positioned at NO. 1, 28 VDC power is routed to the "SW" terminal on the inverter change-over control relay (CN508). From the switch terminal, electrical power is routed through the closed K3 and K1 contacts which energizes relay K2 and the No. 1 inverter power relay (K1067) coil. When relay K2 is energized, inverter ac power is routed to the ac sensitive relay K1 which is then energized and in turn removes power from the heater circuit. The No. 1 inverter is now operational and supplies ac power to the primary and secondary buses.

When a No. 1 inverter power failure occurs, or when the output voltage of the No. 1 inverter drops below 90 volts ac, relay K1 (an ac sensitive relay) inside the inverter change-over control relay becomes de-energized. When de-energized, relay K1 connects dc power to the heater element through normally closed contacts of the K3 relay. Heat from the heater element actuates the thermoswitch after a delay of about 5 seconds. When actuated, the thermoswitch connects dc power to the coil of the K3 relay, and to the coil of the No. 1 ac transfer control relay (K1051). The energized K3 relay locks itself in the energized position and at the same time locks in relay K1051. When locked, the K3 relay also disconnects power from the heater element. After about 10 seconds delay time, the thermoswitch contacts open; however, relay K3 remains locked and thereby keeps relay K1051 energized. The energized No. 1 ac transfer control relay connects power to the coil of the ac transfer relay (K600) and the No. 2 inverter power relay (K1225), causing these relays to energize.

*Airplanes without solid state inverters

The energized No. 2 inverter power relay (K1225) connects dc power to the No. 2 inverter, which becomes operational. The energized ac transfer relay (K600) transfers the secondary (A-phase*) bus and primary (A and C-phase*) bus ac loads to the output of the No. 2 inverter.

The No. 1 inverter fail relay (K646) becomes de-energized when the No. 1 inverter fails, connecting dc power to the "#1 INV OFF" light which illuminates the light.

Should the No. 2 inverter also fail, the instrument inverter control relay (K1003) becomes de-energized and connects dc power to energize the instrument ac bus control relay (K1008) and instrument inverter power relay (K1066). The instrument inverter becomes operational and the AØ output is routed to the primary bus.

When the No. 2 inverter fails, the No. 2 inverter fail relay (K1004) is de-energized, connecting dc power to the "#2 INV OFF" light, causing it to come on.

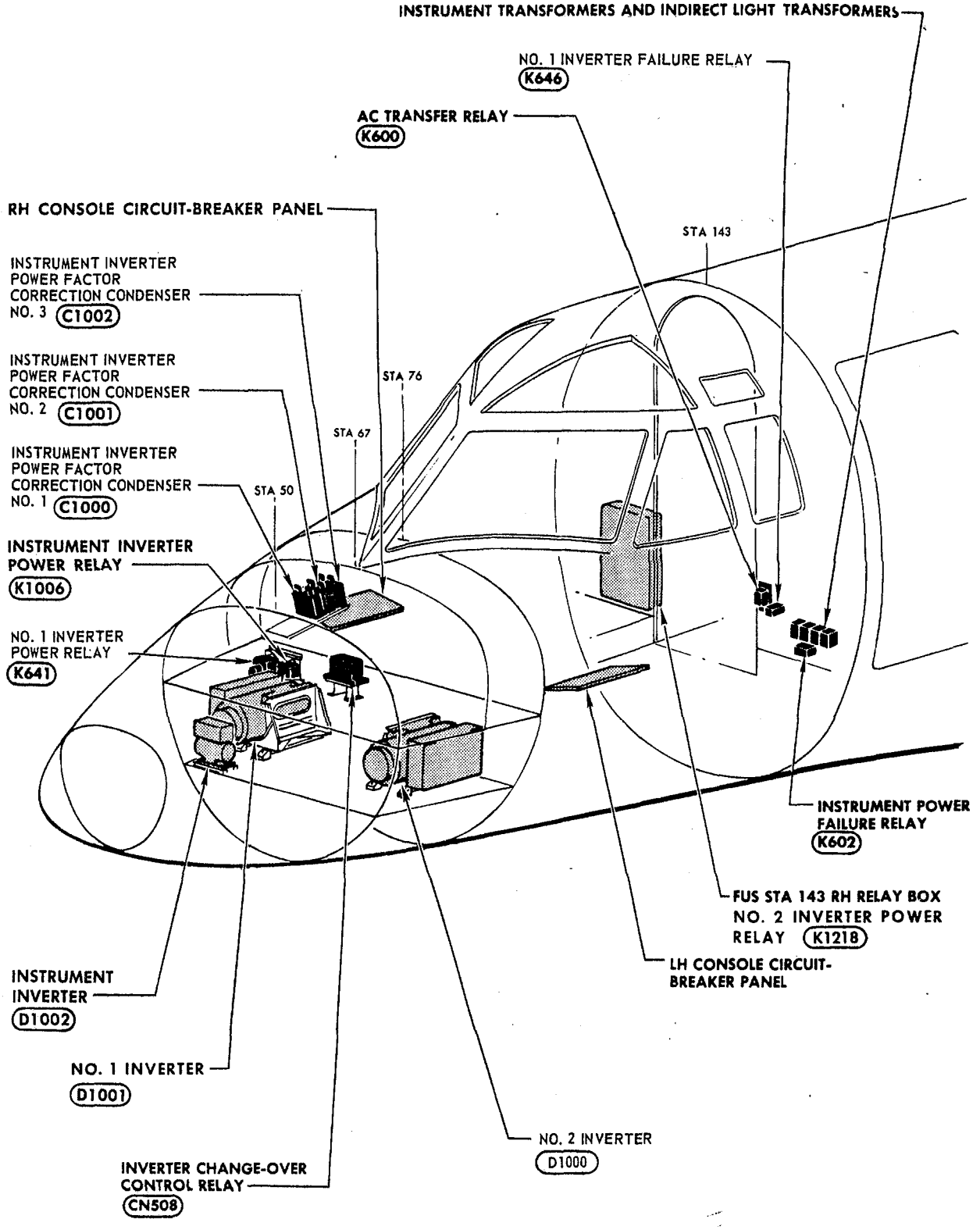
The instrument power fail relay (K602) becomes de-energized when the No. 1 or No. 2 inverter fails, and connects dc power to the "AC INST PWR OFF" light, illuminating the light.

When the No. 2 or instrument inverter becomes operational, the instrument power fail relay becomes energized and disconnects dc power from the indicator light. The light goes out, indicating that ac power is available on the ac primary bus.

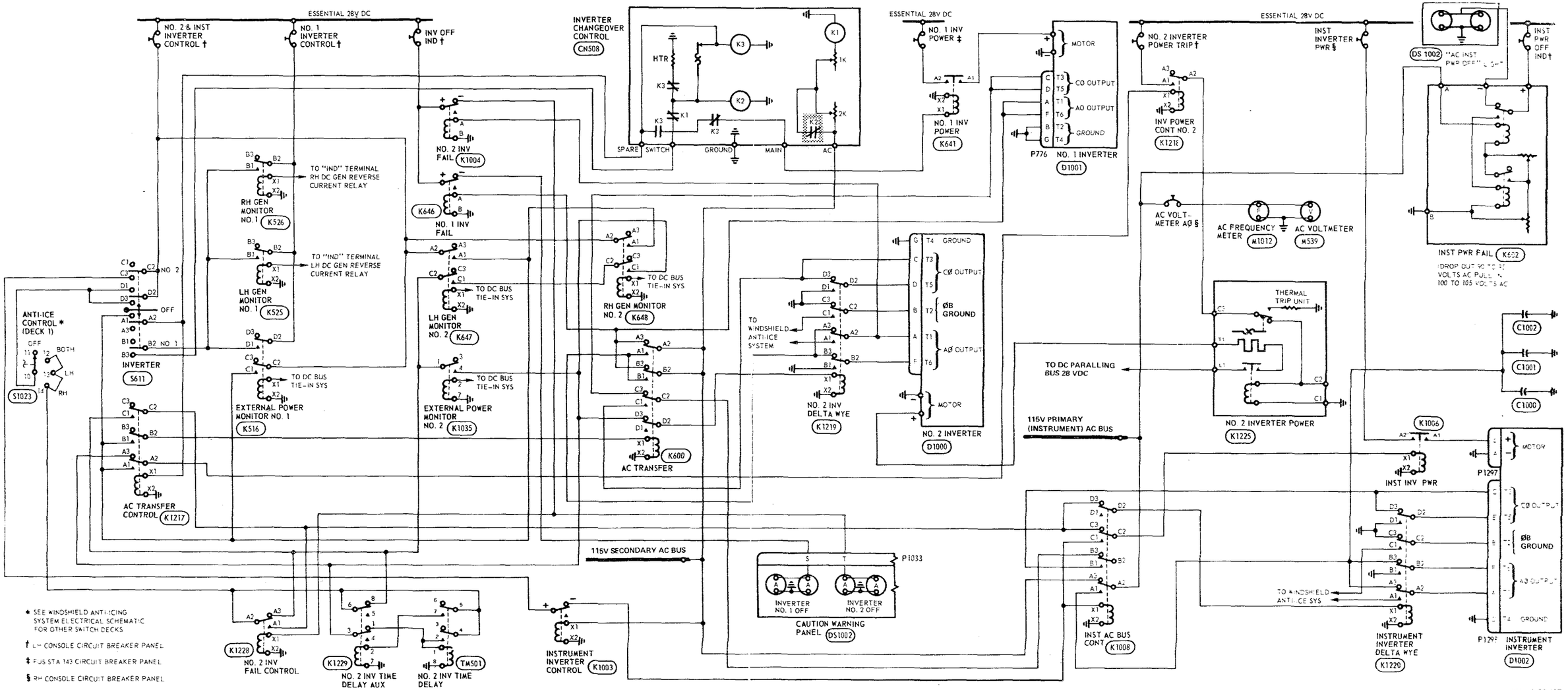
When the "INVERTER" switch is placed at NO. 2, dc power is connected directly to the coil of the No. 1 ac transfer control relay (K1051). The No. 1 ac transfer control relay (K1051), when energized, connects power to the coils of the ac transfer relay (K600) and the No. 2 inverter power control relay (K1225). This causes the circuit to operate in the same manner as previously outlined for the automatic function of the No. 2 inverter power relay, the ac transfer relay, and the instrument power fail relay. Placing the "INVERTER" switch at OFF disconnects dc power from the control circuitry and shuts down all inverters. The de-energized No. 1 inverter fail, No. 2 inverter fail, and instrument power fail relays will connect dc power to the "#1 INV OFF," "#2 INV OFF," and "AC INST PWR OFF" lights, causing them to come on.

With the No. 1 and No. 2 inverter (windshield heat selected) operating, and in the event that a dc generator fails, the No. 2 inverter will fail. When the No. 2 inverter fails, the No. 2 inverter fail control relay (K1228) energizes. With relay K1228 energized, the instrument inverter power relay (K1006) is energized, subsequently energizing the instrument inverter. With the instrument inverter operating and the windshield selector at "BOTH", "RH", or "LH", the left hand windshield is provided with half power.

*Airplanes without solid state inverters



AC Generation System Equipment Location -
NA-370 Airplanes
Figure 24-20-1



S370-2-54-12B

SEE BLOW-UP FICHE NO. ASE0411-ITEM I

AC Generation System Schematic
S/N 370 Airplanes
Figure 24-20-2

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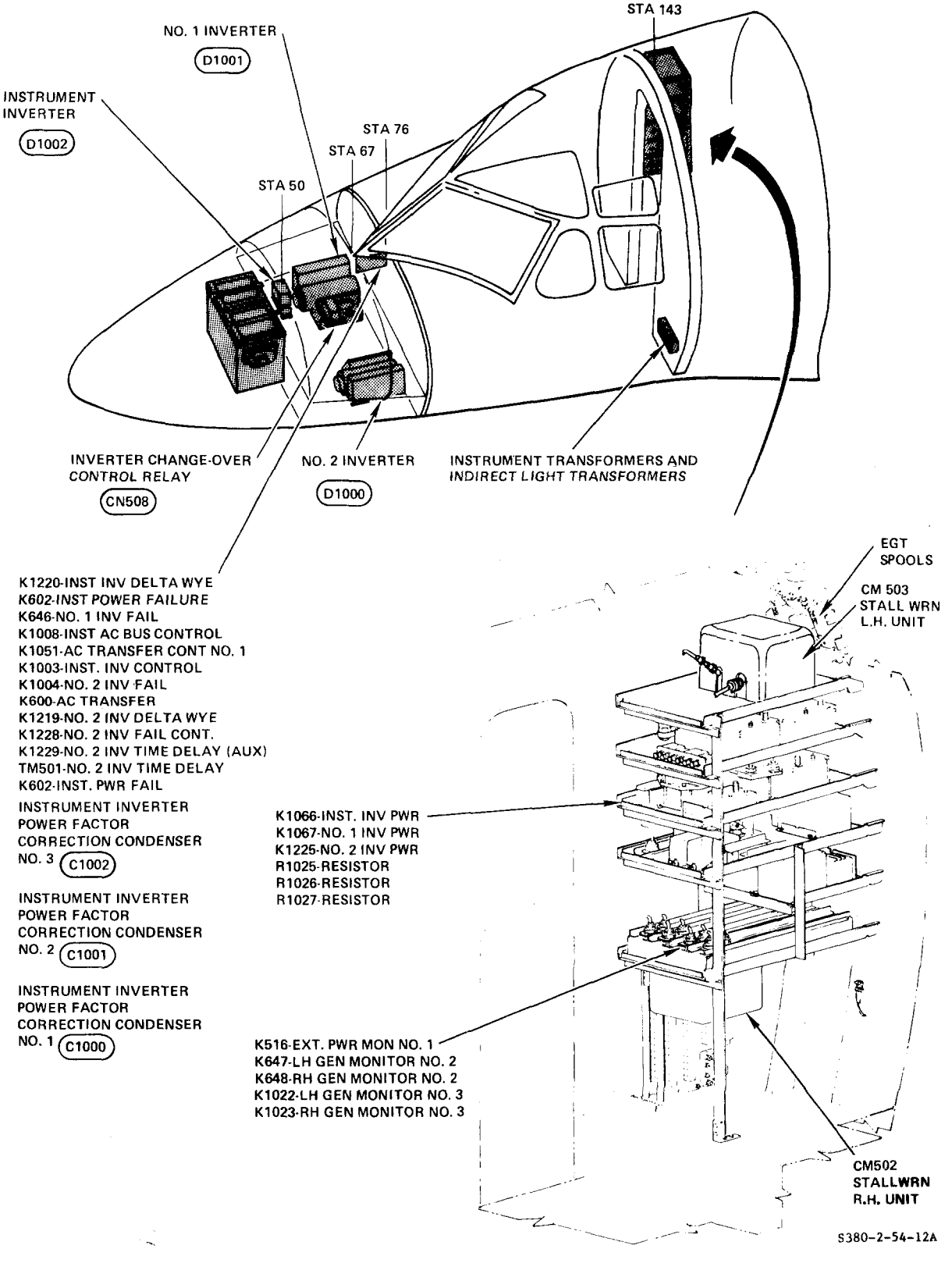

AC Generator System Equipment Location -
NA-380 Airplanes
Figure 24-20-3

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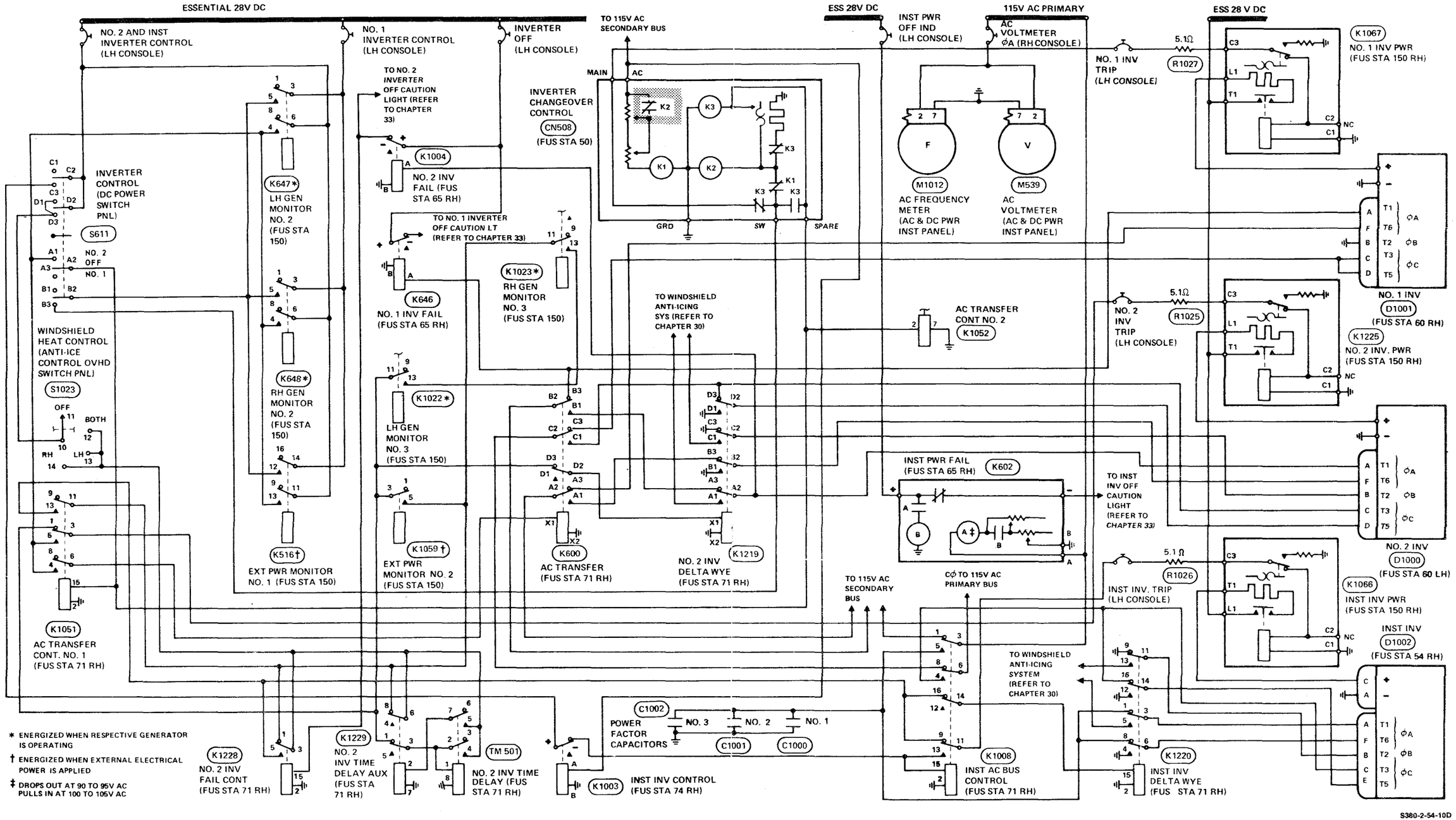
ESSENTIAL 28V DC
NO. 2 AND INST INVERTER CONTROL (LH CONSOLE)
NO. 1 INVERTER CONTROL (LH CONSOLE)
INVERTER OFF (LH CONSOLE)
TO 115V AC SECONDARY BUS
ESS 28V DC
INST PWR OFF IND (LH CONSOLE)
115V AC PRIMARY
AC VOLTMETER φA (RH CONSOLE)
ESS 28 V DC
INVERTER CONTROL (DC POWER SWITCH PNL)
S611
NO. 2 OFF NO. 1
WINDSHIELD HEAT CONTROL (ANTI-ICE CONTROL OVHD SWITCH PNL)
S1023
K1051
AC TRANSFER CONT. NO. 1 (FUS STA 71 RH)
K647*
LH GEN MONITOR NO. 2 (FUS STA 150)
K648*
RH GEN MONITOR NO. 2 (FUS STA 150)
K516†
EXT PWR MONITOR NO. 1 (FUS STA 150)
TO NO. 2 INVERTER OFF CAUTION LIGHT (REFER TO CHAPTER 33)
K1004
NO. 2 INV FAIL (FUS STA 65 RH)
TO NO. 1 INVERTER OFF CAUTION LT (REFER TO CHAPTER 33)
K646
NO. 1 INV FAIL (FUS STA 65 RH)
K1022*
LH GEN MONITOR NO. 3 (FUS STA 150)
K1059†
EXT PWR MONITOR NO. 2 (FUS STA 150)
INVERTER CHANGEOVER CONTROL
CN508
(FUS STA 50)
MAIN
AC
GRD
SW
SPARE
K1023*
RH GEN MONITOR NO. 3 (FUS STA 150)
TO WINDSHIELD ANTI-ICING SYS (REFER TO CHAPTER 30)
K600
AC TRANSFER (FUS STA 71 RH)
K1219
NO. 2 INV DELTA WYE (FUS STA 71 RH)
F
V
M1012
AC FREQUENCY METER (AC & DC PWR INST PANEL)
M539
AC VOLTMETER (AC & DC PWR INST PANEL)
AC TRANSFER CONT NO. 2
K1052
INST PWR FAIL (FUS STA 65 RH)
K602
TO INST INV OFF CAUTION LIGHT (REFER TO CHAPTER 33)
NO. 1 INV TRIP (LH CONSOLE)
5.1Ω
R1027
K1067
NO. 1 INV PWR (FUS STA 150 RH)
NO. 2 INV TRIP (LH CONSOLE)
R1025
K1225
NO. 2 INV. PWR (FUS STA 150 RH)
NO. 1 INV
D1001
(FUS STA 60 RH)
INST INV. TRIP (LH CONSOLE)
R1026
K1066
INST INV PWR (FUS STA 150 RH)
INST INV
D1002
(FUS STA 54 RH)
NO. 2 INV
D1000
(FUS STA 60 LH)
TO 115V AC SECONDARY BUS
Cↀ TO 115V AC PRIMARY BUS
TO WINDSHIELD ANTI-ICING SYSTEM (REFER TO CHAPTER 30)
C1002
POWER FACTOR CAPACITORS
NO. 3
NO. 2
NO. 1
C1001
C1000
K1008
INST AC BUS CONTROL (FUS STA 71 RH)
K1220
INST INV DELTA WYE (FUS STA 71 RH)
K1228
NO. 2 INV FAIL CONT (FUS STA 71 RH)
K1229
NO. 2 INV TIME DELAY AUX (FUS STA 71 RH)
TM 501
NO. 2 INV TIME DELAY (FUS STA 71 RH)
K1003
INST INV CONTROL (FUS STA 74 RH)
* ENERGIZED WHEN RESPECTIVE GENERATOR IS OPERATING
† ENERGIZED WHEN EXTERNAL ELECTRICAL POWER IS APPLIED
‡ DROPS OUT AT 90 TO 95V AC PULLS IN AT 100 TO 105V AC
S380-2-54-10D

AC Generation System Schematic
S/N 380 (Without Solid State Inverters)
Figure 24-20-4

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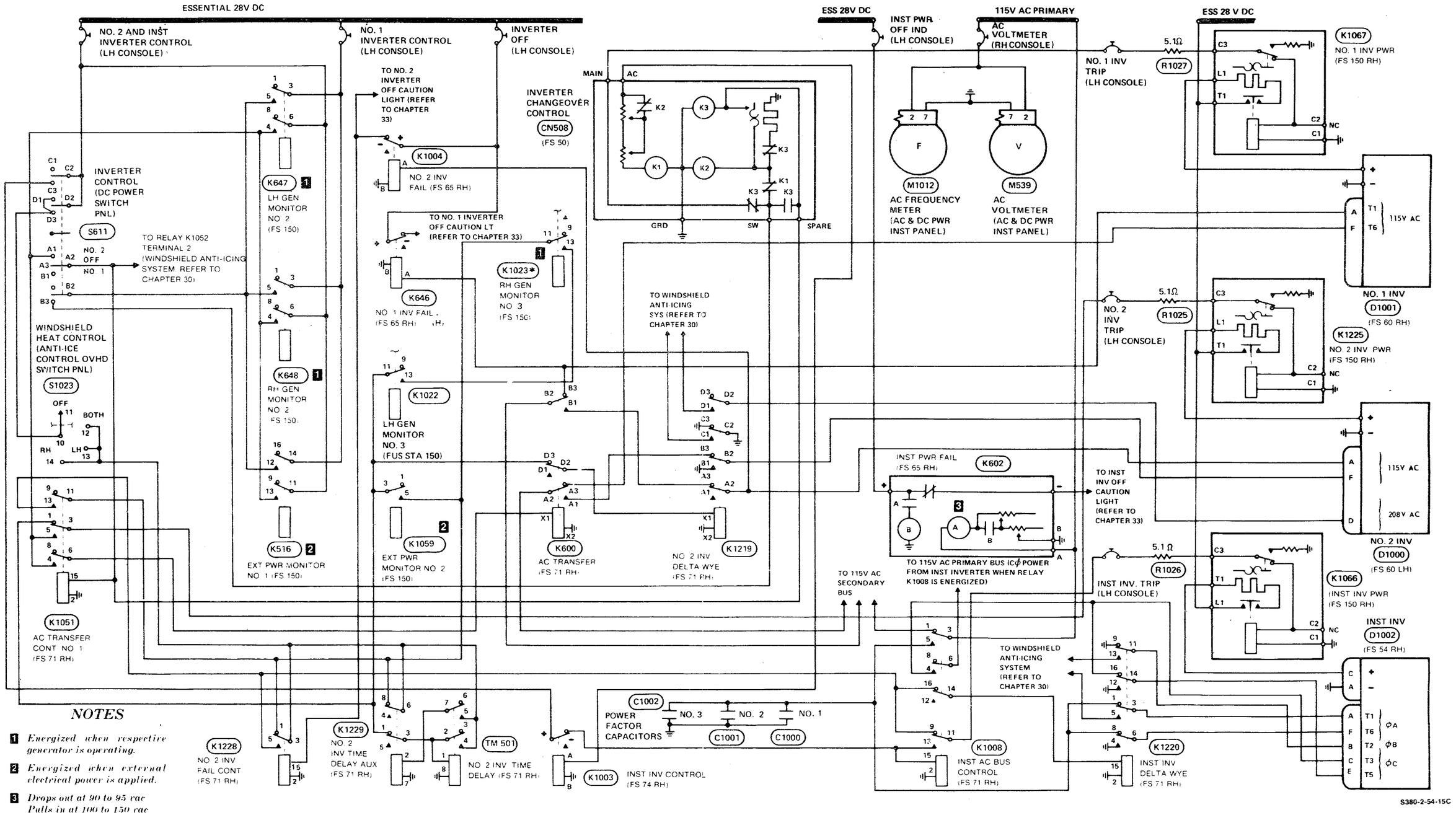

NOTES

1. *Energized when respective generator is operating.*
2. *Energized when external electrical power is applied.*
3. *Drops out at 90 to 95 vac Pulls in at 100 to 150 vac*

AC Generation System Schematic
Model NA 265-80 (With Solid State Inverters)
Figure 24-20-5

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AC GENERATION - TROUBLE SHOOTING

CAUTION: TURN OFF AIRPLANE POWER SOURCES AND DISCONNECT ALL EXTERNAL POWER FROM THE AIRPLANE BEFORE REMOVING TERMINAL BLOCK COVERS OR MAKING TEST METER CONNECTIONS. TURN OFF ALL POWER AND DISCONNECT EXTERNAL POWER FROM AIRPLANE BEFORE DISCONNECTING TEST LEADS FROM AIRPLANE TERMINAL BLOCKS OR OTHER TEST CONNECTIONS.

1. Trouble Shooting AC Power Generation System

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. "#1 INV OFF" INDICATOR LIGHT DOES NOT COME ON WHEN "INVERTER" SWITCH IS AT OFF OR NO. 2.		
(1) Burned-out indicator light.	Remove indicator light. Make continuity check of light.	Replace indicator light.
(2) Defective contacts on K646 relay.	Make power-off continuity check on relay contacts.	Replace relay.
B. "#2 INV OFF" INDICATOR LIGHT DOES NOT COME ON WHEN "INVERTER" SWITCH IS AT OFF.		
(1) Burned-out indicator light.	Remove indicator light. Make continuity check of light.	Replace indicator light
(2) Defective contacts on K1004 relay.	Make power-off continuity check on relay contacts.	Replace relay.
C. "AC INST PWR OFF" INDICATOR LIGHT DOES NOT COME ON WITH "INVERTER" SWITCH AT OFF.		
(1) Burned-out indicator light.	Remove indicator light. Make continuity check on light.	Replace light.
(2) Defective contacts on K602 relay.	Make a power-off continuity check between "plus" and "minus" terminals on relay. No continuity means bad contacts.	Replace relay.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
D. "#1 INV OFF" INDICATOR LIGHT DOES NOT GO OUT WITH "INVERTER" SWITCH AT NO. 1.		
(1) Open coil on K646 relay.	Make power-off continuity check of relay coil.	Replace relay.
(2) Open coil on K641* or K1067† relay.	Make power-off continuity check of relay coil.	Replace relay.
(3) No. 1 inverter failure.	Make power-on voltage output test of main or No. 1 inverter.	Replace inverter.
(4) Open control circuit through CN508 relay.	Make power-off continuity check between relay terminals "SW" and "NO. 1 INVERTER."	Replace relay.
(5) Open airplane wiring.	Make power-off continuity check of airplane wiring.	Repair open circuit as required.
E. "AC INST PWR OFF" INDICATOR LIGHT DOES NOT GO OUT WITH "INVERTER" SWITCH AT NO. 1.		
(1) Open wiring inside K602 relay.	Make power-off continuity check between relay terminals "A" and "B."	Replace relay.
(2) Relay K602 will not "pick up."	Make power-on ac voltage test from relay terminal "A" to ground.	Replace relay.
(3) Open airplane wiring to relay K602 terminals "A" and "B."	Make power-off continuity check of airplane wiring.	Repair open circuit as required.
(4) Defective No. 1 inverter and/or control circuitry.	Refer to probable causes (1) through (5) of trouble D.	Repair as required.
F. "AC INST PWR OFF" INDICATOR LIGHT DOES NOT GO OUT WITHIN 10 SECONDS AFTER "INVERTER" SWITCH IS PLACED AT NO. 2.)		
(1) Open wiring inside K602 relay.	Make power-off continuity check between relay terminals "A" and "B."	Replace relay.
(2) Relay K602 will not "pick up."	Make power-on ac voltage test from relay terminal "A" to ground.	Replace relay.

*NA-370 Airplanes
†NA-380 Airplanes

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
NO. 1 AND NO. 2 INVERTERS		
G. AC OUTPUT VOLTAGE DOES NOT MEASURE WITHIN STATED LIMITS.		
(1) Voltage rheostat not correctly adjusted.†	Make power-on adjustment of voltage rheostat while measuring output voltage.	Set voltage rheostat to give correct output voltage.
(2) Defective voltage rheostat.	Attempt adjustment of output voltage.	Replace inverter.
(3) Defective voltage regulator.	Attempt adjustment of output voltage.	Replace inverter.
H. NO. 1 INVERTER FREQUENCY DOES NOT MEASURE WITHIN STATED LIMITS.†		
(1) Incorrectly adjusted frequency control rheostat.	Attempt adjustment of frequency control rheostat while measuring frequency.	Set frequency control rheostat to give correct output frequency.
(2) Defective frequency control rheostat.	Attempt adjustment of frequency control rheostat.	Replace inverter.
AC TRANSFER RELAY (K600)		
I. RELAY WILL NOT ENERGIZE WHEN "INVERTER" SWITCH IS AT NO. 2.		
(1) Open coil on K647 relay.	Make power-off continuity check of relay coil.	Replace relay.
(2) Open coil on K648 relay.	Make power-off continuity check of relay coil.	Replace relay.
(3) Open coil on K600 relay.	Make power-off continuity check of relay coil.	Replace relay.
(4) Open coil on K1217* or K1051† relay.	Make power-off continuity check of relay coil.	Replace relay.
J. RELAY WILL NOT ENERGIZE AUTOMATICALLY WHEN NO. 1 INVERTER FAILS AND "INVERTER" SWITCH IS AT NO. 1.		
(1) Defective CN508 control relay.	Make power-on check for 28 volts dc between "X1" terminal and ground on K600 relay.	Replace CN508 relay.

*NA-370 Airplanes
†NA-380 Airplanes
†NA-380 Airplanes Without Solid State Inverters

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
(2) Open coil on K600 relay.	Make power-off continuity check of relay coil.	Replace K600 relay.
(3) Open airplane wiring to "X1" and "X2" terminals on K600 relay.	Make power-off continuity check of airplane wiring.	Repair open circuit as required.
(4) Open coil on K1051* relay.	Make power-off continuity check of relay coil.	Replace K1051 relay.
INVERTER CHANGE-OVER CONTROL RELAY (CN508)		
K. RELAY WILL NOT ENERGIZE NO. 1 INVERTER WHEN "INVERTER" SWITCH IS AT NO. 1.		
(1) Defective "INVERTER" switch.	Make power-off continuity check of switch while in NO. 1 position.	Replace switch.
(2) Open circuit through CN508 relay.	Make power-off continuity check between "SW" and "NO. 1 INVERTER" terminals on CN508 relay.	Replace relay.
(3) Open airplane wiring in control circuitry.	Make power-off continuity check of airplane wiring.	Repair open circuit as required.
L. RELAY WILL NOT AUTOMATICALLY SELECT NO. 2 INVERTER AFTER NO. 1 INVERTER HAS FAILED.		
(1) Defective CN508 relay.	Make power-on check for 28 volts dc from "SPARE "INVERTER" terminal to ground.	Replace relay.
(2) Open coil on K600 relay.	Make power-off continuity check on K600 relay coil.	Replace K600 relay.
(3) Open airplane wiring.	Make power-off continuity check of airplane wiring.	Repair open circuit as required.
(4) Open coil on K1217† or K1051* relay.	Make power-off continuity check of relay coil.	Replace relay.

*NA-380 Airplanes
†NA-370 Airplanes

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
(5) Open circuit in K1225 relay.	Make power-off continuity check between terminals "C3" and "C2" and between terminals "C2" and "C1." If continuity is O.K., apply 28-volt dc at terminal "C2" on relay and check for continuity between terminals "T1" and "L1."	Replace relay.

AC GENERATION - MAINTENANCE PRACTICES

1. Removing No. 1 and No. 2 Inverters (Aircraft Without Solid State Inverters)

A. The No. 1 inverter is on the right side of the airplane at fuselage station 60. The No. 2 inverter is at the same fuselage station on the left side of the airplane. The removal and installation procedures are the same for both inverters.

(1) Place all switches at OFF or in airplane-parked position.

(2) Disconnect all external power from airplane.

(3) Remove access doors as required. (See figure 6-00-3 or 6-00-4.)

(4) Remove outboard section of bulkhead at forward end of inverter.

(5) Disconnect airplane electrical wiring from inverter.

(6) Remove two mounting bolts on outboard side of inverter at forward and aft end, securing inverter mount to shock-mounted base.

(7) Slide inverter outboard to disengage pins and remove inverter from airplane.

2. Installing No. 1 and No. 2 Inverters (Aircraft Without Solid State Inverters)

A. For installation, proceed as follows:

(1) Place all switches at OFF or in airplane-parked position.

(2) Disconnect all external power from airplane.

(3) Install mount on replacement inverter with a bolt, two washers, and a nut on each of the four legs on inverter.

(4) Position inverter in airplane and slide inboard to engage mount pins.

(5) Install two bolts securing inverter mount to shock-mounted base.

(6) Connect electrical wiring and electrical connector to inverter.

(7) Install outboard section of bulkhead at forward end of inverter.

(8) Make an operational check of inverter.

(9) Install access doors.

2A. Removing No. 1 and No. 2 Inverters (Aircraft With Solid State Inverters)

A. To remove solid state inverters, proceed as follows:

(1) Remove all electrical power from airplane.

(2) Open lower access door on side of fuselage between stations 50 and 76.

(3) Disconnect electrical plug from aft side of inverter. (See figure 24-20-6.)

(4) Remove power leads from posts on aft side of inverter.

(5) Loosen screws securing phenolic wedge in place on upper outboard edge of inverter and slide wedge outboard.

(6) Remove the two 3/16-inch-diameter bolts securing lower outboard edge of inverter mount to shock-mounted inverter support structure.

(7) Slide inverter and inverter mount outboard and remove from airplane.

(8) Remove four 3/8-inch-diameter bolts securing inverter to inverter mount.

2B. Installing No. 1 and No. 2 Inverters (Aircraft With Solid State Inverters)

A. To install solid state inverters, proceed as follows:

(1) Secure inverter in inverter mount with four 3/8-inch-diameter bolts.

(2) Slide inverter and inverter mount into shock-mounted inverter support structure making certain the alignment/support pins on inboard edge of shock-mounted inverter support structure are properly engaged with inverter mount.

(3) Install two 3/16-inch-diameter bolts through outboard edge of inverter mount into shock-mounted inverter support structure.

(4) Slide phenolic wedge inboard to contact upper outboard edge of inverter and tighten screws securing in place.

(5) Connect power leads to appropriate posts on aft side of inverter.

(6) Connect electrical plug to receptacle on aft side of inverter.

(7) Turn on airplane electrical power.

(8) Using test points on aft side of inverter, measure inverter outputs.

(9) Perform operation check of inverter system.

(10) Reinstall access doors.

3. Removing Instrument Inverter

A. To remove instrument inverter, proceed as follows:

(1) Place all switches at OFF or airplane-parked position.

(2) Disconnect all external power from airplane.

(3) Remove access doors 7 and 8 (figure 6-00-3 or 6-00-4).

(4) Remove electrical plugs from inverter. Install dust caps.

(5) Remove wire bundle clamp from forward bulkhead (outboard end).

(6) Remove screws (6) from aft bulkhead stiffener (outboard section).

(7) Remove aft outboard screw and washer from inverter mount.

(8) Move inverter outboard and remove from airplane.

(9) Remove inverter from mounting plate.

4. Installing Instrument Inverter

A. To install inverter, proceed as follows:

(1) Place all switches at OFF or airplane-parked position.

(2) Disconnect all external power from airplane.

(3) Install inverter on mounting plate. Tighten screws.

(4) Position inverter in bay and slide inboard to engage both mounting plate alignment pins.

(5) Install inverter mount aft outboard screw and washer. Tighten screw.

(6) Connect electrical plugs to inverter. Lockwire.

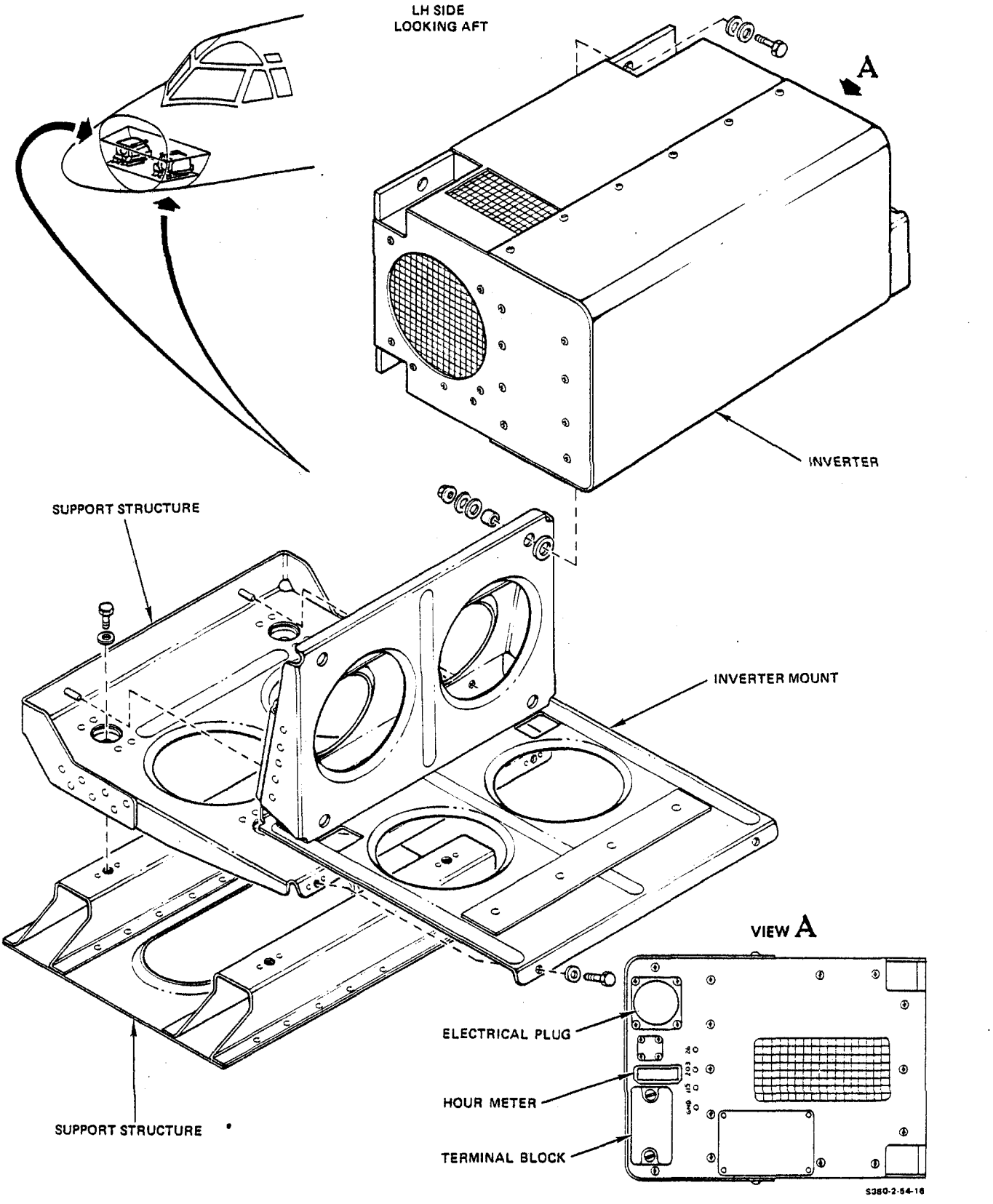
(7) Install stiffener and screws (6) on outboard end of aft bulkhead.

(8) Install wire bundle clamp on forward bulkhead.

(9) Check complete installation for security, clearance and proper lockwire installation. Remove any debris.

(10) Make an operational check on inverter system. (Refer to operational checkout of ac generation system procedure.)

(11) Install access doors 7 and 8.



No. 1 and No. 2 Solid State Inverters
Figure 24-20-6

1. Adjusting No. 1 and No. 2 Inverter Output Voltage and Frequency (Aircraft Without Solid State Inverters)

A. A dc utility external source, a 0- to 150-volt precision ac voltmeter, and a 350- to 450-cycle precision frequency meter will be required to make these adjustments. The adjustment procedure for the No. 1 and No. 2 inverters is the same, except for access panel removal.

(1) Place all switches at OFF or airplane-parked position.

(2) Disconnect all external power from airplane.

(3) Remove access doors as required. (See figure 6-00-3.)

(4) (Deleted)

(5) Connect ac voltmeter and frequency meter test leads to test jack points on inverter.

(6) Connect a dc utility external power source to airplane.

(7) Turn on external power source.

(8) Place "INVERTER" switch at NO. 1 or NO. 2 as applicable.

(9) After about 15 seconds, observe ac voltmeter reading. The meter should read 115 (±5.0) volts ac.

(10) If inverter output voltage is not correct [step (9)], adjust voltage adjustment rheostat (on receptacle end of inverter regulator box) until voltage measures within stated limits.

(11) Observe frequency meter. The meter should read 400 (±4.0) cycles.

(12) If frequency is not correct [step (11)], adjust frequency control rheostat (inside inverter regulator box) until frequency measures within stated limits.

NOTE: To gain access to the frequency control rheostat, loosen dzus fasteners that secure top to regulator box. Remove top from regulator box. Frequency control rheostat is accessible through a cutout area in interior shelf of regulator.

(13) Check ac voltmeter reading. The meter should indicate proper output voltage.

(14) If necessary, make slight adjustment of voltage adjustment rheostat to obtain proper output voltage.

(15) Check frequency meter reading. The meter should indicate proper frequency.

(16) If necessary, make slight adjustment of frequency control rheostat to obtain proper output frequency.

(17) Repeat steps (13) through (16) until both voltage and frequency are correct.

(18) Turn on airplane equipment to place an ac load on inverter.

(19) Observe ac voltmeter and frequency meter readings. Both meter readings should be within stated limits. If not, repeat steps (13) through (17).

(20) Turn equipment off and on several times while observing ac voltmeter and frequency meter. The meter readings should stay within stated limits.

NOTE: If meter readings [step (20)], do not stay within stated limits, turn off equipment, disconnect external power from airplane, and remove and replace inverter. Repeat adjusting procedure for new inverter.

(21) Turn off airplane equipment used to place ac load on inverter.

(22) Place "INVERTER" switch at OFF.

(23) Disconnect external power source from airplane.

(24) Install and secure top cover on inverter regulator box.

(25) Disconnect ac voltmeter and frequency meter test leads from inverter test jacks.

(26) Install access doors.

(27) Make an operational check of inverter system. (Refer to operational check-out of ac power generation system procedure.)

2. Adjusting Instrument Inverter Output Voltage and Frequency

A. A dc utility external power source, a 0- to 150-volt precision ac voltmeter, and a 350- to 450-cycle precision frequency meter will be required to make these adjustments.

(1) Place all switches at OFF or in airplane-parked position.

(2) Disconnect all external power from airplane.

(3) Remove access doors as required. (See figure 6-00-3 or 6-00-4.)

(4) (Deleted)

(5) Connect ac voltmeter and frequency meter test leads to test jacks on inverter.

(6) Connect a dc utility external power source to airplane.

(7) Place "INVERTER" switch at NO. 2.

(8) Disengage "NO. 2 INVERTER POWER TRIP" circuit breaker on right console.

(9) After about 10 seconds, observe ac voltmeter reading. The meter should read 115 (±3.0) volts ac.

(10) If instrument inverter output voltage is not correct [step (9)], adjust voltage adjustment rheostat (on receptacle end of inverter regulator box) until voltage measures within stated limits.

(11) Observe frequency meter. The meter should read 400 (±10.0) cycles.

(12) If frequency is not correct [step (11)], adjust frequency control rheostat (on receptacle end of inverter regulator box) until frequency measures within stated limits.

(13) Check ac voltage reading. If necessary, make a slight adjustment of voltage adjustment rheostat to obtain proper output voltage.

(14) Check frequency meter reading. If necessary, make a slight adjustment of frequency control rheostat to obtain proper output frequency.

(15) Repeat steps (13) and (14) until both voltage and frequency are within stated limits.

(16) Engage "NO. 2 INVERTER POWER TRIP" circuit breaker.

(17) Place "INVERTER" switch at OFF.

(18) Disconnect external power source from airplane.

(19) Disconnect ac voltmeter and frequency meter test leads from inverter.

(20) Install access doors.

(21) Make an operational check on inverter system. (Refer to operational check-out of ac power generation system procedure.)

1. Operational Check-out of AC Power Generation System (Without Solid State Inverter)

A. A 28 vdc utility external power source is required to make this check-out. Proceed as follows:

(1) Place all switches at OFF or in airplane-parked position.

(2) Connect 28 vdc utility external power source to airplane, and turn on external power.

(3) Close the following circuit breakers:

NO. 1 INV TRIP	LH - console
NO. 2 INV TRIP	
INST INV TRIP	
NO. 1 INVERTER CONT	
NO. 2 INVERTER CONT	
INVERTER OFF	
INST PWR OFF IND	
AC VOLTMETER ØA	RH - console

(4) Position communication and navigation equipment switches OFF.

(5) Position INVERTER CONTROL switch to OFF. The following lights on the caution-warning light panel should be illuminated:

NO. 1 INV OFF
NO. 2 INV OFF
AC INST PWR OFF

(6) Position INVERTER CONTROL switch to NO. 1. Check the following indicating lights:

NO. 1 INV OFF - out
NO. 2 INV OFF - illuminated
AC INST PWR OFF - out

NOTE: Use only a precision ac voltmeter and frequency meter for any necessary adjusting of output voltage and frequency. Test jacks are located on the inverters.

(7) Check No. 1 inverter output voltage and frequency on right instrument subpanel. The indications should be 107 to 123 vac and 400 HZ (±4).

NOTE: During inverter transfer the three inverter fail lights will illuminate for approximately 5 to 10 seconds.

(8) Open No. 1 INV TRIP circuit breaker. Check the following indicating lights:

NO. 1 INV OFF - illuminated
NO. 2 INV OFF - out
AC INST PWR OFF - out

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(9) Check No. 2 inverter output voltage and frequency. The indications should be 107 to 123 vac and 400 HZ (± 4).

(10) Open No. 2 INV TRIP circuit breaker. Check the following indicating lights:

NO. 1 INV OFF - illuminated
NO. 2 INV OFF - illuminated
AC INST PWR OFF - out

(11) Check instrument inverter output voltage and frequency. The indications should be 109 to 121 vac and 400 HZ (± 4).

(12) Position INVERTER CONTROL switch to NO. 2. Check the following indicating lights:

NO. 1 INV OFF - illuminated
NO. 2 INV OFF - illuminated
AC INST PWR OFF - illuminated

(13) Position INVERTER CONTROL switch to NO. 1. Check the following indicating lights:

NO. 1 INV OFF - illuminated
NO. 2 INV OFF - illuminated
AC INST PWR OFF - out

(14) Position INVERTER CONTROL switch to OFF.

(15) Close NO. 1 INV TRIP and NO. 2 INV TRIP circuit breakers.

(16) Disconnect dc external power source.

2. Operational Check of AC Power Generation System (With Solid State Inverters)

A. A dc utility external power source should be used for this check. Proceed as follows:

(1) Place all switches at OFF or in airplane parked position.

(2) Connect dc utility external power to airplane external power receptacle (if required), and turn on power unit.

(3) Engage the following circuit breakers:

INVERTER - NO. 1
INVERTER - NO. 2 TRIP

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"INVERTER - INST"

"INVERTER - NO. 1, NO. 2 & INST CONTROL"

"INV OFF IND"

"INST PWR IND"

(4) Place communication and navigation equipment switches at OFF.

(5) Place INVERTER switch at OFF and check that the #1 INV OFF, #2 INV OFF and AC INST PWR OFF lights on the caution-warning panel are illuminated.

(6) Place INVERTER switch at NO. 1 and check the indicating lights on the caution-warning panel. The #1 INV OFF and AC INST PWR OFF lights should be out, and the #2 INV OFF light should be illuminated.

(7) Using precision ac voltmeter and frequency meter, check the No. 1 inverter output voltages and frequencies at the test jacks on the aft end of the inverter for the following values:

110-118 VAC (RMS)
195-208 VAC (RMS)
396-404 Hz

(8) Disengage "INVERTER - NO. 1" circuit breaker. The #2 INV OFF and AC INST PWR OFF indicating lights on the caution-warning panel should be out, and the #1 INV OFF light should be illuminated.

NOTE: During inverter transfer, the three inverter fail lights may illuminate momentarily.

(9) Check the No. 2 inverter output voltages and frequency at the test jacks on the aft end of the inverter for the following values:

110-118 VAC (RMS)
195-208 VAC (RMS)
396-404 Hz

(10) Disengage the "INVERTER - NO. 2 TRIP" circuit breaker. The #1 INV OFF, and #2 INV OFF indicator lights on the caution-warning panel should be illuminated and the AC INST PWR OFF light should be out.

(11) Check instrument inverter output voltage and frequency for the following values:

109-121 VAC
400 Hz (±4)

(12) Place INVERTER switch at NO. 2. Check that the #1 INV OFF, #2 INV OFF and AC INST PWR OFF indicator lights on the caution-warning panel are illuminated.

(13) Place INVERTER switch at NO. 1. Check that the indicator lights on the caution-warning panel for the #1 INV OFF and #2 INV OFF are

illuminated, and that the indicator light for the AC INST PWR OFF is out.

(14) Place INVERTER switch at OFF.

(15) Engage "INVERTER - NO. 1" and "INVERTER - NO. 2 TRIP" circuit breakers.

(16) Remove electrical power from airplane.

DC GENERATION - MODEL NA-265-70 - DESCRIPTION AND OPERATION

1. General

During normal operation, dc electrical power is supplied by two dc generator systems. During an emergency, a limited amount of dc power is supplied by the battery system. A dc external power source can be used to energize the airplane dc busses when the airplane is on the ground. The dc external power source should be plugged into the utility external power receptacle on the left side of the airplane. (See figure 24-30-3.) Before a normal engine start is attempted, a constant-current type dc external power source must be plugged into the engine start external power receptacle. During a normal engine start, the battery system is not connected to the starter bus. During an emergency (self-sufficient) engine start, the battery system is connected to the battery bus and the starter bus. No. 1 battery remains connected to the battery bus for engine ignition. No. 2 battery connects in series with No. 1 battery, to provide 48 volts dc on the starter bus. The 1000-ampere starting load reduces this voltage almost immediately to about 24 volts dc at the beginning of the engine cranking sequence. When an engine reaches about 35% engine rpm, the starter-controller automatically disconnects the starter-generator from the starter bus and de-energizes the battery start relay, and the starter-generator begins to function as a generator. The de-energized battery start relay automatically reconnects the batteries in parallel on the battery bus. The generator reverse-current relays connect the dc generator systems to the paralleling bus.

2. Battery System

The battery system (figure 24-30-2) consists of two nickel-cadmium batteries, the battery bus, and the battery start relay. No. 1 battery connects directly to the battery bus at all times. No. 2 battery normally connects in parallel with No. 1 battery to the battery bus, through normally closed contacts on the battery start relay. During an emergency engine start from battery power, the battery start relay is energized. When energized, the relay connects No. 2 battery in series with No. 1 battery and also connects this series combination to the starter bus. Thus No. 1 battery remains connected to the battery bus and is also connected in series with No. 2 battery. This circuit arrangement provides battery bus power for engine ignition and starter bus power for an engine start from battery power. During normal operation, both batteries are connected to the battery bus. If an in-flight emergency should occur (such as failure of both dc generator systems), battery system power is automatically connected to equipment necessary for flight (on the essential bus), by means of relays in the dc generator systems and dc power distribution system. (For distribution information, refer to text covering description and operation of electrical load distribution.

NOTE: When maintenance on the airplane is anticipated which requires dc power, a dc external power source should be used to power the airplane to avoid excessive power drain from the batteries.

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3. DC Generator Systems

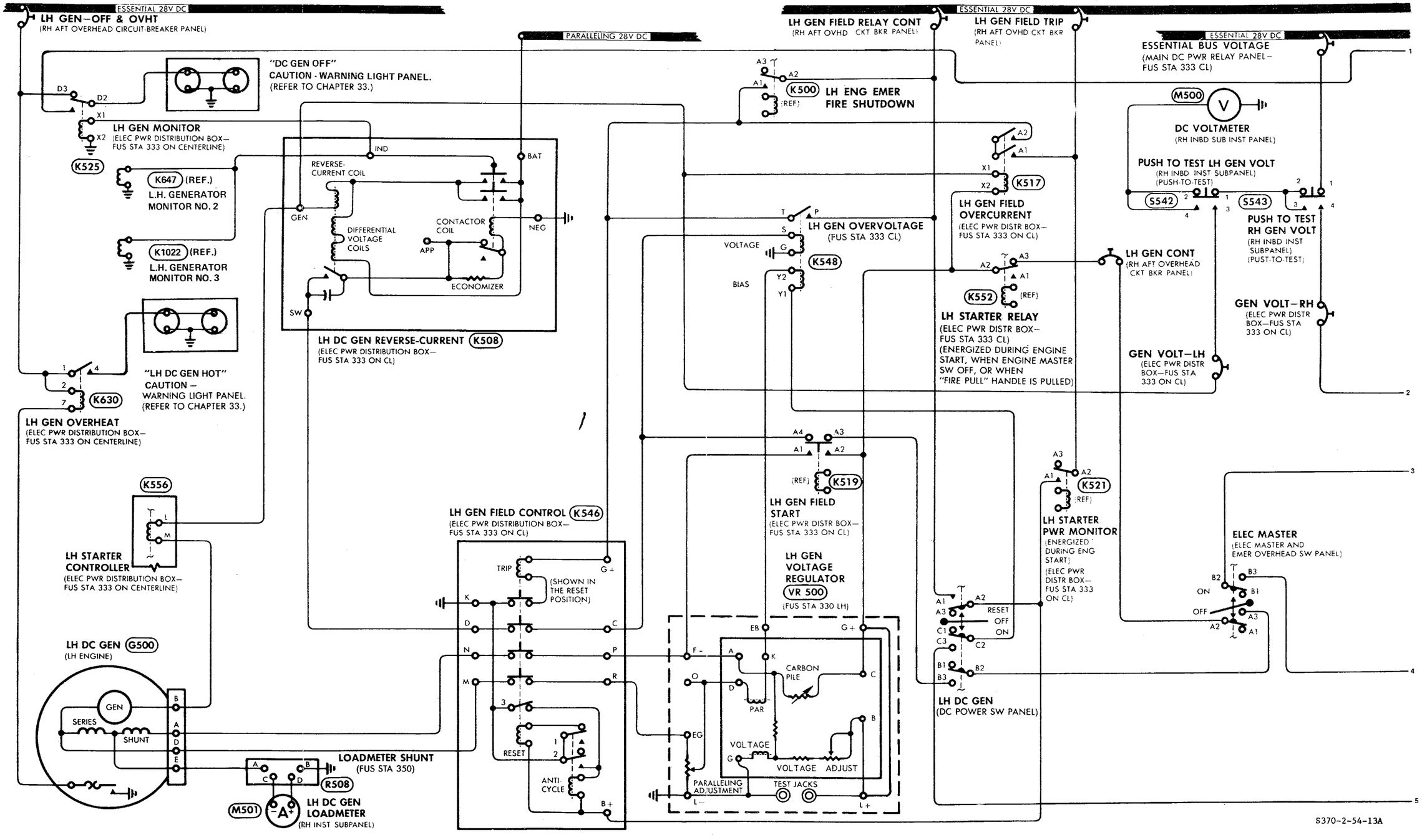
The dc generator system (figure 24-30-1) consists of the left and right generator systems. Each generator system contains a starter-generator, generator field control relay, voltage regulator, reverse-current relay, generator switch, loadmeter, overvoltage protection system, overcurrent protection system, and an overheat detection system. A generator failure indicating system is common to both generator systems. It consists of the left and right generator reverse-current relays, left and right generator monitor relays, and one indicator light. The light indicates a generator system failure, but does not show which generator system has failed. The failed generator system can be determined by observing the loadmeters for an approximate 0 percent load. The dc generator voltmeter circuit is common to both generator systems. It contains a dc voltmeter and two push-to-test switches on the right inboard instrument subpanel. The switches enable individual checking of the output voltage of each generator system. During normal operation, the dc voltmeter indicates the highest dc voltage on the paralleling bus. Each dc generator switch controls the operation of a generator system. However, the "ELEC MASTER" switch simultaneously disconnects both generator systems from the paralleling bus and disables the generators whenever the "ELEC MASTER" switch is placed at OFF.

NOTE: Because both dc generator systems are identical in operation, the operation of the left system only is given.

During normal operation ("LH DC GEN," "BATTERY," and "ELEC MASTER" switches at ON), the engine-driven dc generator develops 28 volts dc, provided the generator field control relay is in the reset position. (See figure 24-30-1.) Momentarily placing the "LH DC GEN" switch at RESET and then ON resets the generator field control relay. This completes the generator shunt field circuit (through the voltage regulator, and through the coil of the overcurrent relay) to terminal "B" on the dc generator. The generator field relay also completes circuitry to the "SW" terminal on the reverse-current relay. The carbon-pile voltage regulator maintains the generator output voltage at 28.4 (±0.7) volts dc. The reverse-current relay contacts (connected to the "SW" terminal) automatically close when the generator voltage to the differential coils exceeds 22 volts, or when the generator voltage rises above the battery voltage (on the paralleling bus) by at least 0.35 to 0.65 volt. Closing of the "SW" contacts energizes the contactor relay which connects the generator output voltage to the paralleling bus, through the reverse-current coil. DC generator system power is then available on the paralleling bus, and is connected to the other busses through the bus tie-in relays.

4. DC Generator Overvoltage Protection Systems

Each of two dc generator overvoltage protection systems is an integral part of each dc generator system. Each overvoltage protection system consists of an overvoltage relay, operating in conjunction with a generator field control relay and a voltage regulator. (See figures 24-30-1 and 24-30-3.) During normal generator system operation, the voltage coil of the overvoltage relay senses the regulated dc generator output voltage. The voltage coil becomes sufficiently energized to close the overvoltage relay contacts



DC Generation System Schematic
Figure 24-30-1 (Sheet 1 of 2)

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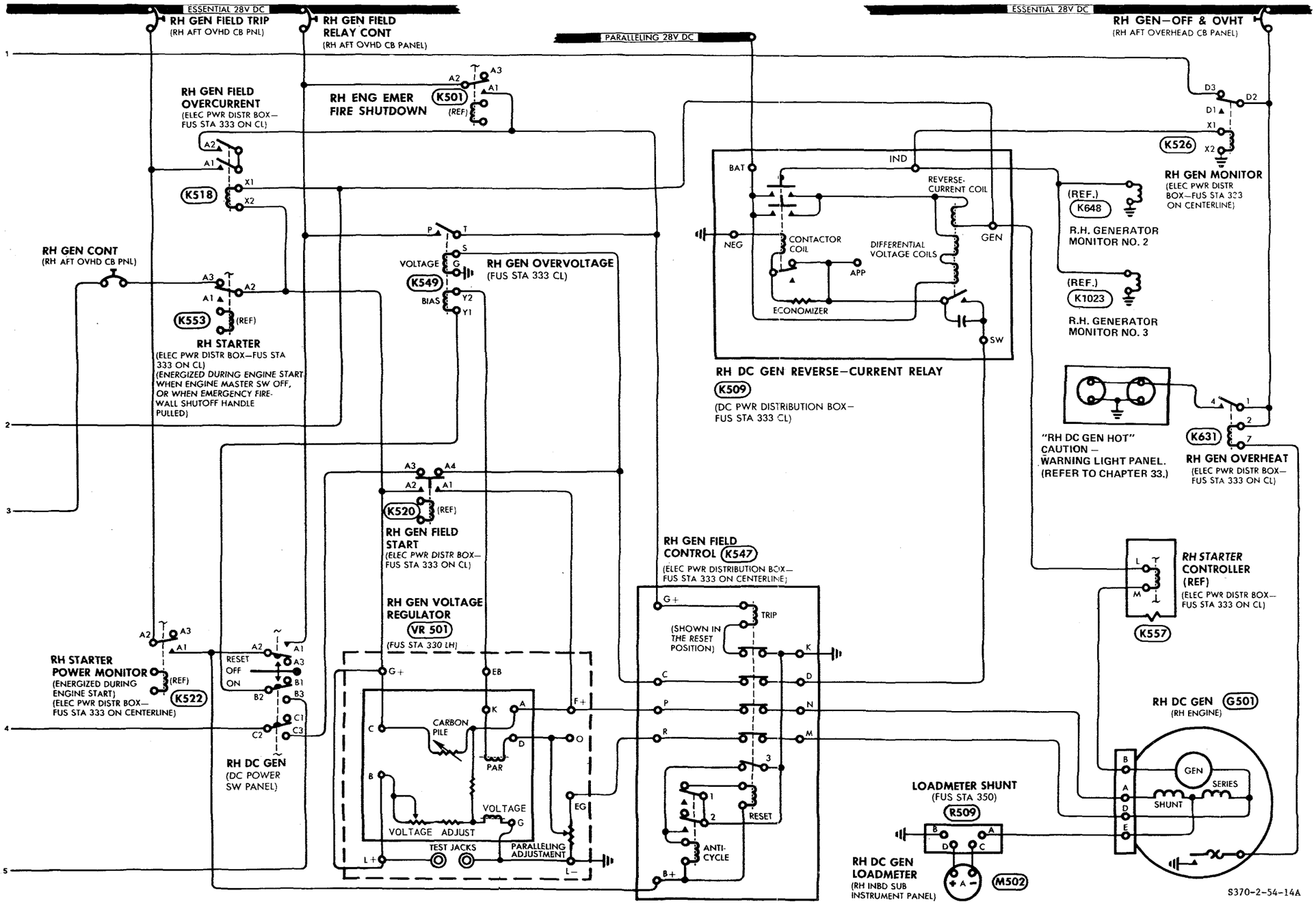

DC Generation System Schematic
Figure 24-30-1 (Sheet 2 of 2)

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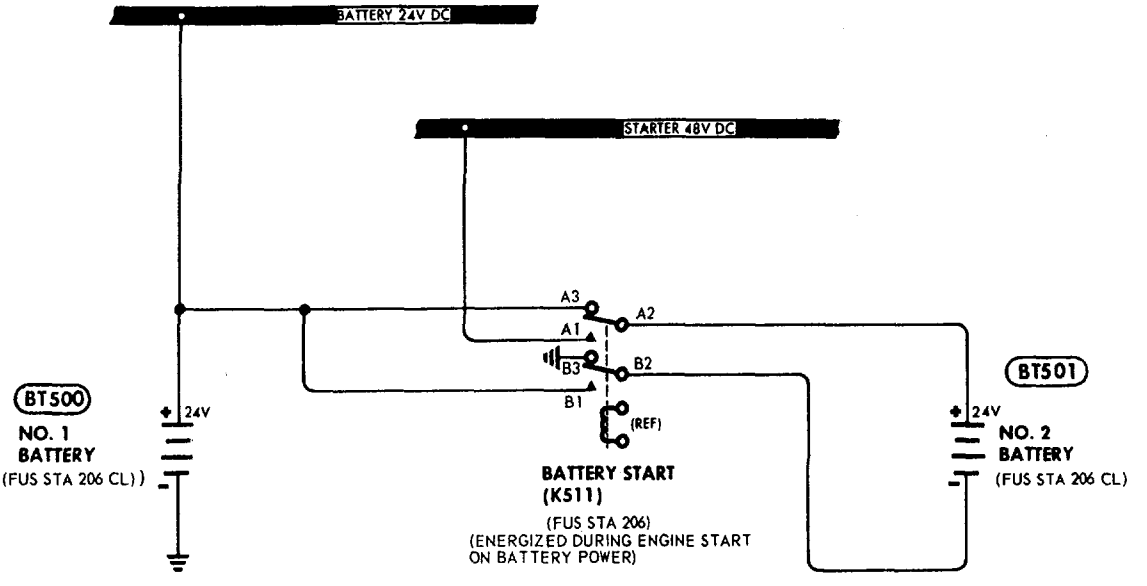

53-0-2-54-15

Battery System Schematic
Figure 24-30-2

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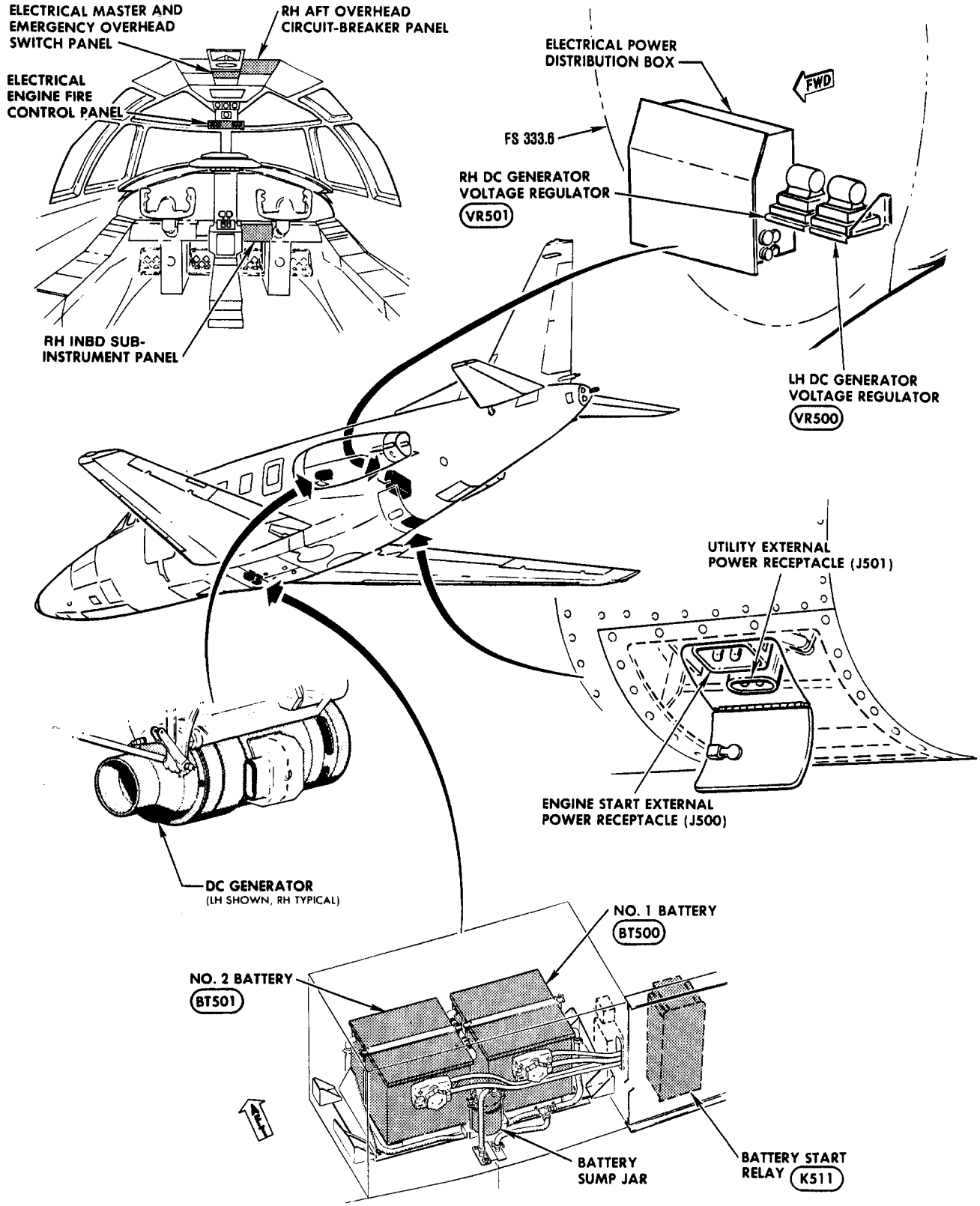

Battery Location Diagram
Figure 24-30-3

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when generator voltage rises above 33 (±1.0) volts dc. When the relay contacts close, dc power from the essential bus is connected to the trip coil of the generator field control relay. This trips the field control relay, disabling the generator (opening the shunt field circuit) and preventing high voltage from damaging the dc-operated equipment in the airplane. The bias coil on the overvoltage relay is energized by a small amount of voltage from the voltage regulator. The output voltage is controlled by adjusting the paralleling adjustment rheostat on the regulator. The voltage on the bias coil is the difference in voltage that exists between the paralleling adjustment voltages of the two generator systems. When these voltages from the generator systems are equal (no voltage difference), the overvoltage relays pick up at 33 (±1.0) volts dc. When the voltages from the generator systems are not equal, the generator system producing the higher voltage has a current flow through its bias coil from terminal "Y1" to terminal "Y2." This direction of current flow aids the action of the overvoltage relay, according to the amount of current flow. One ampere of current flow lowers the overvoltage relay pickup voltage by 5 (±0.5) volts dc. The generator system producing the lower voltage has a current flow through its bias coil from terminal "Y2" to terminal "Y1." This direction of current flow opposes the action of the overvoltage relay, according to the amount of current flow. One ampere of current flow raises the overvoltage of relay pickup voltage by 5 (±0.5) volts dc.

5. DC Generator Overcurrent Protection Systems

Each of two dc generator overcurrent protection systems (figure 24-30-1) is an integral part of each dc generator shunt field circuit. Each overcurrent protection system consists of an overcurrent relay, operating in conjunction with the voltage regulator and generator field control relay. During normal dc generator system operation, the coil of the overcurrent relay senses the generator shunt field current. The coil of the relay becomes sufficiently energized to close the relay contacts when the shunt field current rises above 30 to 36 amperes. If the shunt field current should rise above this value, the overcurrent relay closes and connects essential bus voltage to the trip coil of the generator field control relay. This trips the field control relay, opening the shunt field circuit and reducing the dc generator output to a very low value. If overcurrent protection were not provided, possible damage to the generator shunt field winding and airplane wiring could occur.

6. DC Generator Overheat Detection Systems

Each of two dc generator overheat detection systems (figure 24-30-1) is an integral part of each dc generator system. Each detection system consists of a thermoswitch-type detector, an overheat relay, and two indicator lights. During normal generator system operation, the detector (mounted on the generator housing) senses the generator housing temperature. Whenever the housing temperature rises above 155°C (311°F), the detector contacts close and complete the ground connection for the overheat relay. The relay then becomes energized and connects essential bus dc power to the overheat detector lights. The lights come on, indicating a dc generator overheat condition. The detection system indicates an overheat condition only. The overheated generator system must be manually turned off by placing the generator switch at OFF.

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7. DC Generator Failure Indicating System

The dc generator failure indicating system (figure 24-30-1) is common to both dc generator systems. It consists of the left and right generator monitor relays, the left and right generator reverse-current relays, and one indicator assembly. During normal operation, both generator system reverse-current relays and monitor relays are energized and the indicator light is extinguished. If either generator system fails, the corresponding generator reverse-current relay and monitor relay become de-energized. The de-energized monitor relay connects essential bus voltage to the indicator light. The light then comes on, indicating a generator system failure. To determine which generator system has failed, observe the dc loadmeters on the right inboard instrument subpanel. The generator system having a very low or no loadmeter reading is the failed system. The generator system that remains operational will have a high loadmeter reading.

8. Batteries

Two 24-volt, 34-ampere-hour nickel-cadmium batteries are mounted on a retractable-type shelf aft of fuselage station 206 on the centerline. They are accessible through a fuselage access door. (See figure 24-30-3.) Quick-disconnect plugs connect the battery cables to the batteries. The metal cases of the batteries are vented to atmosphere through flexible rubber tubing connected to vents on the skin of the fuselage. The inlet vent connects into the battery case. Battery fumes are expelled through an outlet connection on each battery case into a sump jar. The jar is mounted between the batteries. Battery fumes are neutralized by an impregnated felt pad inside the sump jar. The pad contains a solution of distilled water and boric acid (3 percent by weight). The neutralized fumes are expelled overboard through the outlet vent on the underside of the fuselage. The difference in air pressure between the inlet and outlet vents causes air to flow from the inlet vent to the outlet vent.

NOTE: Do not use the battery system for ground operation of airplane dc equipment. Whenever maintenance on the airplane requires dc power, connect a dc external power source to the utility external power receptacle.

- A dc external power source can be used to charge the batteries, provided the "ELEC MASTER" and "BATTERY" switches are placed at ON when the external power source is connected to the utility external power receptacle.

9. DC Generators

Each of the two engine-driven, 30-volt, 400-ampere starter-generators is installed on an engine accessory drive pad. (See figure 24-30-3.) Each starter-generator is attached to the engine accessory driven pad by a quick attach disconnect adapter (QAD). The starter-generator is secured to the

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QAD by a clamp-type band. Guide pins on the QAD determine the position of the generator on the QAD. The aft end of the starter-generator housing is encircled by a muff assembly (part of the QAD). This assembly routes ram air overboard through the outlet vent. The muff assembly is joined to the outlet vent by mechanical pressure which presses against the rubber lip on the muff outlet. The forward end of the generator housing connects to the ram-air inlet, through a section of flexible hose. The flexible hose is held in position by two hose clamps. Ram-air cooling is effective only when the airplane is in motion. A fan which is an integral part of the generator provides cooling on the ground.

10. DC Generator Reverse-current Relays

The reverse-current relays are inside the electrical power distribution box at fuselage station 333 on the bulkhead. The distribution box is accessible from the aft end of the passenger compartment. (See figure 24-30-3.) Each reverse-current relay contains a differential voltage relay, and a contactor relay. (See figure 24-30-1.) When a generator system is operating, the reverse-current relay connects the output of the generator to the paralleling bus if the generator output polarity is correct, if the generator output voltage exceeds 22 volts, or if the generator output voltage exceeds the paralleling bus (battery and other generator) voltage by at least 0.35 volt. The coil of the differential relay senses the generator output polarity and voltage. When these are correct, the differential relay contacts close and energize the contactor relay. The contactor relay connects the output of the generator to the paralleling bus through the reverse-current coil. Whenever the generator output voltage drops below the paralleling bus voltage, current flows through the reverse-current coil in a direction opposite to the normal generator current. When this reverse current exceeds 18 to 35 amperes, the differential relay contacts open and automatically de-energize the contactor relay. The de-energized contactor relay disconnects the faulty generator system from the paralleling bus.

11. DC Generator Field Control Relays

The generator field control relays are inside the electrical power distribution box at fuselage station 333 on the bulkhead. The distribution box is accessible from the aft end of the passenger compartment. (See figure 24-30-3.) Each generator field control relay has three coils. (See figure 24-30-1.) When a generator system is operating, the control relay when in the reset position connects the generator to the other components in the system. If the generator output voltage rises above 33 (±1.0) volts dc, or if dc generator field overcurrent is experienced, the trip coil of the field control relay is energized and, as such, disconnects the generator from the system. The control relay remains in the trip position until it is manually reset by placing the generator switch at RESET. This energizes the reset coil, resetting the relay for normal system operation. The anticycle relay prevents continuous cycling of the field control relay when a constant trip voltage is applied to the trip coil. The anticycle feature prevents cycling when the generator switch is held at reset with a system fault producing a continuous trip signal.

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12. DC Generator Loadmeters

The two dc generator loadmeters are on the right inboard instrument subpanel. (See figure 24-30-3.) They are accessible from the cockpit area. When the airplane is standing on the ramp with both dc generator systems operating (and no dc external power connected to the airplane), each loadmeter should read about 0.15 (15 percent). During normal taxi conditions, each loadmeter should read about 0.33 (33 percent). During normal flight conditions, each loadmeter should read about 0.45 (45 percent). If one dc generator system is turned off (or fails), its loadmeter will read 0.00 (0 percent). The loadmeter in the remaining operational generator system will then read the total load, or about twice its normal reading, depending upon the operating condition of the airplane.

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DC GENERATION - TROUBLE SHOOTING

CAUTION: WHEN TROUBLE SHOOTING AN ELECTRICAL CIRCUIT, IT MAY BE NECESSARY TO REMOVE TERMINAL BLOCK COVERS AND DISCONNECT AIRPLANE WIRING FROM TERMINAL STUDS. WHEN SUCH A PROCEDURE IS NECESSARY, DISCONNECT ALL POWER FROM THE CIRCUIT BEFORE MAKING ANY ATTEMPT TO TROUBLE SHOOT THE CIRCUIT. WHEN TROUBLE SHOOTING, DO NOT APPLY POWER TO CIRCUIT UNTIL AFTER IT HAS BEEN DETERMINED THAT NO DAMAGE WILL OCCUR TO AIRPLANE EQUIPMENT OR WIRING. ALWAYS RECONNECT ANY TEMPORARILY DISCONNECTED CIRCUIT WIRING AT COMPLETION OF TROUBLE SHOOTING PROCEDURE. FOR WIRING HOOKUP INFORMATION, REFER TO WIRING DIAGRAM MANUAL.

1. Trouble Shooting DC Power Generation System

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
	BATTERY	
A. LOSS OF CAPACITY.		
(1) Discharged battery.	Check ampere charging rate. If more than 10 amperes (5 amperes per battery), the batteries require charging until below 10 amperes.	Charge battery.
(2) Internal short circuit in one or more cells.	Check voltage output of each cell. If cell voltage is 1.05 volts or less, cell is discharged.	Replace discharged battery cell. (Cells are removable and replaceable.)
(3) External short circuit.	Check airplane wiring harness for short circuit.	Repair short circuit. Recharge battery.
(4) Extended service life.	Check voltage of each cell.	Replace battery.
	DC GENERATOR	
B. NO OUTPUT VOLTAGE FROM GENERATOR.		
(1) Internal generator short circuit.	Make voltage and continuity check of generator.	Replace generator.
(2) Generator system wiring short circuit, or open circuit.	Make continuity check of generator system wiring.	Repair short circuit, or open circuit.
C. LOW-VOLTAGE OUTPUT FROM GENERATOR.		
(1) Loose internal connection in generator.	Make voltage output check of generator.	Replace generator.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
(2) Loose wiring harness connection.	Make voltage output check of generator.	Tighten loose connection.
(3) Grease, dirt, or oil on generator commutator.	Make voltage output check of generator. Visually check condition of commutator.	Replace generator.
D. FLUCTUATING OUTPUT CURRENT OR VOLTAGE FROM GENERATOR.		
(1) Loose brush connection.	Visually inspect brush connections.	Tighten connection.
(2) Incorrectly fitted brushes.	Visually inspect brush wear and fit.	Replace generator.
(3) Internal failure of generator.	Make an operational check-out of generator system under load.	Replace generator.
REVERSE-CURRENT RELAY		
E. RELAY WILL NOT CLOSE AT REGULATED VOLTAGE.		
(1) Generator control switch at OFF.	Visually inspect switch position.	Momentarily place switch at RESET, then at ON.
(2) Generator voltage output too low.	Check generator output voltage.	Adjust voltage regulator to 28.4 (±0.7) volts dc.
(3) Improperly adjusted relay.	Check generator output voltage on paralleling bus.	Replace relay.
F. RELAY WILL NOT OPEN ON REVERSE CURRENT OF 18 TO 35 AMPERES.		
(1) Improperly adjusted relay.	Check battery voltage on "GEN" terminal of relay.	Replace relay.
GENERATOR FIELD CONTROL RELAY		
G. RELAY DOES NOT TRIP ON OVERVOLTAGE.		
(1) Dirty contacts on overvoltage relay.	Make power-off continuity check of overvoltage relay contacts.	Replace overvoltage relay.
(2) Loose connection at pin "G" or other wiring.	Check wiring harness for loose connection.	Tighten loose connection.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
(3) Open trip coil.	Make power-off continuity check of trip coil.	Replace relay.
H. FALSE TRIPPING.		
(1) Faulty over-voltage relay.	Make test of overvoltage protection system.	Replace overvoltage relay.
I. RELAY WILL NOT RESET ELECTRICALLY.		
(1) Faulty generator control switch.	Make power-off continuity check of reset circuit.	Replace control switch.
(2) Loose connection to reset coil.	Make power-off continuity check of reset circuit.	Tighten connection.
J. RELAY CYCLES CONTINUOUSLY.		
(1) Anticycle relay faulty.	Pull out on "FIRE PULL" handle. Relay cycles.	Replace field control relay.
LOADMETERS		
K. NO LOADMETER READING.		
(1) Loose connection.	Visually inspect connections.	Tighten connections.
(2) Open circuit.	Make power-off continuity check of loadmeter circuit.	Repair circuit.
(3) Faulty loadmeter.	Make continuity check of loadmeter.	Replace loadmeter.
L. GENERATORS WILL NOT STAY PARALLELED.		
(1) Misadjusted voltage regulator (or regulators).	Make an operational check-out of dc generator system.	Make voltage and paralleling adjustments to voltage regulators.
(2) Faulty voltage regulator.	Attempt voltage and paralleling adjustments to voltage regulators.	Replace voltage regulator (or regulators). Make an operational check-out of dc generator systems. Adjust as required for parallel operation.
(3) Faulty "LH DC GEN" or "RH DC GEN" switch.	Make power-off continuity test of paralleling circuit.	Replace switch.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
(4) Loadmeters not calibrated.	Temporarily interchange loadmeters; then make an operational check-out of dc generator system.	Replace loadmeters with calibrated loadmeters. Make voltage and paralleling adjustments to voltage regulators.
STARTER		
M. STARTER SLIPS DURING ENGINE START		
(1) Incorrect torque setting on starter-generator drive shaft.	Engine fails to reach rpm required for start.	Remove starter-generator and torque drive shaft. Refer to 24-30-2, paragraph 5, procedure A, step 1A.

DC GENERATION

MAINTENANCE PRACTICES

CHECKING, CYCLING, AND REPAIRING BATTERY

CAUTIONS

- *Servicing and repair procedures for nickel-cadmium batteries must be performed in an area designated for this type of battery. Under no circumstances shall lead-acid battery tools be used to service nickel-cadmium batteries.*
- *Fumes from lead-acid batteries must not be permitted in the area designated for nickel-cadmium battery servicing and repair. Failure to observe this caution will result in a contaminated nickel-cadmium battery. Contamination will render the battery useless in a very short time.*

NOTES

- *To obtain max service life for a nickel-cadmium battery, proper maintenance should be performed at specific intervals. (Refer to chapter 5.) However, each airplane is subjected to different operating and environmental conditions. In high ambient temperatures and/or flights of short duration, battery maintenance may have to be increased to prolong battery life.*

- *For checking, cycling, and repairing nickel-cadmium batteries use maintenance instructions provided by the battery manufacturer.*

Battery Removal

NOTE

Two batteries are located in the lower part of the fuselage between FS 206 and 223 (on CL). The batteries may be reached through an access door and are mounted on a movable tray which may be lowered or raised.

1. Remove all electrical power from airplane.
2. Set BATTERY switch to OFF.
3. Under fuselage at FS 206, disengage Camlock fasteners securing support for battery and sump jar vent line ports.
4. Remove battery access door.
5. Remove battery tray support locknuts.
6. Using a 7/16-in. socket wrench, turn battery hoist fitting cw to lower battery tray. (Fitting is located in bay adjacent to battery compartment and is accessible through a small spring-loaded access door.)
7. Disconnect battery cable quick-disconnect connector and temperature sensor plugs. Seal and stow quick-disconnect and cap sensor plugs.
8. Disconnect and cap or plug battery vent and drain lines to prevent any accumulated electrolyte from dripping from lines.
9. Remove battery tie-down straps and rods.

WARNINGS

- *Use care to avoid injury when lifting batteries. Each battery weighs approximately 84 lbs.*
- *Wear protective clothing to avoid caustic burns.*
- *Use care to avoid spilling electrolyte when moving batteries.*
- *Flood affected area immediately with clean water if any electrolyte is spilled on clothing or hands.*
- *Flood eyes immediately with clean water (or with a 5% solution of boric acid and water) several times if any electrolyte is splashed into eyes, then seek first aid as soon as possible.*

10. Remove batteries from tray.

Battery Maintenance

Once battery is removed from airplane, use maintenance instructions provided by the battery manufacturer.

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MAINTENANCE PRACTICES (Cont'd)

Battery Installation

WARNING

Wear protective clothing to avoid caustic burns. When moving batteries use care to avoid spilling electrolyte. Use care to avoid injury when lifting batteries. Each battery weighs 84 lbs.

1. Remove all electrical power from airplane.
2. Set BATTERY switch to OFF.
3. Inspect battery tray for cleanliness.
4. Ensure that insulator strips are in place.
5. Check that battery covers are securely fastened.
6. Place batteries onto support trays. Check that cable connectors are facing outbd on each side.
7. Secure batteries in place with tie-down straps and rods.
8. Service sump jar and install sump jar and all vent hoses.

NOTE

The min pin removal force is one pound. The desired removal force is 5 ±2 lbs.

9. Check quick disconnect plug pin sockets for looseness by inserting a 0.375 (±0.0005) in. steel pin one-half in. into each socket. Record force to remove pin.
10. Install quick disconnect cable connector and temperature sensor plugs. Safety wire.

CAUTION

Do not turn hoist fitting to force tray against supports. Such overtorquing can damage hoist.

11. Using 7/16-in. socket wrench, turn battery hoist fitting ccw to raise battery tray. Engage guide pins and raise tray until support bolts extend below tray.
12. Attach tray cradle locknuts and tighten until tray is secure against cradle.
13. Close access doors and secure support for battery and sump jar vent ports.
14. Perform operational check of battery system. (Refer to section 24-30-3.)

BATTERY VENT SYSTEM SERVICING

Battery Sump Jar Servicing Procedures

1. Disconnect camlock fasteners that secure battery and sump jar vent ports support, open access door, and lower battery tray.
2. Remove vent hoses from sump jar. Inspect tubes for visible damage and/or obstructions.
3. Remove bolts securing sump jar to tray.
4. Remove sump jar pad.
5. Saturate pad with a 3% (by weight) solution of boric acid and water. Use only enough solution to saturate pad.
6. Replace pad in sump jar and install sump jar lid.
7. Place sump jar on battery tray and secure with bolts.
8. Connect battery vent hoses.
9. Raise battery tray, close access door, and secure vent ports support with camlock fasteners.

Battery System Vent and Drain Tubes Servicing Procedures

WARNING

Do not put mouth in direct contact with an open end of a battery system vent or drain tube as a means to blow clear any obstruction in tube. Failure to observe this warning can result in severe mouth and throat burns to maintenance personnel.

NOTE

Do not permit any accumulation of electrolyte or moisture in vent or drain tube to drip onto airplane structure.

1. Observe battery servicing precautions.
2. Disconnect lower end of vent or drain tube and place it in a glass jar.

NOTE

Use a 2 to 5 psi dry-air pressure source to clear any obstructions in vent or drain tube.

3. Apply dry air pressure to upper end of vent or drain tube to remove any obstructions.

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MAINTENANCE PRACTICES (Cont'd)

Battery System Vent and Drain Tubes Servicing Procedures (Cont'd)

4. Replace vent or drain tube section if obstruction cannot be removed.
5. Connect lower end of vent or drain tube; then remove glass jar from airplane.

ELECTRICAL POWER DISTRIBUTION BOX RELAY

Removal

NOTE

The removal and installation procedures for the dc generator field control, monitor, overheat, and reverse-current relays, which are inside the electrical power distribution box, are essentially the same.

1. Set all switches at OFF or airplane-parked position.
2. Disconnect all external power from airplane.
3. Locate FS 333 bulkhead at aft end of passenger compartment.
4. Loosen the six Dzus fasteners securing lower trim close-out panel to airplane structure.
5. Lean top edge of panel fwd to clear Dzus fasteners.
6. Lift panel upward and fwd; then removal panel from airplane.
7. Loosen Dzus fasteners securing hinged lid of electrical power distribution box.
8. Open hinged lid.
9. Locate relay (by "k" number) on decal attached to cover of relay box from which relay is to be removed.
10. Disconnect electrical wiring from relay terminals.
11. Remove mounting screws that secure relay to distribution box.
12. Remove relay from distribution box.

Installation

1. Position relay on distribution box.
2. Secure relay to box with mounting screws.
3. Connect wiring to relay terminals. [For wire identification information, refer to Sabreliner Wiring Diagram Manual, Report No. NA-69-421 (70 Model) or SR-73-26 (80 Model).]
4. Close hinged lid on distribution box.
5. Secure lid to distribution box with Dzus fasteners.
6. Place lower trim close-out panel in position at FS 333 bulkhead.
7. Ensure guide pins on lower edge of trim close-out panel mate with recesses, and that Dzus fasteners line up with airplane structure.
8. Secure trim close-out panel to structure with Dzus fasteners.
9. Make an operational check of the system which contains the relay.

DC STARTER/GENERATORS

DC Starter/Generators Removal

NOTES

- ***Each engine-driven dc starter/generator is installed on an engine accessory drive pad. (See figure 24-30-3.)***
- ***Each dc starter/generator is accessible after the lower half of the engine nacelle has been released and allowed to swing downward and inbd on its hinge point.***
- ***Because removal and installation procedures are the same for each starter/generator, only the procedures for one starter/generator are given.***

1. Set all switches at OFF or airplane-parked position.
2. Disconnect all external power from airplane.
3. Remove lower half of engine nacelle.
4. Remove Marman clamp that secures flexible section of inlet duct to inlet duct on dc generator.
5. Disconnect flexible inlet duct from dc generator.
6. Work rubber hood off dc generator terminals.
7. Slide hood fwd along wiring harness until hood clears generator terminals.

NOTE

Tag leads for terminal identification during installation procedure.

8. Disconnect electrical wiring from dc generator terminals.
9. Disconnect electrical wiring from thermoswitch-type detector on dc generator housing.

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MAINTENANCE PRACTICES (Cont'd)

DC Starter/Generators Removal (Cont'd)

NOTE

The dc generator weighs about 64 lbs. This weight must be supported during the remainder of the removal procedure.

10. Loosen large bolt that secures quick attach disconnect (QAD) clamp around aft end of dc generator.
11. While maintaining dc generator in a level position, carefully slide dc generator fwd until slide end of generator shaft clears accessory drive pad.
12. Remove dc generator from engine nacelle area.

DC Starter/Generator Installation

NOTES

- ***The dc starter/generator weighs about 64 lbs. This weight must be supported until the unit is securely clamped to the quick attach disconnect (QAD) mounting on the engine accessory drive pad.***
- ***Starter/generators having a letter "A" after the serial number have deeper indexing holes in the mounting flange. Before installing a starter/generator, check length of QAD indexing pins and depth of indexing holes. Use QAD which has proper length pins to ensure proper installation of starter/generator.***
- ***Before installing a starter/generator, flash exciter shunt field (in proper direction) to make certain that shunt field retains sufficient residual magnetism. Loss of residual magnetism will result in failure of the starter/generator to build up to proper voltage when operating as a dc generator. (For flashing information, refer to flashing dc starter/generator field procedure).***

1. Disconnect all external power from airplane.

NOTE

The following step is optional and may be used to assure proper torque setting of drive shaft spring prior to starter/generator of installation. This step, however, is required if slippage during a start was suspected.

2. Check torque setting of drive shaft per steps A thru F.

NOTE

Use care to prevent damage to spline when holding drive shaft for removing or installing self-locking nut.

   A. Remove nut, washer, and damper spring from fan end of drive shaft assembly.
   B. Inspect parts for any damage. Replace nut if self-locking feature is worn.
   C. Temporarily install a 0.875 in. spacer, washer, and nut on shaft. Tighten nut until spacer bottoms against shoulder of a fan hub. (Fabricate spacer locally from 3/8-in. ID or larger tubing. Avoid any spacer that does not slip easily onto shaft.)
   D. Count and record number of shaft threads exposed above nut.
   E. Remove nut, washer, and spacer.
   F. Install damper spring, washer, and nut. Tighten nut to expose number of threads recorded (step D).

3. Lubricate starter/generator drive spline with grease.
4. Lubricate clamping bolt threads and clamping "V" groove with anti-seize compound.
5. If the starter/generator mount adapter has been removed, proceed per steps A thru E.
   A. Remove cover plate from engine (if installed).
   B. Install gasket on engine pad.
   C. Position mount adapter on engine.
   D. Install washers and nuts.
   E. Torque nuts to 160-185 in.-lbs.
6. Install generator overheat detector. (Refer to DC Generator Overheat Detector Removal).
7. Position dc generator so thermoswitch-type detector points downward from underside of generator housing.
8. Ensure generator splined drive shaft points toward QAD.
9. Support weight of dc generator and hoist generator into position immediately fwd of QAD mounting.
10. Level dc generator fwd and aft.

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MAINTENANCE PRACTICES (Continued)

DC Starter/Generator Installation (Cont'd)

11. Maintain DC generator in a level position and move it aft.
12. Engage splined drive shaft of DC generator with accessory drive pad.

NOTES

- ***The aft face of the DC generator should now fit firmly against the QAD, and the guide pins in the QAD should engage holes in the aft face of the DC generator so that the generator cannot be rotated.***
- ***Ensure thermoswitch-type detector points downward at the time guide pins engage.***

13. Engage guide pin holes with guide pins on QAD.
14. Ensure circular goove in edge of aft face of DC generator aligns with clamp ring in QAD.

NOTE

The fwd end of the starter/generator should be supported whenever the clamp bolt is being tightened or torqued, to ensure proper seating of the clamp.

15. Support fwd end of starter/generator and tighten clamp bolt. Torque clamp bolt to 120 in.-lbs, back off bolt, and repeat torquing operation several times until torque wrench stops advancing when proper torque is reached.

NOTES

- ***The clamp is properly seated when no more than approximately five threads are showing between the clamp ends.***
- ***The size of the gap between the aft edge of the clamp and the fwd edge of the exhaust air scroll (QAD) should be constant.***

16. Visually check for proper seating of clamp.
17. Safetywire QAD clamping bolt.
18. Connect electrical wiring to thermoswitch-type detector (P675).
19. Connect electrical wiring to DC generator terminals. [Refer to tags installed during removal or to Sabreliner Wiring Diagram Manual, Report No. NA-69-421 (70 Model) or SR-73-26 (80 Model).]
20. Slide rubber hood aft along wiring harness and into position over DC generator terminal block.
21. Connect flexible inlet duct to inlet housing on fwd end of DC generator.
22. Secure flexible duct to DC generator inlet with a Marman clamp.
23. Install lower half of engine nacelle.
24. Make a operational check of DC generator.

DC Generator Shunt and Ground Jumper Lead — Inspection and Cleaning

NOTE

This procedure is performed during engine change.

1. Remove bolts securing ground jumper(s) between DC shunt and ground plate.
2. Remove shunt.
3. Inspect jumpers for evidence of oxidation (green color).
4. Inspect jumpers for flexibility and any evidence of overheating. (Overheating will cause braid to be more rigid near the terminal ends.)
5. Clean shunt, jumpers, and pylon structure-to-jumper mating surface.
6. Install shunt.
7. Install jumpers, torque bolts to 175-190 in.-lbs.

DC GENERATOR BRUSH

DC Generator Brush Inspection

1. Open engine cowling and remove DC generator from engine accessory drive pad. (Refer to procedure for removing DC generator.)
2. Remove DC generator brush inspection band.
3. Pull brush spring back far enough to permit removal of brush and remove brush from brush holder.

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MAINTENANCE PRACTICES (Continued)

DC Generator Brush Inspection (Cont'd)

CAUTIONS

- ***Do not remove more than one brush at a time. To ensure proper fit against commutator, return brush to brush holder in same position as it was before removal.***
- ***Do not get oil or grease on generator commutator or brushes.***

4. Inspect brush surface that comes in contact with commutator. At least 70 percent of this surface should show contact (wear).
5. Inspect brush length. Brushes having a diagonal wear groove should be replaced when the diagonal line is worn away.
6. Return brush to brush holder (in its original position).
7. Return brush spring to its original position against brush. Ensure spring pressure holds brush firmly against commutator.
8. Repeat steps 3 thru 7 for remaining brushes.
9. Install brush inspection band.

NOTE

No crack should exceed 1/4 in. in length. No terminal block should have more than seven cracks of any length. Cracks at mounting holes should not extend more than half the distance to outer perimeter of block.

10. Inspect terminal blocks for cracks.
11. Reinstall generator on accessory drive pad. Check complete installation for security and proper safeties. Close cowling.
12. Operate engine and check output of generator.

DC Generator Brush Replacement

1. Open engine cowling and remove DC generator from engine accessory drive pad. (Refer to procedure for removing DC starter/generator.)

NOTE

Note safetywire routing before removal for reinstallation.

2. Remove safetywire, and remove brush cover (band) from generator.
3. Clean any oil or grease from generator commutator and brush holder areas.
4. Blow carbon dust from brush holder areas with 25-35 psi dry compressed air.
5. Remove safetywire, and remove screw that secures brush leads to electrical terminal.
6. Lift brush tension spring, remove old brush set, and install new brush set. Release spring and check spring is properly seated and centered on brush.
7. Check brush set is properly installed. (Brush set is held firmly against commutator by spring but can be moved up and down in holder against spring.)
8. Secure brush leads to electrical terminal with screw. Tighten screw and install safetywire.
9. Repeat steps 5 thru 8 for remaining brush sets.
10. Inspect brush installations and install brush cover (band).
11. Position cover so that screw on brush cover is on same side of generator as generator overtemperature switch. Install safetywire on brush cover screw.
12. Run-in new brushes by operating generator on bench at 3000 to 8000 rpm (generator rpm) for approximately 10 hours with a 300-amp load.
13. Check for consistent voltage level during run-in. If voltage level is not consistent after first hour, it may be neccessary to repeat brush replacement procedure.
14. Reinstall generator on engine. (Refer to procedure for installing DC starter/generator.)

DC Starter/Generator Field Flashing

NOTE

Do not make or break battery connection to starter/generator at "A" or "E-" terminal on starter/generator. To do so will result in severe damage to terminals.

1. Connect an insulated wire between negative (-) terminal of a 12-volt battery and the "E-" terminal on starter/generator.
2. Connect an insulated wire between positive (+) terminal of a 12-volt battery and one side of single-pole, single-throw knife switch.

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MAINTENANCE PRACTICES (Cont'd)

DC Starter/Generator Field Flashing (Cont'd)

3. Open knife switch.
4. Connect an insulated wire from other side of knife switch to "A" terminal on starter/generator.
5. Close knife switch for 3 to 5 seconds, then open knife switch.
6. Repeat step 5 several times to make certain that shunt field is properly flashed.
7. Disconnect knife switch and wiring from starter/generator.
8. Install starter/generator.

DC Generator Overheat Detector Removal

NOTE

Each of the two overheat detectors is mounted on a dc generator housing. (See figure 24-30-3.) The detector is accessible after the lower half of the engine nacelle is removed. Because removal and installation procedures are the same for either overheat detector, only the procedures for one overheat detector are given.

1. Set all switches at OFF or airplane-parked position.
2. Disconnect all external power from airplane.
3. Remove lower half of engine nacelle.
4. Locate overheat detector on underside of dc generator housing.
5. Disconnect electrical wiring from overheat detector.
6. Remove air outlet adapter.
7. Unscrew overheat detector from dc generator housing.

DC Generator Overheat Detector Installation

1. Set all switches at OFF or airplane-parked position.
2. Disconnect all external power from airplane.
3. Inspect tapped mounting hole for overheat detector to ensure no dirt or foreign material is present. (The tapped mounting hole is on the underside of the installed dc generator.)
4. Place adapter (Bendix Part No. 1530507), figure 24-30-6, on end of overheat detector which is on top when detector is held in installed position. Make sure rounded side of adapter is on top (points toward CL of generator) and is centered on end of detector.
5. Screw overheat detector and adapter into tapped hole in generator housing until detector "bottoms out" and compresses tip of adapter. Do **not** use a wrench. Tighten detector finger-tight only. Do not upset position of adapter on end of detector.

NOTE

Do not back off detector more than one complete turn.

6. Unscrew or back off detector just enough to permit connector portion of detector to face 45 degrees to left of CL of dc generator housing (points toward generator cable terminals).
7. Hold detector in position with wrench, and tighten detector locknut against generator housing to 70 (+0/-5) in.-lbs with torque wrench.
8. Connect electrical wiring to overheat detector.
9. Install air outlet adapter. Position flange offset under overheat detector.
10. Install lower half of engine nacelle.

DC Generator System Output Voltage Adjustment

Equipment/Tools

- 2 Precision 0-to-50 Vdc Voltmeters

Procedure

1. Set all switches at OFF or airplane-parked position.
2. Disconnect all external power.
3. Gain access to voltage regulators, on lower left portion of cabin aft bulkhead.
4. Connect dc voltmeters to right and left voltage regulator test jacks. (The jacks are on the bases of the regulators.)
5. Check proper polarity by connecting positive (+) terminal of voltmeter to red-colored jack and negative (-) terminal of voltmeter to black-colored test jack.

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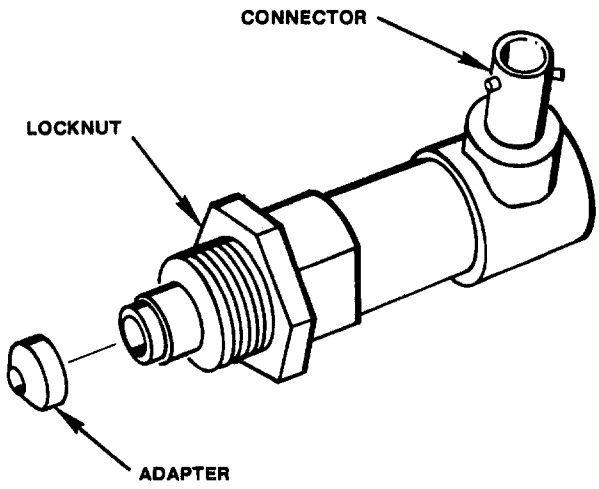

Thermoswitch — Type Overheat Detector
Figure 24-30-6

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MAINTENANCE PRACTICES (Cont'd)

DC Generator System Output Voltage Adjustment (Cont'd)

Procedures (Cont'd)

6. Connect external electrical power to airplane.
7. Start engines. (Refer to chapter 71.)

NOTE

The BATTERY and DC GEN switches are to remain at OFF during these procedures.

8. Set engine rpm at IDLE.

NOTES

- ***The voltage regulators must be operated for a min of 30 minutes to attain operating temperature, or until voltage stabilizes, before any attempt is made to adjust the dc generator system output voltage. Both voltage regulators must have the same part numbers.***
- ***The RH DC GEN switch is cycled through RESET, ON and OFF to make certain that the generator field control relay is reset. The switch should remain in the OFF position until after the output voltage of the generator system is properly adjusted.***

9. Set RH DC GEN switch momentarily to RESET, at ON, and then back to OFF.
10. Check RH dc voltmeter reading of 28.4 (±0.7) vdc.
11. Adjust voltage adjustment rheostat (screwdriver slot) on RH regulator until approximately 28.4 volts is indicated on voltmeter, if the voltmeter does not read approximately 28.4 volts. Do not adjust rheostat (knurled knob) on regulator mounting base in this step.
12. Press down and hold PUSH TO TEST RH GEN VOLT switch.
13. Check voltage. The airplane dc voltmeter should read approximately 28.4 volts.
14. Rotate airplane dc voltmeter instrument adjustment screw on face of instrument until indicator reads ±1% (0.28 to 0.29 volts) of voltmeter if the airplane dc voltmeter does not read within 1% of voltmeter.
15. Release PUSH TO TEST RH GEN VOLT switch.
16. Repeat steps 9 thru 11 for left dc generator.

NOTE

Steps 17 thru 22 verify proper operation of left and right dc generator reverse current relays.

17. Set RH DC GEN switch momentarily at RESET and then ON. Leave LH DC GEN switch at OFF.
18. Disconnect external electrical power.
19. Check voltage. The airplane dc voltmeter should read approximately 28.4 volts.
20. Set RH DC GEN switch at OFF.
21. Set LH DC GEN switch momentarily at RESET, then ON.
22. Check voltage. The airplane dc voltmeter should read approximately 28.4 volts.

DC Generator System Maximum Output Adjustment

NOTE

Perform procedures in DC Generator System Output Voltage Adjustment before performing max output voltage procedures.

1. Position LH and RH engine throttles at IDLE.
2. Set RH DC GEN switch momentarily at RESET and then ON. Set LH DC GEN switch at OFF.
3. Monitor RH dc generator loadmeter. Increase dc bus load, if required, until meter reads approximately 0.3 (30%) load.
4. Check voltage. The airplane dc voltmeter and voltmeter should read approximately 28.4 volts.
5. Set RH DC GEN switch at OFF.
6. Repeat steps 1 thru 4 for LH dc generator.
7. Set LH DC GEN switch at ON.
8. Increase LH engine speed to 65% (±5%) rpm.
9. Check voltage. The voltmeter should read 28.6 (±0.1) volts.
10. Reduce engine rpm to IDLE if approximately 28.6 volts is not obtained. Increase voltage adjustment rheostat in increments of 0.1 volt.
11. Repeat steps 8 thru 10 until approximately 28.6 volts is obtained at 65% rpm.

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MAINTENANCE PRACTICES (Cont'd)

DC Generator System Maximum Output Adjustment (Cont'd)

12. Adjust dc load on bus system to highest value possible near full load of 1.0 (100%).
13. Increase LH engine speed momentarily to 90% (±5%) rpm.
14. Check voltage. The voltmeter should read 29.0 (+0.0/-0.2) volts.
15. Reduce engine rpm to IDLE, and place LH DC GEN switch at OFF if approximately 29 volts is not obtained. Increase voltage adjustment rheostat in increments of 0.1 volt.
16. Set LH DC GEN switch at ON.
17. Repeat steps 12 thru 15 until approximately 29 volts is obtained at 90% rpm.
18. Set LH DC GEN switch at OFF.
19. Repeat steps 7 thru 17 for RH dc generator.
20. Perform procedures in Parallel Operation Adjustment.

DC Generator System Parallel Operation Adjustment

NOTE

Ensure that max voltage output has been obtained for individual generators in accordance with DC Generator System Maximum Output Adjustment before performing paralleling procedures.

1. Set RH and LH DC GEN switches momentarily at RESET, then ON if dc generators are not on the line.
2. Increase speed of both engines to 65% (±5%) rpm.
3. Adjust dc load on bus system until generator loadmeter indicates at least 0.25 (25%).

NOTE

A difference in loadmeter readings of more than one-half scale division indicates that the generators are not sharing the load equally. To balance load, proceed with the following steps.

4. Turn knurled knob on voltage regulator base, for generator indicating lowest loadmeter reading, a slight amount cw. (Increase voltage.) Turn knurled knob on voltage regulator base, for generator indicating highest loadmeter reading, a slight amount ccw. (Decrease voltage.)

NOTE

Do not adjust rheostat (screwdriver slot) on regualtor unit in this step.

5. Repeat step 4 until loadmeter readings are within one-half scale division.
6. Increase dc load to highest value possible near full load.
7. Repeat step 4 until loadmeter readings are within one-half scale division.
8. Reduce engine rpm to IDLE.
9. Recheck voltage output. (Refer to procedures in DC Generator System Maximum Output Adjustment.)
10. Repeat paralleling and voltage adjustment until voltage is 29.0 (+0.0/-0.2) vdc at 90% rpm and load is equally divided between generators to within one-half scale division.
11. Shut down engines. (Refer to chapter 71.)
12. Set all switches at OFF or airplane-parked position.
13. Disconnect voltmeters.
14. Install access and trim panels.

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WARNING: WHEN GROUND MAINTENANCE PROCEDURES REQUIRE THE REMOVAL OF PROTECTIVE COVERINGS (SUCH AS TAPE, RUBBER COVERS, ETC), THESE COVERINGS MUST BE REINSTALLED BEFORE ANY ELECTRICAL POWER IS CONNECTED TO THE AIRPLANE. FAILURE TO OBSERVE THIS WARNING COULD RESULT IN A FIRE AND POSSIBLE LOSS OF THE AIRPLANE, CAUSED BY AN ACCIDENTAL SHORT CIRCUIT OF THE UNPROTECTED ELECTRICAL CONDUCTOR.

CAUTION: BEFORE REMOVING TERMINAL BLOCK COVERS TO MAKE IT EASIER TO TEST OR ADJUST THE ELECTRICAL SYSTEM, DISCONNECT ALL EXTERNAL POWER FROM THE AIRPLANE AND PLACE ALL AIRPLANE SWITCHES AT OFF OR AIRPLANE-PARKED POSITION.

NOTE: A precision 0- to 50-volt dc voltmeter, a constant-voltage type dc external power source of 28 volts dc, and a constant-current type dc external power source of 1000 amperes capacity are required to check and test the dc power electrical system.

- The dc generators and voltage regulators should be at operating temperature before the generator systems are adjusted.

1. Testing Battery System

NOTE: The specific gravity readings of the individual battery cells do not indicate the state-of-charge of a nickel-cadmium battery. The specific gravity remains the same, whether the battery is charged or discharged. The state of charge is determined by the amount of charging current. The batteries are considered charged when the charging current is less than 10 amperes for both batteries.

A. To test battery system, proceed as follows:

(1) Place all airplane switches at OFF or airplane-parked position.

(2) Connect a 28.5-volt, constant-voltage type dc external power source to airplane utility external power receptacle. Turn on external power source.

(3) Place "ELEC MASTER" switch at ON.

(4) Make certain "BATTERY" switch is at OFF.

(5) Note dc external power source loadmeter reading.

(6) Place "BATTERY" switch at ON.

(7) Note dc external power source loadmeter reading.

(8) Compute the difference between loadmeter readings [step (7) minus step (5)].

NOTE: If the difference [step (8)] exceeds 10 amperes, the batteries are not operating at full capacity. They should remain on charge until the difference in loadmeter readings is less than 10 amperes.

(9) Repeat steps (5) through (8) until difference in loadmeter readings is less than 10 amperes.

(10) Place "BATTERY" switch at OFF.

(11) Place "ELEC MASTER" switch at OFF.

(12) Disconnect dc external power source from airplane.

2. Testing DC Generator System

A. To test dc generator system, proceed as follows:

(1) Make certain "ELEC MASTER" switch is at ON.

(2) Start right engine. (Refer to Chapter 71.)

(3) Start left engine.

(4) Place "LH DC GEN" switch at OFF.

(5) Note dc voltmeter. The meter should read 28.4 (±0.7) volts dc, the output voltage of the right dc generator system.

(6) Note left dc generator loadmeter. The meter should read 0.00 (0.0 percent).

(7) Note right dc generator loadmeter. The meter should read 0.30 (30 percent).

(8) Place "LH DC GEN" switch at RESET, then at ON.

(9) Place "RH DC GEN" switch at OFF.

(10) Note left dc generator loadmeter. The meter should read 0.30 (30 percent).

(11) Note right dc generator loadmeter. The meter should read 0.00 (0.0 percent).

(12) Note dc voltmeter. The meter should read 28.4 (±0.7) volts dc, the output voltage of the left dc generator system.

(13) Place "RH DC GEN" switch momentarily at RESET, then at ON.

(14) Note both dc generator loadmeters. Each meter should read about 0.15 (15 percent).

NOTE: If loadmeters do not read the same (within one-half scale division), the electrical load is not equally shared between the dc generator systems. To equalize load between the dc generator systems, adjust voltage adjustment rheostats and paralleling rheostats on voltage regulators.

(15) While observing dc loadmeters, place "INVERTER" switch at ON, at OFF, and again at ON. Both loadmeter readings should rise, fall, then rise together, indicating that both dc generators remain operational and share the dc load.

NOTE: If the sudden increase in dc load causes one of the dc generators to trip off, a faulty or misadjusted generator system is indicated. Investigate and correct as required.

(16) Place "RH DC GEN" switch at OFF.

(17) Shut down left engine. (Refer to Chapter 71.)

(18) While left engine is windmilling downward in speed, observe left dc generator loadmeter. The meter reading should indicate a rapid decrease in load (from 0.30) toward zero. Upon reaching zero, the meter should continue reading downward to a minus 0.1 reading, then rapidly return to zero.

NOTE: The minus 0.1 reading indicates proper operation of the reverse-current relay. The loadmeter reading must not exceed 0.1 in the reverse direction. If it does, or if the pointer fails to return to zero immediately after reading in the reverse direction, the reverse-current relay is faulty and should be replaced.

(19) Place "RH DC GEN" switch at ON.

(20) Note right dc generator loadmeter. The meter should read about 0.30 (30 percent), or full load.

(21) Shut down right engine.

(22) Repeat step (18) as applied to right dc generator loadmeter. Observe same requirements of operation.

(23) Place "BATTERY" switch at OFF.

(24) Place "ELEC MASTER" switch at OFF.

INSPECTION/CHECK

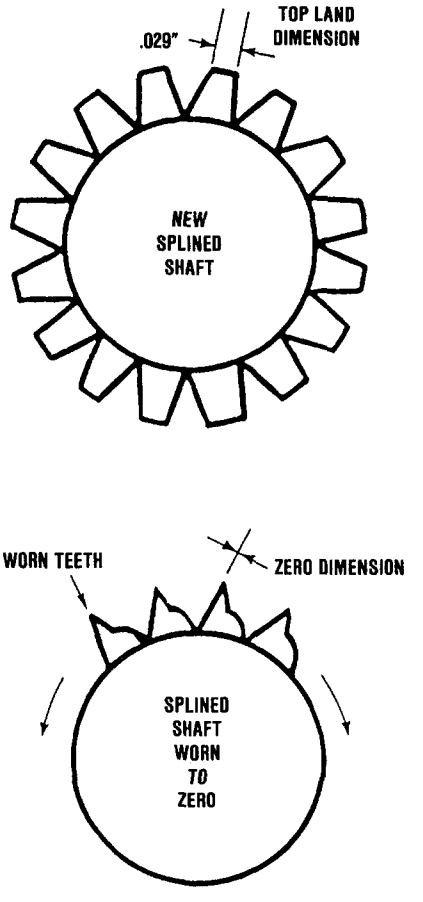
INSPECT — STARTER/GENERATOR SHAFT SPLINES — S/N 370-2 thru -6, -8, and -9

1. Remove starter/generator per instructions in this chapter.
2. Inspect shaft splines for wear as shown in Detail A. The "top land" of new spline teeth is .039 for 12-tooth splines and .029 for 16-tooth splines.

NOTE

The splines are safe down to a "top land" dimension of zero, but if service records indicate the "top land" dimension may reach zero during next service period, the drive shaft should be replaced.

3. Lubricate starter/generator shaft splines with MIL-G-81322A-GPW high temperature grease.
4. Install starter/generator per instructions in this chapter.
5. Secure airplane as desired.



Detail A

NEW

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DC GENERATION - MODEL NA-265-80 - DESCRIPTION AND OPERATION

1. General

During normal operation, dc electrical power is supplied by two dc generator systems. During an emergency or during ground operation (with no external ground power available) when engines are not operating, dc power is supplied by the battery system. Provisions are included for connecting an external dc power source for extended ground operation and engine starting, if desired. External power is applied through a power receptacle on the right side of the airplane. (See figure 24-30A-1.) For an engine start using external power, a constant-current type dc electrical power unit is plugged into the external power receptacle to energize the essential and start busses. For a normal battery (self-sufficient) start the battery system is connected to the start and essential busses in such a way that batteries No. 1 and No. 2 are connected in series providing 48 volts on the start bus for the starter-generator and battery No. 1 alone provides 24 volts to the essential bus for engine ignition. (See figure 24-30A-2.) After the engine rpm reaches about 42 percent RPM the starting sequence ends and the starter-generator begins to function as a generator. The generator reverse-current relays connect the generators to the essential bus.

2. Battery System - Airplanes Without S.O.K. 17

The battery system (figure 24-30A-2) consists of two nickel-cadmium batteries, two battery control switches, two battery isolation relays, a battery relay, and control circuitry. The battery system equipment locations are shown in figure 24-30A-1.

With the airplane on the ground and all switches off, each battery is connected to the coil and to normally open contacts in the respective isolation relay. When the "ELEC MASTER" and "BATTERY" switches are positioned at ON, the coil of the isolation relay is grounded through the battery disconnect relay K1092 (or through the generator monitor and external power monitor relays provided that no external power is connected to the airplane and the engines are not operating). The isolation relays can also be energized by placing the "BATTERY" switches to EMERG, which furnishes a direct path to ground for the isolation relay coils, provides that the "ELEC MASTER" switch is at ON position. When either isolation relay is energized the battery relay (K510) is energized and the affected battery is connected to the essential bus. In the event that all generator power is lost during flight, battery power will be available to the essential bus. During an engine start on battery power the battery start relay (K511) energizes and connects the batteries in series to the start bus. Battery No. 1, however, still supplies power to the essential bus through the isolation and battery relays. During a battery start the "BATTERY" switches may be placed in the ON (providing a battery-hot condition does not exist) or EMERG position to provide a ground path for the isolation relays. With external power connected or the APU operating and no battery overheat condition, the "BATTERY" switches may be placed in the ON or EMERG position to provide a ground path for the isolation relays. With no external power applied and with the APU operating, the "BATTERY" switches may be placed in the ON or EMERG position to provide a ground path for the isolation relays.

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On airplanes changed by Sabreliner Optional Kit No. 26 the batteries are connected in parallel during engine starts. The parallel connection may not be as effective as the series connection during cold weather engine starts. When the ambient temperature is below freezing (0 degrees C) two or more engine start attempts may be required to raise battery temperatures enough to get minimum engine RPM. The operator may elect at his option to change the airplane wiring back to the series connection at low O.A.T.s and eliminate the possibility of repeated starting attempts being necessary.

2A. Battery System - Airplanes With S.O.K. 17

The battery system (figure 24-30A-2) consists of two nickel-cadmium batteries, two battery control switches, two battery isolation relays, a battery relay, APU start disable, and control circuitry. The battery system equipment locations are shown in figure 24-30A-1.

With the airplane on the ground and all switches off, each battery is connected to the coil and to normally open contacts in the respective isolation relay. When the "ELECT MASTER" and "BATTERY" switches are positioned at ON, the coil of the isolation relay is grounded through the battery disconnect relay K1092 (or through the generator monitors and external power monitor relays, provided that no external power is connected to airplane and the engines are not operating. The isolation relays can also be energized by placing the "BATTERY" switches to EMERG, which furnishes a direct path to ground for the isolation relay coils, provided that the ELECT MASTER switch is at the ON position. When the APU is not operating and when either isolation relay is energized the battery relay (K510) is energized and the affected battery is connected to the essential bus. In the event that all generator power is lost during flight, battery power will be available to the essential bus.

With the engine start switch (S1230) in the COLD position, during an engine start on battery power, the battery start relay K511 energizes and connects the batteries in series to the start bus. With the APU operating, the APU will furnish the required electrical power to the essential bus. When the APU is not operating, the NO. 1 battery supplies the required electrical power to the essential bus during the starting cycle. When the engine start switch (S1230) is at the NORMAL position, relay K511 remains de-energized which allows the batteries to remain connected in parallel to the starter bus. Refer to Chapter 80 for a complete description and operation of the starting system.

During a battery start the "BATTERY" switches may be placed in the ON (providing a battery-hot condition does not exist) or EMERG position to provide a ground path for the isolation relays. With external power connected or the APU operating and no battery overheat condition, the "BATTERY" switches may be placed in the ON or EMERG position to provide a ground path for the isolation relays. With no external power applied and with the APU operating, and a battery hot condition, the "BATTERY" switches must be placed in the EMERG position to provide a ground path for the isolation relays.

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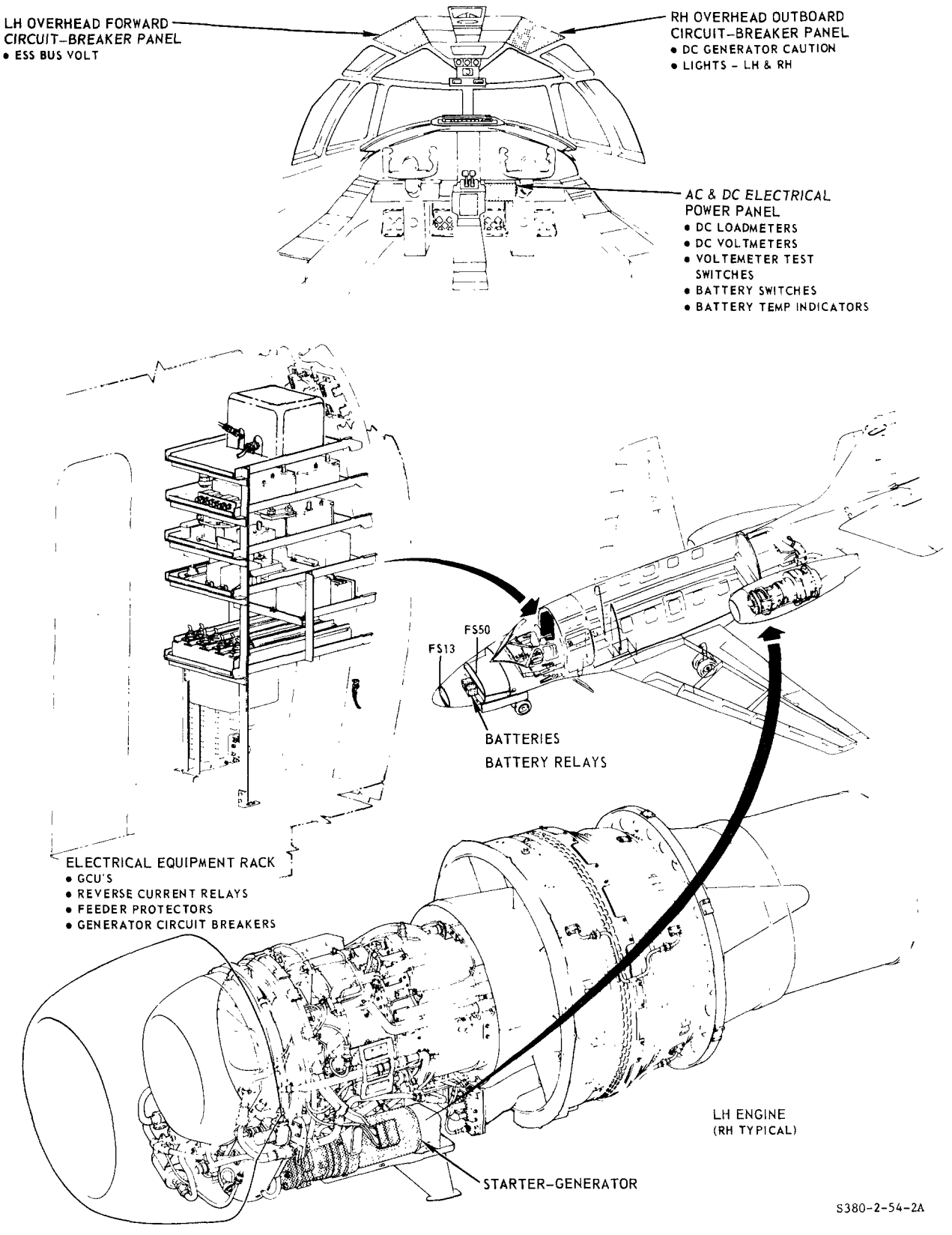

DC Generation System Component Locations
Figure 24-30A-1

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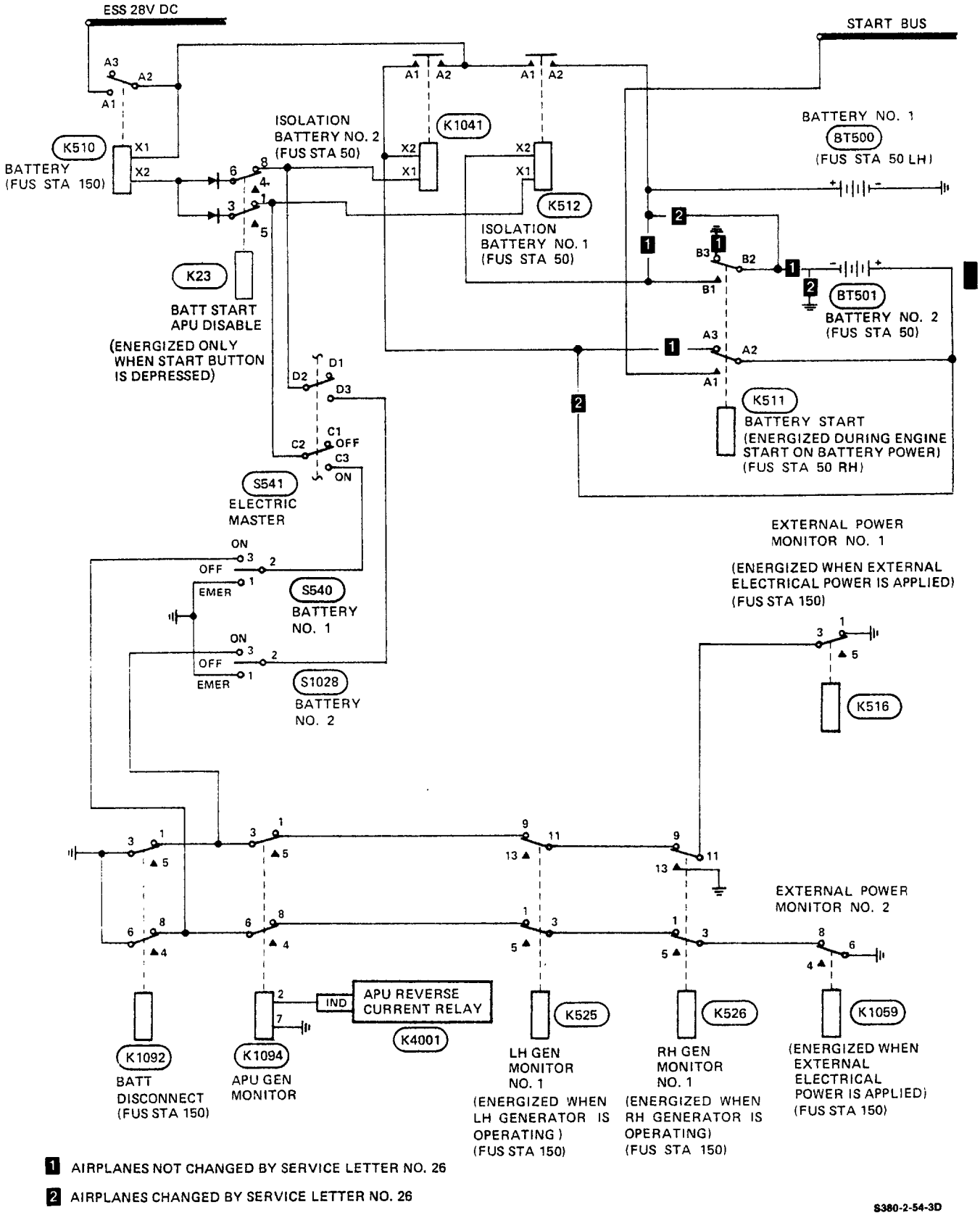

Battery System Electrical Schematic
Figure 24-30A-2

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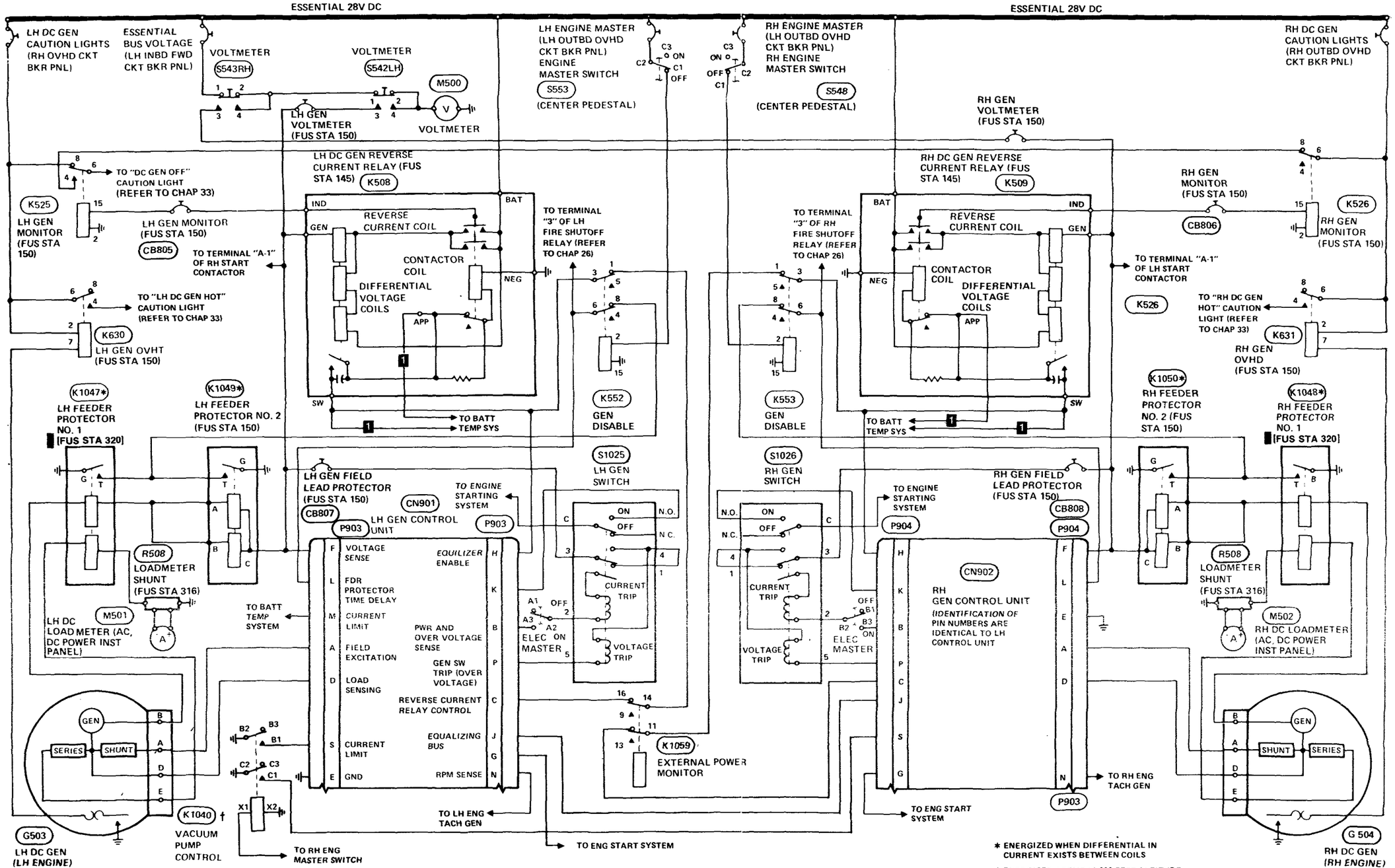
ESSENTIAL 28V DC
LH DC GEN CAUTION LIGHTS (RH OVHD CKT BKR PNL)
ESSENTIAL BUS VOLTAGE (LH INBD FWD CKT BKR PNL)
VOLTMETER S543RH
VOLTMETER S542LH
M500
LH GEN VOLTMETER (FUS STA 150)
VOLTMETER
LH ENGINE MASTER (LH OUTBD OVHD CKT BKR PNL) ENGINE MASTER SWITCH S553 (CENTER PEDESTAL)
RH ENGINE MASTER (LH OUTBD OVHD CKT BKR PNL) RH ENGINE MASTER SWITCH S548 (CENTER PEDESTAL)
ESSENTIAL 28V DC
RH DC GEN CAUTION LIGHTS (RH OUTBD OVHD CKT BKR PNL)
RH GEN VOLTMETER (FUS STA 150)
TO "DC GEN OFF" CAUTION LIGHT (REFER TO CHAP 33)
K525
LH GEN MONITOR (FUS STA 150)
CB805
LH GEN MONITOR (FUS STA 150)
LH DC GEN REVERSE CURRENT RELAY (FUS STA 145) K508
TO TERMINAL "A-1" OF RH START CONTACTOR
TO TERMINAL "3" OF LH FIRE SHUTOFF RELAY (REFER TO CHAP 26)
TO TERMINAL "3" OF RH FIRE SHUTOFF RELAY (REFER TO CHAP 26)
RH DC GEN REVERSE CURRENT RELAY (FUS STA 145) K509
RH GEN MONITOR (FUS STA 150)
CB806
K526
RH GEN MONITOR (FUS STA 150)
TO TERMINAL "A-1" OF LH START CONTACTOR
TO "LH DC GEN HOT" CAUTION LIGHT (REFER TO CHAP 33)
K630
LH GEN OVHT (FUS STA 150)
TO "RH DC GEN HOT" CAUTION LIGHT (REFER TO CHAP 33)
K631
RH GEN OVHD (FUS STA 150)
REVERSE CURRENT COIL
CONTACTOR COIL
DIFFERENTIAL VOLTAGE COILS
IND
BAT
GEN
NEG
APP
SW
TO BATT TEMP SYS
K1047*
LH FEEDER PROTECTOR NO. 1 [FUS STA 320]
K1049*
LH FEEDER PROTECTOR NO. 2 (FUS STA 150)
K552
GEN DISABLE
K553
GEN DISABLE
K1050*
RH FEEDER PROTECTOR NO. 2 (FUS STA 150)
K1048*
RH FEEDER PROTECTOR NO. 1 [FUS STA 320]
LH GEN FIELD LEAD PROTECTOR (FUS STA 150)
CB807
CN901
P903
LH GEN CONTROL UNIT
TO ENGINE STARTING SYSTEM
S1025
LH GEN SWITCH
S1026
RH GEN SWITCH
RH GEN FIELD LEAD PROTECTOR (FUS STA 150)
CB808
P904
R508
LOADMETER SHUNT (FUS STA 316)
M501
LH DC LOAD METER (AC, DC POWER INST PANEL)
TO BATT TEMP SYSTEM
VOLTAGE SENSE
FDR PROTECTOR TIME DELAY
CURRENT LIMIT
FIELD EXCITATION
LOAD SENSING
CURRENT LIMIT
GND
EQUILIZER ENABLE
PWR AND OVER VOLTAGE SENSE
GEN SW TRIP (OVER VOLTAGE)
REVERSE CURRENT RELAY CONTROL
EQUALIZING BUS
RPM SENSE
CURRENT TRIP
VOLTAGE TRIP
ELEC MASTER
CN902
RH GEN CONTROL UNIT
(IDENTIFICATION OF PIN NUMBERS ARE IDENTICAL TO LH CONTROL UNIT
M502
RH DC LOADMETER (AC, DC POWER INST PANEL)
K1059
EXTERNAL POWER MONITOR
TO RH ENG TACH GEN
TO ENG START SYSTEM
TO LH ENG TACH GEN
TO ENG START SYSTEM
TO RH ENG MASTER SWITCH
K1040 †
VACUUM PUMP CONTROL
G503
LH DC GEN (LH ENGINE)
G 504
RH DC GEN (RH ENGINE)
GEN
SERIES
SHUNT

* ENERGIZED WHEN DIFFERENTIAL IN CURRENT EXISTS BETWEEN COILS

† ENERGIZED ABOVE 14,500 FEET ALTITUDE (REFER TO CHAPTER 21)

[1] S/N 380-25 AND SUBS CHANGED BY SB 74-22.

S380-2-54-1D

DC Generator System Electrical Schematic
Figure 24-30A-3

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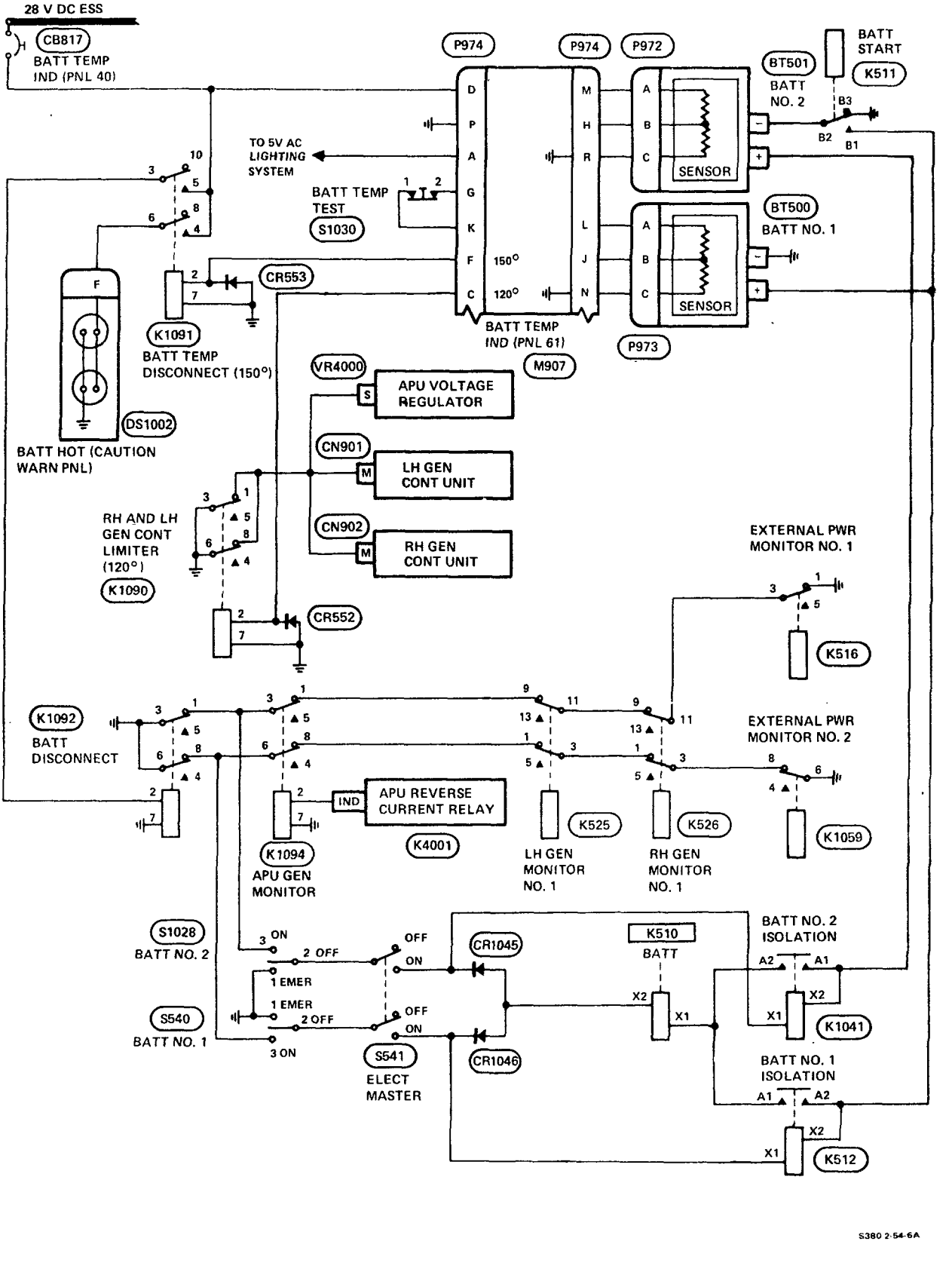

Battery Temperature System Electrical Schematic
S/N 380-1 Thru -24 Without S/B 74-22
Figure 24-30A-4

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2B. Battery Temperature and Control System - 380-1 Thru -24 Without S/B 74-22.

The battery temperature and control system provides a visual indication of a battery overtemperature condition. The system (when actuated) will automatically disconnect the batteries from the battery bus if either battery temperature reaches 150° F. The system consists of two battery temperature sensors (attached to a battery cell interconnect), a battery temperature indicator, a battery hot caution light, and the necessary control relays and electrical wiring to complete the system. **(See figure 24-30A-4.)** During normal aircraft operation, if the temperature of either battery reaches 120°F, the sensor applies a signal to the indicator which in turn amplifies this signal to a sufficient amplitude to energize the generator control limiter relay (K1090). When relay K1090 is energized, a ground circuit is removed from the engine generator control units and APU generator voltage regulator which in turn reduces the generator output voltage from 28.5 volts to 26.5 volts dc. When the generator voltage is reduced, the charging rate to the batteries is reduced. During this mode of operation, the amber caution light on the indicator will be illuminated. With a charging source available, if the battery temperature reaches 150°F, an electrical signal from the sensor is routed to the indicator and from the indicator to the coil of the temperature disconnect relay (K1091). Relay K1091 is then energized which in turn applies electrical power to the battery hot warning light and to the coil of the battery disconnect relay (K1092). Relay K1092 is then energized removing the ground circuit to the battery isolation relays, K512 and K1041 and the battery relay K510. With relays K512, K1041 and K510 de-energized, the batteries are disconnected from the essential bus. During the 150°F overtemperature condition, the automatic disconnect function of the circuit can be overridden by placing either battery switch to the emergency position. This provides a direct ground path to its respective battery isolation relay which in turn energizes the battery relay (K510) connecting the applicable battery to the essential bus.

A battery temperature system test switch is provided to check for proper operation of the system. With the dc generators operating, depressing the battery temperature test switch causes the pointers to pass through 120°F, illuminating the amber caution light and reducing bus voltage approximately 2 volts. When the 150°F temperature is reached, the red warning light will illuminate and the batteries will be removed from the essential bus.

2C. Battery Temperature and Control System - 380-1 Thru -24 With S/B 74-22 and 380-25 and Subs.

The battery temperature and control system provides a visual indication of a battery overtemperature condition. The system (when actuated) will automatically disconnect the batteries from the battery bus if either battery temperature reaches 150°F. The system consists of two battery temperature sensors (attached to a battery cell interconnect), a battery temperature indicator, a battery hot caution light, and the necessary control relays and electrical wiring to complete the system. (See figure 24-30-3B.) During normal aircraft operation, if the temperature of either battery reaches 120°F, the sensor applies a signal to the indicator which in turn amplifies this signal to a sufficient amplitude to energize the RH and LH generator control limiter relay (K1090). With relay K1090 energized, electrical power from the 28 VDC essential bus is routed to the 120°F and test monitor relay (K1096).

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With relay K1096 energized, an electrical circuit is completed to the generator voltage limiter relay (K1095) which is then energized, subsequently removing the ground path from the engine generator control units and APU generator voltage regulator. This action reduces the generator output voltage from 28.5 volts to 26.5 volts dc. When the generator voltage is reduced, the charging rate to the batteries is reduced. During this mode of operation, the amber caution light on the indicator will be illuminated. At the same time contacts 5 and 3 of relay K1096, route power to the battery reverse current delay relay (TM513). Relay TM513, when de-energized, provides a holding circuit for the reverse current relays (through generator monitor relays K1022 and K1023). After a pre-set time delay, relay TM513 energizes, which in turn removes the electrical hold circuit from the reverse current relays, subsequently returning the reverse current relays to normal operation. The time delay relay allows sufficient time for the system to stabilize after the reduced charge rate. With a charging source available, if the battery temperature reaches 150°F, an electrical signal from the sensor is routed to the indicator and from the indicator to the coil of the temperature disconnect relay (K1091). Relay K1091 is then energized, which in turn applies electrical power to the battery hot warning light, and to the coil of the battery disconnect relay (K1092). Relay K1092 is then energized removing the ground circuit to the battery isolation relays, K512 and K1041 and the battery relay K510. With relays K512, K1041 and K510 de-energized, the batteries are disconnected from the essential bus. During the 150°F overtemperature condition, the automatic disconnect function of the circuit can be overridden by placing either battery switch to the emergency position. This provides a direct ground path to its respective battery isolation relay which in turn energizes the battery relay (K510) connecting the applicable battery to the essential bus.

A battery temperature system test switch is provided to check for proper operation of the system. With the dc generators operating, depressing the battery temperature test switch causes the pointers to pass through 120°F, illuminating the amber caution light and reducing bus voltage approximately 2 volts. When the 150°F temperature is reached, the red warning light will illuminate and the batteries will be removed from the essential bus.

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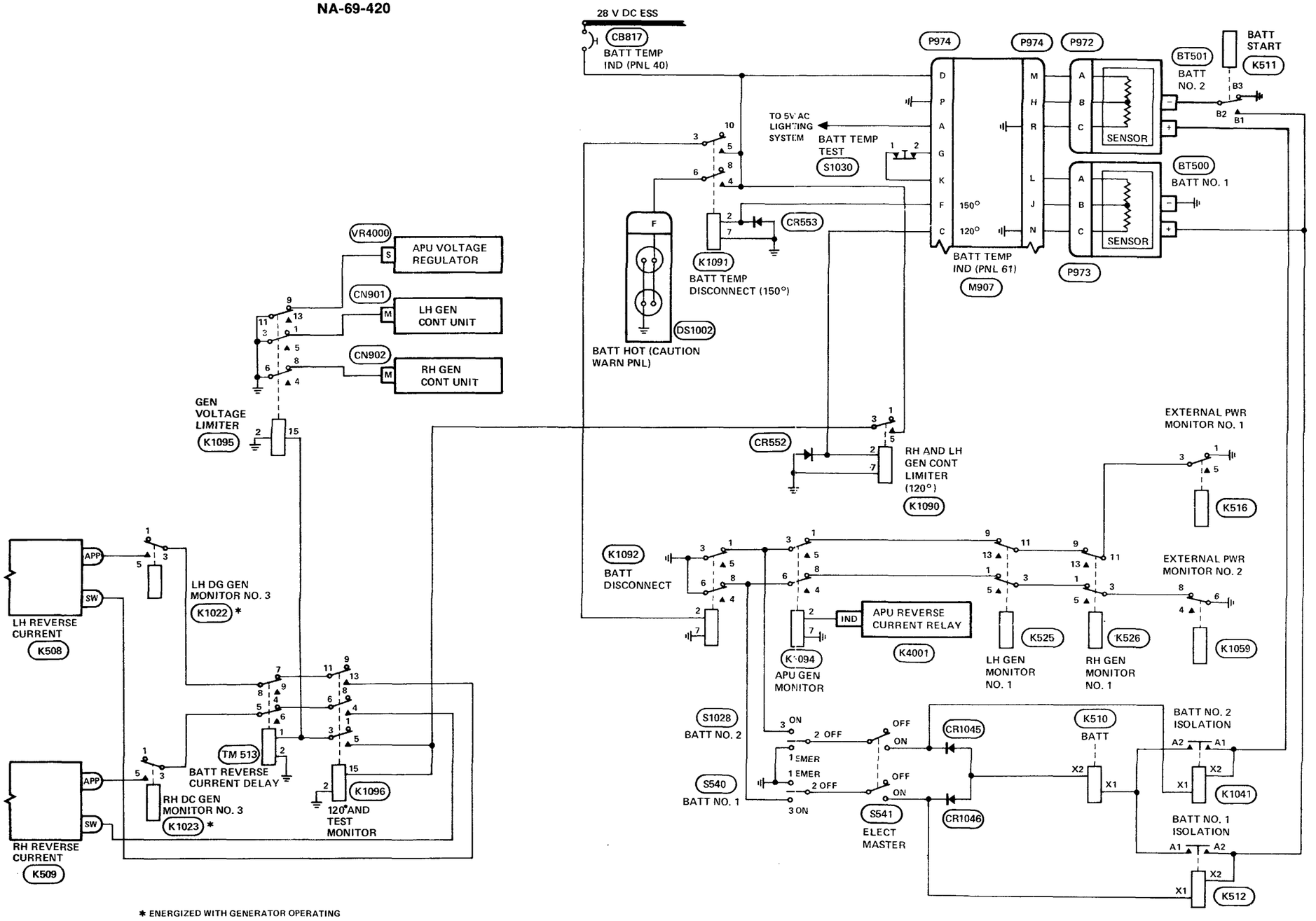

* ENERGIZED WITH GENERATOR OPERATING

S380-2-54-18A

Battery Temperature System Electrical Schematic
S/N 380-25 and Subs and Airplanes With S/B 74-22
Figure 24-30A-5

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3. DC Generator Systems

The dc generator system (figure 24-30A-3) consists of the left and right generator systems. Each generator system consists of a starter-generator, two feeder protectors, a loadmeter, a generator switch, a reverse-current relay, a disable relay, an overheat detection and indicating system, and a generator control unit. A generator failure indicating circuit is common to both generators, consisting of the left and right reverse current relays, the left and right generator monitor relays and the "DC GEN OFF" indicator light. The light indicates a generator failure, but the loadmeters must be checked to determine which generator failed. A dc generator voltmeter circuit with a push-to-test switch for each generator is also common to both generator systems. The voltmeter normally indicates essential bus voltage but when either test switch is depressed the affected generator voltage is indicated.

NOTE: Because both dc generator systems are identical, the operation of the left system only is given.

With the left engine master switch and "DC GENERATOR - LH" switch on and with the engine operating the starter-generator develops 28-volt dc power. This power is applied through the No. 1 and No. 2 feeder protectors to the "GEN" terminal on the reverse current relay. From the "GEN" terminal the 28-volt dc power is routed through the "DC GENERATOR - LH" switch contacts (switch on) and through the current trip coil to pin "B" of the generator control unit (GCU) and through the voltage trip coil to the two feeder protector relays. The power to pin "B" of the GCU is routed through internal relay contacts and out pin "C" of the GCU. Power is then applied through the external power monitor relay (K1059) and the generator disable relay to the "SW" terminal on the reverse current relay and to pin "H" on the GCU. Power on the "GEN" terminals of the reverse current relay energizes the reverse current coil, closing the contacts, and allowing power from the "SW" terminal to energize the contactor coil. This connects generator output through the reverse current relay to the essential bus.

4. Generator Control Unit (GCU)

The generator control unit provides voltage regulation, overvoltage protection, equalization between generator systems, and current limiting, in addition to starter system control.

The voltage regulation circuits maintain generator output at 28.5 (±0.5) volts by controlling field excitation. The overvoltage circuit senses a voltage in excess of 32.0 (±0.2) volts and completes a ground circuit for the overvoltage trip coil in the "DC GENERATOR - LH" switch causing the switch to move to off. The circuit is designed so that if overvoltage is caused by the right generator on the bus, the left generator will not be de-energized. Equalization is accomplished by interconnected differential amplifiers in each GCU so that if one generator is carrying more than half the total electrical load the voltage of that generator is decreased to decrease its load current. Equalization is externally adjustable by a potentiometer in each GCU. Current limiting circuits are provided to

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prevent overloading the engine. At altitudes up to 14,500 feet, current is limited to 300 (±25) amperes at engine rpm below 60 percent. When engine rpm is above 60 percent, the current can increase proportionally as programmed in the GCU at the factory. At altitudes above 14,500 feet, programming is changed so that current is limited to 300 (±25) amperes up to 70 percent rpm. Current is then proportional to engine rpm, but much less than for the low altitude program. The current limiting circuit uses inputs from a tachometer generator to control generator current. Altitude sensing is accomplished through use of the vacuum pump control relay, K1040. (Refer to Section 21.) The anti-runaway circuit provides current to the generator shunt field during generator operation.

5. Generator Feeder Protectors

The generator feeder protectors are relays which are sensitive to current differential between two feeder leads passing through them. If current in one lead exceeds current in the other by 150 to 200 amperes, the feeder protector relay will energize applying a ground pin "L" (through contacts of the generator disable relay) of the GCU. When this ground circuit is applied to pin "L", a time delay circuit (approximately 300 milli seconds) within the GCU is energized. This time delay circuit prevents the generator from being removed from the essential bus in the event of a momentary current surge. If the current differential between the two feeder protector studs remains longer than the time delay, a relay within the GCU is energized providing a ground to pin "P". With pin "P" grounded the voltage trip coil in the generator switch is energized, moving the generator switch to OFF, subsequently removing the dc generator from the essential bus.

6. Generator Switch/Circuit Breaker

The "DC GENERATOR - RH" (S1026) and "DC GENERATOR - LH" (S1025) switches are each provided with a current trip coil and a voltage trip coil, either of which will actuate the switch to the OFF position when energized. The switches can only be turned on manually, but can be turned off electrically or manually. **The current trip coil is energized when generator current exceeds 310 ±10 amperes for one second (limits set during manufacture).**
The voltage trip coil is energized when any of the following conditions exist:

(1) Engine fire extinguishing system fire shutoff relay energized.

(2) Feeder protector No. 1 or No. 2 energized.

(3) Engine reaches 42 percent rpm during start cycle.

7. DC Generator Overheat Detection Systems

Each of two dc generator overheat detection systems **(figure 24-30A-1)** is an integral part of each dc generator system. Each detection system consists of a thermoswitch-type detector, an overheat relay, and two indicator lights. During normal generator system operation, the detector (mounted on the generator housing) senses the generator housing temperature. Whenever the housing temperature rises above 155°C (311°F), the detector contacts close and complete the ground connection for the overheat relay. The relay then becomes energized and connects essential bus dc power to the overheat detector lights. The lights come on, indicating a dc generator overheat condition. The detection system indicates an overheat condition only. The overheated generator system must be manually turned off by placing the generator switch at OFF.

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8. DC Generator Failure Indicating System

The dc generator failure indicating system (figure 24-30A-1) is common to both dc generator systems. It consists of the left and right generator monitor relays, the left and right generator reverse-current relays, and one indicator assembly. During normal operation, both generator system reverse-current relays and monitor relays are energized and the indicator light is extinguished. If either generator system fails, the corresponding generator reverse-current relay and monitor relay become de-energized. The de-energized monitor relay connects essential bus voltage to the indicator light. The light then comes on, indicating a generator system failure. To determine which generator system has failed, observe the dc loadmeters on the right inboard instrument subpanel. The generator system having a very low or no loadmeter reading is the failed system. The generator system that remains operational will have a high loadmeter reading.

9. DC Generator Voltmeter

The dc generator voltmeter and push to test switches are on the electrical power panel above the generator switches. The voltmeter indicates essential bus voltage normally, but indicates generator voltage when the desired generator voltage test button is depressed. The generator voltage test buttons are located below and on each side of the voltmeter.

10. Batteries

Two 24-volt, 34-ampere-hour nickel-cadmium batteries are mounted in the nose equipment compartment forward of fuselage station 50. They are accessible through an access door on the left side. (See figure 24-30A-1.) Quick-disconnect plugs connect the battery cables to the batteries. The metal cases of the batteries are vented to atmosphere through flexible rubber tubing connected to vents on the skin of the fuselage. The inlet vent connects into the battery case. Battery fumes are expelled through an outlet connection on each battery case into a sump jar. The jar is mounted between the batteries. Battery fumes are neutralized by an impregnated felt pad inside the sump jar. The pad contains a solution of distilled water and boric acid (3 percent by weight). The neutralized fumes are expelled overboard through the outlet vent on the underside of the fuselage. The difference in air pressure between the inlet and outlet vents causes air to flow from the inlet vent to the outlet vent.

NOTE: It is recommended that the battery system should not be used for ground operation of airplane dc equipment. Whenever maintenance on the airplane requires dc power, a dc external power source should be connected to the utility external power receptacle.

- A dc external power source can be used to charge the batteries, provided the "ELEC MASTER" and "BATTERY" switches are placed at ON when the external power source is connected to the utility external power receptacle.

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11. DC Generators

Each of the two engine-driven, 30-volt, 300-ampere starter-generators is installed on an engine accessory drive pad. (See figure 24-30A-3.) Each starter-generator is attached to the engine accessory driven pad by a quick attach disconnect adapter (QAD). The starter-generator is secured to the QAD by a clamp-type band. Guide pins on the QAD determine the position of the generator on the QAD. An outlet assembly and an inlet assembly are provided on the generator for cooling ram air. The inlet mates with ram air ducting in the engine cowling and the outlet is attached to an overboard duct routed to the main drain cluster in the cowling. Ram air cooling is effective when the airplane is moving. With the airplane stationary, cooling is accomplished by an internal fan in the generator.

12. DC Generator Reverse-current Relays

The reverse current relays are mounted on the second shelf from the top (No. 102) in the electrical equipment rack on the right side of the fuselage at fuselage station 150. Each reverse-current relay contains a differential voltage relay, and a contactor relay. (See figure 24-30A-3.) When a generator system is operating, the reverse-current relay connects the output of the generator to the essential bus if the generator output polarity is correct, if the generator output voltage exceeds 22 volts, or if the generator output voltage exceeds the essential bus (battery and other generator) voltage by at least 0.35 volt. The coil of the differential relay senses the generator output polarity and voltage. When these are correct, the differential relay contacts close and energize the contactor relay. The contactor relay connects the output of the generator to the essential bus through the reverse-current coil. Whenever the generator output voltage drops below the essential bus voltage, current flows through the reverse-current coil in a direction opposite to the normal generator current. When this reverse current exceeds 18 to 35 amperes, the differential relay contacts open and automatically de-energize the contactor relay. The de-energized contactor relay disconnects the faulty generator system from the essential bus.

13. DC Generator Loadmeters

The two dc generator loadmeters are on the right inboard instrument subpanel. (See figure 24-30A-3.) They are accessible from the cockpit area. When the airplane is standing on the ramp with both dc generator systems operating (and no dc external power connected to the airplane), each loadmeter should read about 0.15 (15 percent). During normal taxi conditions, each loadmeter should read about 0.33 (33 percent). During normal flight conditions, each loadmeter should read about 0.45 (45 percent). If one dc generator system is turned off (or fails), its loadmeter will read 0.00 (0 percent). The loadmeter in the remaining operational generator system will then read the total load, or about twice its normal reading, depending upon the operating condition of the airplane.

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DC GENERATION - TROUBLE SHOOTING

CAUTION: WHEN TROUBLE SHOOTING AN ELECTRICAL CIRCUIT, IT MAY BE NECESSARY TO REMOVE TERMINAL BLOCK COVERS AND DISCONNECT AIRPLANE WIRING FROM TERMINAL STUDS. WHEN SUCH A PROCEDURE IS NECESSARY, DISCONNECT ALL POWER FROM THE CIRCUIT BEFORE MAKING ANY ATTEMPT TO TROUBLE SHOOT THE CIRCUIT. WHEN TROUBLE SHOOTING, DO NOT APPLY POWER TO CIRCUIT UNTIL AFTER IT HAS BEEN DETERMINED THAT NO DAMAGE WILL OCCUR TO AIRPLANE EQUIPMENT OR WIRING. ALWAYS RECONNECT ANY TEMPORARILY DISCONNECTED CIRCUIT WIRING AT COMPLETION OF TROUBLE SHOOTING PROCEDURE. FOR WIRING HOOKUP INFORMATION, REFER TO WIRING DIAGRAM MANUAL.

1. Trouble Shooting DC Power Generation System

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
	BATTERY	
A. LOSS OF CAPACITY.		
(1) Discharged battery.	Check ampere charging rate. If more than 10 amperes (5 amperes per battery), the batteries require charging until below 10 amperes.	Charge battery.
(2) Internal short circuit in one or more cells.	Check voltage output of each cell. If cell voltage is 1.05 volts or less, cell is discharged.	Replace discharged battery cell. (Cells are removable and replaceable.)
(3) External short circuit.	Check airplane wiring harness for short circuit.	Repair short circuit. Recharge battery.
(4) Extended service life.	Check voltage of each cell.	Replace battery.
	DC GENERATOR	
B. NO OUTPUT VOLTAGE FROM GENERATOR.		
(1) Internal generator short circuit.	Make voltage and continuity check of generator.	Replace generator.
(2) Generator system wiring short circuit, or open circuit,	Make continuity check of generator system wiring.	Repair short circuit, or open circuit.
(3) Generator control unit (GCU) faulty.	Disconnect electrical plugs from both units and reconnect to opposite unit (switch units) then	Replace faulty GCU.

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PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
	check generator operation. If suspected generator functions properly but previously good one now malfunctions, GCU is faulty.	
C. LOW OUTPUT VOLTAGE FROM GENERATOR.		
(1) GCU out of adjustment.	Check generator output voltage.	Adjust GCU.
(2) GCU faulty.	Attempt voltage adjustment. If not successful, switch GCU electrical connectors to see if trouble changes also.	Replace GCU.
(3) Loose internal connection in generator.	Make voltage output check of generator.	Replace generator.
(4) Loose wiring harness connection.	Make voltage output check of generator.	Tighten loose connection.
(5) Grease, dirt, or oil on generator commutator.	Make voltage output check of generator. Visually check condition of commutator.	Replace generator.
D. FLUCTUATING OUTPUT CURRENT OR VOLTAGE FROM GENERATOR.		
(1) Loose brush connection.	Visually inspect brush connections.	Tighten connection.
(2) Incorrectly fitted brushes.	Visually inspect brush wear and fit.	Replace generator.
(3) Internal failure of generator.	Make an operational checkout of generator system under load.	Replace generator.
(4) GCU faulty.	Switch electrical connectors on GCU's and check to see if trouble follows.	Replace GCU.
E. REVERSE-CURRENT RELAY WILL NOT CLOSE AT REGULATED VOLTAGE.		
(1) Generator control switch at OFF.	Visually inspect switch position.	Momentarily place switch at RESET, then at ON.

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PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
(2) Generator voltage output too low.	Check generator output voltage.	Adjust voltage at GCU to 28.0 (±0.1) volts dc.
(3) Improperly adjusted relay.	Check generator output voltage on essential bus.	Replace relay.
F. RELAY WILL NOT OPEN ON REVERSE CURRENT OF 18 TO 35 AMPERES.		
(1) Improperly adjusted relay.	Check battery voltage on "GEN" terminal of relay.	Replace relay.
G. GENERATOR OVERVOLTAGE TRIP INOPERATIVE.		
(1) Faulty generator switch.	Perform continuity check of generator switch voltage trip coil.	Replace switch.
(2) Faulty GCU.	Switch GCU electrical connectors and check to see if trouble follows.	Replace GCU.
H. GENERATOR OVERCURRENT TRIP INOPERATIVE.		
(1) Faulty generator switch.	Perform continuity check of generator switch current trip coil.	Replace switch.
(2) Faulty GCU.	Switch GCU electrical connectors and check to see if trouble follows.	Replace GCU.
I. NO LOADMETER READING.		
(1) Loose connection.	Visually inspect connections.	Tighten connections.
(2) Open circuit.	Make power-off continuity check of loadmeter circuit.	Repair circuit.
(3) Faulty loadmeter.	Make continuity check of loadmeter.	Replace loadmeter.
J. GENERATORS WILL NOT STAY PARALLELED.		
(1) Misadjusted GCU.	Make an operational checkout of dc generator system.	Make voltage and paralleling adjustments to voltage regulators.

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PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
(2) Faulty GCU.	Attempt voltage and paralleling adjustments to voltage regulators.	Replace GCU. Make an operational check-out of dc generator systems. Adjust as required for parallel operation.
(3) Loadmeters not calibrated.	Temporarily interchange loadmeters; then make an operational check-out of dc generator system.	Replace loadmeters with calibrated loadmeters. Make voltage and paralleling adjustments to voltage regulators.

2. Trouble Shooting Battery Temperature System

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. BATTERY CURRENT DOES NOT REDUCE AT 120°F TEMPERATURE.		
(1) Faulty RH and LH generator control limiter relay (K1090).	Apply 28 volts dc to relay coil and check for actuation of relay.	Replace relay.
(2) Battery temperature indicator (M907) faulty.	Remove sensor and apply heat lamp until sensor is actuated. Check for 28 volts dc at pin "C" of P974. If voltage is not present, battery temperature indicator (M907) is faulty.	Replace indicator.
(3) Sensor faulty.	Check for continuity across sensor.	Replace sensor.
B. BATTERIES NOT ISOLATED FROM GENERATOR CIRCUIT AT 150°F TEMPERATURE.		
(1) Faulty temperature sensor.	Refer to ISOLATION PROCEDURE of A. (3).	Replace sensor.
(2) Faulty battery disconnect relay(s) K1092 or K1091.	Remove P974 from battery temperature indicator. Apply 28 volts dc to pin "F" of P974. Check for 28 volts dc at Terminal 2 of K1091 and K1092.	Replace relay(s).

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1. Checking, Cycling and Repairing Battery

CAUTION: SERVICING AND REPAIR PROCEDURES ON NICKEL-CADMIUM BATTERIES MUST BE PERFORMED IN AN AREA DESIGNATED STRICTLY FOR THIS TYPE OF BATTERY. UNDER NO CIRCUMSTANCES SHALL LEAD-ACID BATTERY SERVICING TOOLS BE USED TO SERVICE NICKEL-CADMIUM BATTERIES. ALSO, FUMES FROM LEAD-ACID BATTERIES MUST NOT BE PERMITTED IN THE AREA DESIGNATED FOR NICKEL-CADMIUM BATTERY SERVICING AND REPAIR. FAILURE TO OBSERVE THIS CAUTION WILL RESULT IN A CONTAMINATED NICKEL-CADMIUM BATTERY. CONTAMINATION WILL RENDER THE BATTERY USELESS IN A VERY SHORT TIME.

- ON NA-380-1 AND LATER AIRPLANES, WHEN MOVING THE AIRPLANE IN REVERSE, EXTREME CAUTION SHOULD BE USED TO PREVENT THE NOSE WHEELS FROM RAISING OFF THE GROUND. THE AIRPLANE IS MORE SUSCEPTIBLE TO THIS CONDITION WITH THE BATTERIES REMOVED AND/OR WITH PERSONNEL IN THE CABIN AREA. IF THE BATTERIES ARE REMOVED AND PRIOR TO MOVING THE AIRCRAFT, AN EQUIVALENT AMOUNT OF BALLAST SHOULD BE INSTALLED.

NOTE: To obtain the maximum service life of a nickel-cadmium battery, proper maintenance should be performed at specific intervals (refer to Chapter 5); however, each airplane is subjected to different operating and environmental conditions. In high ambient temperatures and/or flights of short duration, battery maintenance may be performed more frequently as required. This increase in maintenance should prolong battery life.

- The following procedures provide general information for checking, cycling, and repair of nickel-cadmium batteries (removing, checking, discharging, disassembling, cleaning, checking parts, adjusting specific gravity, assembling, charging, cycling, testing and installing). It is suggested; however, that, as applicable, the maintenance instructions provided by the battery manufacturer be followed when such instructions are available.

A. Removing Batteries

WARNING: USE CARE TO AVOID INJURY WHEN LIFTING BATTERIES. EACH BATTERY WEIGHS 84 POUNDS.

- WEAR PROTECTIVE CLOTHING TO AVOID CAUSTIC BURNS. WHEN MOVING BATTERIES USE CARE TO AVOID SPILLING ELECTROLYTE.

CAUTION: TO PREVENT AN AFT CG CONDITION, AN EQUIVALENT AMOUNT OF BALLAST SHOULD BE INSTALLED AT EACH BATTERY LOCATION SUBSEQUENT TO BATTERY REMOVAL.

(1) Remove all electrical power from airplane.

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(2) In cockpit, place both BATTERY switches at OFF.

(3) Remove access doors from each side of battery compartment.

(4) Disconnect battery cable quick disconnect connectors and sensor plugs. Install dust cap on sensor plugs. Temporarily insulate and stow cable quick disconnects.

(5) Disconnect vent hoses from batteries and from sump jar. Remove sump jar and loose vent hoses from airplane. Protect all openings.

(6) Remove battery tie-down straps and rods.

(7) Remove batteries from airplane.

B. Checking Battery

Examine battery for evidence of the following conditions:

(1) Damage and distortion of battery case and cover.

(2) Burned or damaged contacts on external electrical power receptacle.

(3) Cracks and evidence of leakage on top of each battery cell.

(4) Security of cell connectors. Check condition and security of battery temperature sensor(s).

(5) Evidence of corrosion on and around cell connectors and cell vent caps.

(6) Cracks, corrosion, deterioration, and leakage of all cell vent caps.

C. Discharging Battery

NOTE: The battery must be fully discharged before any attempt is made to disassemble the battery. Following the addition of distilled water to the battery cell electrolyte, the battery must be discharged as part of the cycling procedure required to thoroughly mix the electrolyte with the distilled water added to the cells. The battery is also discharged to reveal the presence of a faulty cell or cells.

To discharge battery, proceed as follows:

(1) Obtain a suitable load bank which is continuously variable from 0 to 5 ohms, has a 600-watt capacity, and is equipped with a dc ammeter capable of reading 20 amperes of discharge current.

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(2) Connect load bank to battery, and adjust load bank resistance to cause a discharge current as applicable to battery installed: 11 amperes for 22-ampere-hour battery or 17 amperes for 34-ampere-hour battery.

(3) Time discharge operation for a 2-hour period.

(4) At 15-minute intervals during discharging procedure, check each individual battery cell terminal voltage. Use a 0- to 5-volt dc voltmeter.

(5) Mark (or otherwise identify) any cell whose terminal voltage drops below 0.95-volt dc during 2-hour discharging period.

NOTE: All cells having a terminal voltage of less than 0.95-volt dc at the completion of the 2-hour discharge period (or any cells that reverse polarity) are considered to be suspect and should be marked for possible replacement. (Refer to procedures for cycling battery and testing battery capacity for further maintenance procedures.)

(6) After completion of 2-hour discharge period, resume discharging procedure, but at a 5-ampere rate. (Readjust load bank resistance to establish 5-ampere rate.)

(7) Continue checking individual cell terminal voltages periodically.

(8) When terminal voltage of a cell drops to 0.50-volt dc, short out cell with a suitable jumper wire.

(9) When all cells have been discharged and shorted out with individual jumper wires, disconnect load bank from battery.

(10) Leave individual jumper wires connected across battery cells for at least 8 hours, or overnight.

(11) Remove jumper wires from battery cell terminals.

D. Disassembling Battery

NOTE: Before disassembling battery, carefully note positions of cell connectors and cell separators for subsequent reassembly. (See figure 24-30A-6.) Use insulated (or taped) hand tools to remove cell connectors. Use battery cell puller (figure 24-30A-9) to remove cells from battery case.

(1) Disassemble battery as required (see figure 24-30A-6).

(a) After battery cover is removed, loosen all cell vent caps. (Use cell vent cap wrench.) Retighten cell vent caps before proceeding with disassembly.

(b) Use cell puller (figure 24-30A-9) to remove battery cells from battery case.

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(c) Stow battery cells in an upright position to avoid possible spillage of cell electrolyte.

(d) Remove cell separators as required during the disassembly procedure.

E. Cleaning Disassembled Battery

(1) Remove any corrosion from all re-usable parts by brushing parts with a stiff-bristled fiber brush.

(2) Wash all re-usable parts with a 3 percent solution (by weight) of boric acid and distilled water. Exercise care to prevent entry of solution into battery cells. (Make sure cell vent caps are tight.)

(3) Thoroughly dry all re-usable parts.

F. Checking Disassembled Battery Parts

NOTE: Replace all parts found to be faulty when checked as outlined in steps (1) through (5).

(1) Thoroughly check all battery parts for any damage, cracks, deterioration, or excessive wear.

(2) Check all threaded parts for crossed or damaged threads.

(3) Check individual battery cell cases for evidence of leakage.

(4) Check battery manufacturer's part number on side of each cell case. All cells should have the correct part number for the battery in which cells are to be installed.

(5) Replace all battery cells found to be unserviceable during discharge procedure.

G. Adjusting Specific Gravity of Cell Electrolyte

To check specific gravity of electrolyte in each battery cell, proceed as follows:

(1) Wear proper protection during this procedure, such as a face shield, rubber gloves, and a rubber apron.

(2) Obtain a glass bowl which is absolutely clean.

(3) Loosen and remove battery cell vent cap (use cell vent cap wrench); then carefully and slowly empty cell electrolyte from cell into glass bowl. Exercise care to avoid splashing and loss of electrolyte.

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(4) Obtain a hydrometer designated only for nickel-cadmium battery servicing.

(5) Insert end of hydrometer into electrolyte contained in glass bowl. Suck up electrolyte into barrel of hydrometer.

(6) While holding hydrometer in a vertical position over glass bowl, note specific-gravity reading. The reading should be between 1.24 and 1.32. Return electrolyte to glass bowl.

(7) If specific gravity of electrolyte measures above 1.24, return electrolyte to battery cell. (Use hydrometer to transfer electrolyte from glass bowl to battery cell.)

(8) Install cell vent cap on battery cell. (Use cell vent cap wrench.) Do not overtighten cell vent cap.

(9) If specific gravity of cell electrolyte is below 1.24, obtain a 30 percent solution (by weight) of potassium hydroxide and distilled water.

(10) Mix a small portion of 30 percent solution into cell electrolyte in glass bowl. (Do this slowly and carefully, and gently stir electrolyte while adding solution.)

CAUTION: IT IS STRONGLY RECOMMENDED THAT THE 30 PERCENT SOLUTION OF POTASSIUM HYDROXIDE AND DISTILLED WATER BE OBTAINED IN READY-MIXED FORM, RATHER THAN ATTEMPTING TO LOCALLY MIX THE SOLUTION. IMPROPER MIXING PROCEDURES MAY RESULT IN A VIOLENT EXPLOSIVE REACTION.

(11) Repeat steps (5) and (6) to recheck specific gravity of electrolyte.

(12) When specific gravity of electrolyte is within specifications, return electrolyte to battery cell.

(13) If required, repeat steps (3) through (12) for remaining battery cells.

(14) Check electrolyte level in all battery cells. Adjust level, if necessary, using hydrometer or syringe.

H. Assembling Battery

(1) Assemble battery in reverse order of disassembly (figure 24-30A-6), noting the following:

(a) Check polarity of each cell before installing cell in battery case.

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(b) Check manufacturer's part number on side of each cell, before installing cell in battery. All cells must have the correct part number for the battery in which the cell is to be installed.

(c) Install battery cells according to polarity diagram shown in figure 24-30A-6.

(d) Insert cell separators (if removed) as required during the assembly procedure.

(e) Tighten cell connector socket-head cap screws to a torque value not to exceed 35- to 50-inch pounds.

I. Charging Battery

To charge battery after it has been assembled, proceed as follows:

(1) Connect battery to a shop dc generator system. Observe proper polarity connections to dc generator system (positive terminal of battery connected to positive terminal of dc generator).

(2) Turn on dc generator system, and adjust to an output of 28.5-volt dc. Note dc generator system charging current periodically.

(3) When charging current decreases to approximately 10 amperes or less, turn off dc generator system temporarily.

(4) Connect a 0- to 25-ampere dc ammeter in series with battery. Observe proper polarity in connection to dc ammeter.

(5) Turn on dc generator system, and resume charging procedure.

(6) Observe 0- to 25-ampere dc ammeter periodically. When ammeter indicates a charging current of one ampere or less, turn off dc generator system.

(7) Disconnect battery from dc generator system.

J. Cycling Battery

The battery cycling procedure consists of completely discharging, then recharging, the battery. A battery should be cycled immediately after distilled water has been added to the battery cell electrolyte. The battery should also be cycled whenever it becomes necessary to attempt correction of a cell unbalanced condition, or to correct a reversal of polarity of one or more cells in the battery. To cycle battery, proceed as follows:

(1) Completely discharge battery. (Refer to procedure for discharging battery.)

(2) Charge battery. (Refer to procedure for charging battery.)

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K. Testing Capacity of Battery

To test capacity of a battery, proceed as follows:

NOTE: Make sure that battery is fully charged and that electrolyte level in battery cells is correct, before making a capacity test of battery.

(1) Obtain a suitable load bank (continuously variable from 0 to 5 ohms), having a 600-watt capacity, and equipped with a dc ammeter capable of reading 20 amperes of discharging current.

(2) Connect load bank to battery. (Observe proper polarity so that dc ammeter will indicate discharging current correctly.)

(3) Adjust load bank resistance to cause a discharge current of 17 amperes.

(4) Time discharge operation for a 2-hour period.

(5) At 15-minute intervals during 2-hour discharging period, check terminal voltage of battery. (Use a 0- to 25-volt dc voltmeter.) If battery terminal voltage drops below 18.0-volt dc before completion of 2-hour period, discontinue test. This indicates that the battery has failed the capacity test.

(6) At completion of 2-hour discharging period, disconnect load bank from battery.

(7) Measure battery terminal voltage. The voltage should measure 18.0-volt dc or above, if the battery has passed the capacity test.

(8) Recharge battery. (Refer to procedure for charging battery.)

NOTE: A battery which fails to pass the capacity test may have a cell (or cells) unbalance condition, or a reversal of polarity of one or more cells. This condition may possibly be corrected by cycling the battery. After cycling, the battery should be given the capacity test again. If the battery fails to pass the second capacity test, the battery should be fully discharged, and defective cells should be replaced. After cells have been replaced, the battery should be charged and then given the capacity test.

L. Installing Batteries

WARNING: WEAR PROTECTIVE CLOTHING TO AVOID CAUSTIC BURNS. WHEN MOVING BATTERIES USE CARE TO AVOID SPILLING ELECTROLYTE.

- USE CARE TO AVOID INJURY WHEN LIFTING BATTERIES. EACH BATTERY WEIGHS 84 POUNDS.

(1) Remove all electrical power from airplane.

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(2) In cockpit, check that both BATTERY switches are positioned at OFF.

(3) Inspect battery compartment for cleanliness and inspect battery support trays to ensure that insulator strips are in place.

(4) Check that battery covers are securely fastened.

(5) Place batteries onto support trays. Make sure that connectors are facing outboard on each side.

(6) Secure batteries in place with tie-down straps and rods.

(7) Service sump jar (if required) and install sump jar and all vent hoses.

(8) Check quick disconnect plug pin sockets for looseness. Insert a steel pin 0.375 ± 0.0005 inch diameter) a distance of one-half inch into each socket. Record force required to remove pin. The minimum removal force is one pound. The desired removal force is 5 ± 2 pounds.

(9) Install cable connectors and sensor plugs. Safety wire.

(10) Inspect compartment and components for cleanliness and security.

(11) Close access doors.

(12) Perform test of battery system.

(13) Perform operational check-out of battery temperature system.

2. Servicing Battery Vent System

A. To service battery sump jar, proceed as follows:

(1) Remove access door on left side of fuselage nose.

(2) Remove vent hoses from sump jar. Inspect tubes for visible damage and/or obstructions.

(3) Remove bolts securing sump jar to tray.

(4) Remove sump jar pad.

(5) Saturate pad with a 3 percent (by weight) solution of boric acid and water. Use only enough solution to saturate pad.

(6) Replace pad in sump jar and install sump jar lid.

(7) Place sump jar on battery tray and secure with bolts.

(8) Connect battery vent hoses.

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(9) Install access door.

B. Service battery system vent and drain tubes as follows:

WARNING: DO NOT PUT MOUTH IN DIRECT CONTACT WITH AN OPEN END OF A BATTERY SYSTEM VENT OR DRAIN TUBE AS A MEANS TO BLOW CLEAR ANY OBSTRUCTION IN TUBE. FAILURE TO OBSERVE THIS WARNING CAN RESULT IN SEVERE MOUTH AND THROAT BURNS TO MAINTENANCE PERSONNEL.

(1) Observe battery servicing precautions.

(2) Disconnect lower end of vent or drain tube and place it in a glass jar.

NOTE: Do not permit any accumulation of electrolyte or moisture in vent or drain tube to drip onto airplane structure.

(3) Apply dry air pressure to upper end of vent or drain tube.

NOTE: Use a 2 to 5 psi dry-air pressure source to clear any obstructions in vent or drain tubes.

(4) Continue blowing dry air down vent or drain tube to remove obstruction.

(5) If obstruction cannot be removed, completely disconnect section of vent or drain tube and remove it from airplane. Replace faulty section with a new tube.

(6) Connect lower end of vent or drain tube; then remove glass jar from airplane.

3. Nickel - Cadmium Battery Inspection

NOTE: Polystyrene cell cases found in any battery must be replaced with an equivalent cell having a nylon cell case or the battery must be replaced with an equivalent battery which contains approved nylon cells.

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A. To inspect batteries, proceed as follows:

(1) Remove access door on left side of fuselage nose.

(2) Disconnect tiedown rods, unlatch and lift battery cover.

(3) Inspect cell links and cell tops for evidence of heat damage.

(4) If battery is found to have evidence of heat damage, the battery must be replaced with a serviceable battery. Service battery sump jar if necessary.

(5) Close and latch battery cover and secure with tiedown rods.

(6) Install access door.

3A. Battery Maintenance Following An Overtemperature Condition

WARNING: NICKEL-CADMIUM BATTERIES ARE SUBJECT TO PHYSICAL DAMAGE AND GASSING IF OPERATED (CHARGED) AT HIGH TEMPERATURES. THERMAL RUNAWAY DURING CHARGING CAN RESULT IN BATTERY FIRE OR EXPLOSION.

NOTE: Thermal runaway of a nickel-cadmium battery can occur only when connected to a charging source. Continual charging at high temperatures can cause decomposition of the water (electrolysis) resulting in the generation of additional heat. A high temperature lowers internal resistance resulting in a higher charging current flow which, in turn, can cause a faster rate of gassing (water decomposition) and more heat. If allowed to continue, this condition can lead to complete self destruction and severe internal arcing.

- Improper liquid level, cell imbalance, loose intercell connectors, and damaged or leaking cells can cause excessive battery operating temperatures.

A. Procedure Following a Battery Hot Indication

If a battery HOT condition was indicated (during flight or check-out of airplane) by the battery temperature indicator, the affected battery (or batteries) should be removed from the airplane, for checking, cycling, and (if necessary) repair. (Refer to "Checking, Cycling and Repairing Battery".)

B. Procedure Following a Battery Warm Indication

NOTE: If the battery temperature indicator WARM light illuminated during a flight, the batteries should be inspected to prevent possible battery failure.

(1) Remove all external electrical power from airplane.

(2) In cockpit, place both BATTERY switches at OFF.

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(3) Remove battery compartment access doors from either side of fuselage nose.

(4) Loosen and remove battery tiedown straps, and unlatch and remove covers from both batteries.

(5) Inspect battery cell tops and intercell connectors for any evidence of heat damage, electrolyte spewage, and cell damage (cell distortion or leakage).

(6) Using a vent cap tool, loosen cell vent caps.

NOTE: Do not attempt to measure liquid levels if battery is (discharged) as liquid is absorbed into plates resulting in a false liquid level indication.

- Do not attempt to measure specific gravity of electrolyte.

(7) Using a clean polystyrene tube, check liquid level in each cell. The liquid level should be about 1/8-inch above top of cell plates (1/4-inch immediately after receiving a charge, 1/8-inch three or more hours after receiving a charge). To use tube, insert end of tube through vent opening (remove vent cap) until stopped by plates. Place finger over top of tube and carefully withdraw tube until lower end can be seen. (Do not move tube away from vent opening since electrolyte might spill out.) Observe height of liquid in lower end of tube, then release liquid into cell by releasing finger from top of tube. (The tube used should be straight, about six inches long, about 1/8-inch inside diameter, and open at both ends. Tube may be of glass, but polystyrene is recommended. Tube must be clean and must never have been used with acid.)

CAUTION: WATER USED IN NEXT STEP MUST BE FREE OF ANY CONTAMINATES. CONTAMINATION WILL RENDER THE BATTERY USELESS IN A VERY SHORT TIME.

(8) If liquid observed in tube (step 7) is less than required add a small amount of distilled or demineralized water. If liquid is too low to be observed in tube, the battery should be removed from the airplane for checking and cycling.

(9) If inspection (step 5) reveals a slight discoloration at ends of intercell connectors, torque cell connector socket-head screws to 35-50 inch-pounds.

(10) If inspection (step 5) reveals an unsatisfactory condition (other than noted in steps 8 and 9) remove battery for checking, cycling, and repair.

(11) If inspection (steps 5 and 7) reveals no unsatisfactory condition, replace and tighten vent caps (use vent cap tool), replace and secure battery covers, and replace battery tiedown straps.

(12) Service battery sump jar (if necessary) and install access doors.

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4. Removing DC Starter-Generator

NOTE: Each starter-generator is installed on a quick-attach-detach (QAD) adapter on an accessory drive pad on the lower left side of the respective engine. Except for removal of the lower cowling sections (lower outboard on left engine and lower inboard on right engine), procedures for removing and installing the starter-generator are the same for both engines.

A. To remove dc starter-generator, proceed as follows:

(1) Make sure all switches are off and external electrical power disconnected.

(2) Open engine cowling section as applicable.

(3) Remove screw securing starter-generator cooling air duct to engine overboard drain manifold.

(4) Remove clamp securing cooling air duct to starter-generator air discharge fitting and remove duct.

(5) Lift rubber boot from starter-generator electrical terminals and slide up on electrical leads to gain access to terminals.

(6) Disconnect electrical leads from starter-generator terminals. Tag leads for terminal identification during installation procedure.

(7) Loosen QAD clamp bolt and remove clamp and starter-generator.

5. Installing DC Starter-Generator

A. Flashing DC Starter-Generator Field

NOTE: Before installing a starter-generator, flash the exciter shunt field (in proper direction) to make certain that shunt field retains sufficient residual magnetism. Loss of residual magnetism will result in failure of the starter-generator to build up to proper voltage when operating as a dc generator.

- Flash shunt field before installing starter-generator in airplane.
- A 12-volt battery and a single-pole, single-throw knife switch are required.
- Do not make or break battery connection to starter-generator at "A" or "E-" terminal on starter-generator. To do so will result in severe damage to terminals.

(1) Connect an insulated wire between negative (-) terminal of a 12-volt battery and "E-" terminal on starter-generator.

(2) Connect an insulated wire between positive (+) terminal of battery and one side of single-pole, single-throw knife switch.

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MAINTENANCE PRACTICES (Continued)

3. Open knife switch.
4. Connect an insulated wire from other side of knife switch to "A" terminal on starter-generator.
5. Close knife switch for 3 to 5 seconds, then open knife switch.
6. Repeat step (5) several times, to make certain that shunt field is properly flashed.
7. Disconnect knife switch and wiring from starter-generator.
8. Install starter-generator.

To install DC starter-generator, proceed as follows:

1. Make sure all switches are off and external electrical power disconnected.

NOTE

If QAD mounting adapter has been removed, install gasket (AN4047-1) and adapter and torque retaining nuts 230 to 325 inch-pounds

2. Lubricate starter-generator drive spline with Plastilube Moly No. 3 or equivalent grease.

2A. Check oil drain hole, located at lowest point of ram air inlet duct for any obstruction. Clean hole as required.

3. Position starter-generator with electrical terminals and ram air inlet duct facing outboard, then with starter-generator level, move forward and engage starter-generator splines carefully with accessory drive pad.
4. Engage starter-generator alignment pin with QAD hole then install QAD clamp.
5. While supporting aft end of generator, tighten clamp bolt. Torque clamp bolt to 150 inch-pounds; then back off bolt. Repeat torquing operation several times until clamp is fully seated when proper torque is reached.
6. Visually check starter-generator installation to ensure that clamp is properly seated and starter-generator seated on adapter and properly indexed.
7. Safety QAD clamping bolt with lockwire.
8. Connect electrical leads to their respective terminals. (Refer to tags or to Wiring Diagram Manual.) Torque 3/16 inch stud nut to 20-30 inch-pounds and 3/8 inch stud nut to 120-130 inch-pounds.
9. Slide rubber hood into place over terminals.
10. Install cooling air exhaust duct on starter-generator and position clamp in place.

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(11) Insert duct exhaust opening into engine main drain manifold and secure with screw.

(12) Install clamp at starter-generator end of duct and tighten.

(13) Close engine access door.

(14) Perform an operational check of dc starter-generator.

6. Adjusting DC Generator System Output Voltage

NOTE: A precision 0- to 50-volt dc voltmeter is required to perform this adjustment.

- In the following procedure, the generator output voltage is adjusted to 28.5 volts dc. This voltage is drived to obtain optimum battery life (due to battery recharge rate) at a standard day temperature of 59°F; however, environmental temperatures at which the aircraft is to be operated may be in a higher temperature range which ultimately increases the charge rate (due to decreasing internal battery resistance) and reduces battery life. The following tables list various recommended voltage settings versus temperatures at which the aircraft is to be operated. The charts are applicable to 19 and 20 cell batteries as noted.

Maximum Anticipated Ground Temperature	Gen. Voltage Reg. Setting-With 19 Cell Battery
Up to 30°F	29.0V
31°F to 60°F	28.5V
61°F to 80°F	28.0V
81°F to 100°F	27.5V
Over 100°F	27.0V

Maximum Anticipated Ground Temperature	Gen. Voltage Reg. Setting-With 20 Cell Battery
Up to 30°F	29.0V
31°F to 60°F	29.0V
61°F to 80°F	29.0V
81°F to 100°F	28.5V
Over 100°F	28.0V

The lower generator output voltage settings provide a lower battery charging rate which in turn increases the length of time for the battery to obtain a full charge. If mission requirements (short flights) are often encountered, the standard day setting (28.5 volts) may be required to obtain a fully charged battery.

A. To adjust dc generator system output voltage, proceed as follows:

(1) Remove all electrical power to airplane.

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(2) Gain access to voltage regulators. Regulators are on the fourth tray of the relay rack at fuselage station 150.

(3) Connect the red (positive) lead of the precision dc voltmeter to the red test jack on the LH generator control unit (CN901). Connect the black (negative) lead of the precision dc voltmeter to the black test jack on the LH generator control unit.

(4) Connect external electrical power to airplane.

(5) Start LH engine. (Refer to Chapter 71.)

(6) With the "ELECTRICAL MASTER" and "BATTERY" switches at ON, disconnect external electrical power from airplane.

(7) Position "LH DC GENERATOR" at ON.

(8) Advance engine speed to 75% rpm. Operate engine for 10 to 15 minutes at this speed for temperature stabilization of the generator control unit.

(9) Note the left hand generator loadmeter. Load system to approximately 0.25 indication on loadmeter.

(10) Position "BATTERY" switches at OFF.

(11) Note the precision voltmeter reading. The meter should read 28.5 volts.

(12) If voltmeter does not read 28.5 volts, adjust voltage adjustment (screwdriver slot) on left hand generator control unit until 28.5 volts is indicated on precision voltmeter. (Clockwise rotation of adjustment screw increases voltage.)

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(13) Position "LH DC GENERATOR" at OFF. Retard engine speed to IDLE RPM.

(14) Connect precision voltmeter to right hand generator control unit (CN902) test jacks. Observe correct polarity.

(15) Start right hand engine.

(16) Position "RH DC GENERATOR" switch at ON.

(17) Advance right engine speed to 75% rpm. Operate engine for 10 to 15 minutes at this speed for temperature stabilization of the generator control unit.

(18) Note the right hand generator loadmeter reading. Load system to approximately 0.25 indication on loadmeter.

(19) Check that "BATTERY" switches are at OFF.

(20) Note the precision voltmeter reading. The meter should read 28.5 volts.

(21) If voltmeter does not read 28.5 volts, adjust voltage adjustment (screwdriver slot) on right hand generator control unit until 28.5 volts is indicated on precision voltmeter (clockwise rotation of adjustment screw increases voltage).

(22) Retard engine speed to IDLE RPM.

(23) Perform procedures in step B.

B. To perform parallel adjustment of DC Generator System, proceed as follows:

NOTE: Perform procedures in step A before performing parallel adjustment. Do not operate AUP during these procedures.

(1) Press and hold "PUSH TO TEST RH GEN VOLT" switch. The airplane dc voltmeter should read 28.5 volts.

(2) If airplane dc voltmeter does not read within one percent of precision voltmeter, adjust airplane dc voltmeter instrument adjustment screw on face of instrument until indicator reads ±1 percent of precision voltmeter. Release test switch.

(3) Position "LH DC GENERATOR" at ON.

(4) Press and hold "PUSH TO TEST LH GEN VOLT" test switch. The airplane dc voltmeter should read 28.5 volts.

(5) Repeat step (2).

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NOTE: If the cockpit dc voltmeter adjustment screw is adjusted, the zero indication of the meter may be slightly off when electrical power is removed from the airplane.

(6) Position "BATTERY" switches at ON.

(7) Increase dc bus load, if necessary, until loadmeter indicates approximately 0.3 (30 percent) load.

(8) Position "LH DC GEN" switch at OFF and note loadmeter reading.

(9) Check for 28.5 volts on precision voltmeter.

(10) Position "LH DC GEN" switch at ON and the "RH DC GEN" switch at OFF and note loadmeter reading.

(11) Connect precision voltmeter (preferably a digital voltmeter) to left hand generator control unit. Observe correct polarity of the meter leads.

(12) Check for 28.5 volts on precision voltmeter.

NOTE: The loadmeter should read the same in step (8) and (10).

(13) Position "RH DC GEN" switch at ON.

(14) Increase right and left engine speed to 65 percent rpm.

(15) Increase load on the dc bus system until each loadmeter indicates one-fourth of full load (100 amperes each generator).

(16) If the generators are not sharing the load (as indicated by the loadmeters) within one scale division, the generator control unit paralleling adjustment must be adjusted.

NOTE: When adjusting the parallel potentiometer, adjust the potentiometer for the generator that has the highest loadmeter reading slightly counterclockwise. Adjust the opposite generator parallel potentiometer the same amount in the clockwise direction.

(17) Repeat step (16) until the RH and LH loadmeters are within one scale division of each other.

(18) Recheck system voltage (28.5 VDC) at primary bus. Adjust potentiometers on generator control units if required. If voltage adjustments are performed, recheck generator paralleling subsequent to voltage adjustment.

C. Reverse Current Relay Functional Test

NOTE: Ensure that the voltage and parallel adjustments are correct before performing this check.

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(1) With a .25 (25 percent) load on the dc bus system and the engines operating at idle rpm, position "RH DC GEN" and "LH DC GEN" generator switches at OFF. Check that "BATTERY" switches are positioned at ON.

(2) Position "LH DC GEN" at ON. Slowly increase engine rpm until the right hand loadmeter indicates a load. Increase engine speed to cruise rpm for 10 seconds.

(3) Shut down left engine. (Refer to Chapter 71.)

(4) While left engine is windmilling downward in speed, observe left dc generator loadmeter. The meter should indicate a rapid decrease in load toward zero. Upon reaching zero, the meter should continue reading downward to a minus 0.1 reading, then rapidly return to zero. If loadmeter does not return to zero, position the "LH DC GEN" switch to OFF.

NOTE: The minus 0.1 reading indicates proper operation of the reverse current relay. The loadmeter reading must not exceed 0.1 in the reverse direction. If it does, or if the pointer fails to return to zero immediately after reading in the reverse direction. The reverse current relay is faulty and should be replaced.

(5) Position "RH DC GEN" switch at ON.

(6) Shut down right engine. (Refer to Chapter 71.)

(7) Repeat step (4) as applied to right dc generator loadmeter. Observe same requirements of operation.

(8) Remove electrical power from airplane.

(9) Remove test equipment and secure access doors and/or equipment as necessary.

7. Removing Electrical Power Distribution Relays

A. The removal and installation procedures for the generator control unit, reverse-current relays and the various monitor relays are installed forward of fuselage station 76 or in the electrical equipment rack at fuselage station 150 **(see figure 24-30A-10).** To remove relays from electrical rack assembly, proceed as follows:

(1) Place all switches at OFF and remove external power from airplane.

(2) Remove trim panels as necessary to gain access to relay rack.

(3) Locate applicable relay on rack **(see figure 24-30A-10).**

(4) Remove screws securing shelf to rack and slide shelf inboard.

(5) Disconnect electrical wiring (non plug-in type) from relay terminals.

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(6) Remove mounting screws from relay and remove relay from shelf.

8. Installing Electrical Power Distribution Relays

A. To install electrical relays on electrical rack shelves, proceed as follows:

(1) Ensure that all electrical power is removed from airplane.

(2) Position relay on applicable shelf and secure with attaching screws **(see figure 24-30A-10).**

(3) Connect electrical wiring (non-plug-in type) to relay terminals. (Refer to Wiring Diagram Manual.)

(4) Position shelf on rack and secure with screws.

(5) Replace trim panels.

(6) Perform an operational check-out of system which contains the relay.

8A. Removing and Installing Feeder Protection Relays (Station 150)

A. Remove feeder protection as follows:

(1) Place all switches at OFF or airplane parked position and remove external power from airplane.

(2) Remove trim panels as necessary to gain access to relay rack.

(3) Remove screws that secure the second (from the top) shelf and slide shelf inboard.

(4) Disconnect electrical leads from feeder protection relay. Tag leads (as an aid to proper reinstallation).

(5) Remove nut and washer from "GEN" stud of reverse current relay and lift feeder protection relay from stud.

B. Install feeder protection relay as follows:

NOTE: Feeder protection relays (K1049 and K1050) are sensitive differential current relays that become energized whenever the current flow though either one of the two large studs exceeds the current flow through the other stud by 150 to 200 amperes. Since its operation is based on differential flux magnetic forces, the relay must be installed in a manner that minimizes the influence of magnetic flux from outside sources, (such as a nearby bus bar or a large relay).

(1) Ensure that all electrical power is removed from airplane. Check that BATTERY and other switches are positioned at OFF.

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(2) Position feeder protection relay on GEN stud of reverse current relay on top of bus bar. Secure with washer and nut.

(3) Visually check that lower surface of feeder protection relay is positioned parallel to cover of reverse current relay and that sides of feeder protection relay are parallel to sides of reverse current relay.

(4) If necessary, loosen GEN stud nut and reposition feeder protection relay. Tighten nut and repeat step (3).

(5) Connect cable leads to feeder protection relay studs.

(6) Connect grounded lead wire to "G" terminal and trip signal wire to "T" terminal.

(7) Reposition relay shelf into rack and secure with screws.

(8) Replace trim panels.

(9) Perform an operational check-out of the generator system that contains this feeder protection relay.

9. Replacement of Battery Temperature Sensors

A. To replace battery temperature sensors, proceed as follows:

(1) Remove all electrical power from airplane.

(2) Gain access to battery compartment.

(3) Disconnect battery plugs and vent connections.

(4) Remove battery(s) from airplane.

(5) Remove cover from battery and disconnect sensor wires to receptacle.

(6) Remove cell connector with sensor attached.

B. To replace battery temperature sensor, proceed as follows:

(1) Install cell connector with sensor and connect electrical wiring to receptacle.

(2) Install battery(s) in airplane and connect electrical plugs and vent hoses.

(3) Perform operational check-out of battery temperature system.

(4) Check system for security and applicable safeties.

(5) Install access doors.

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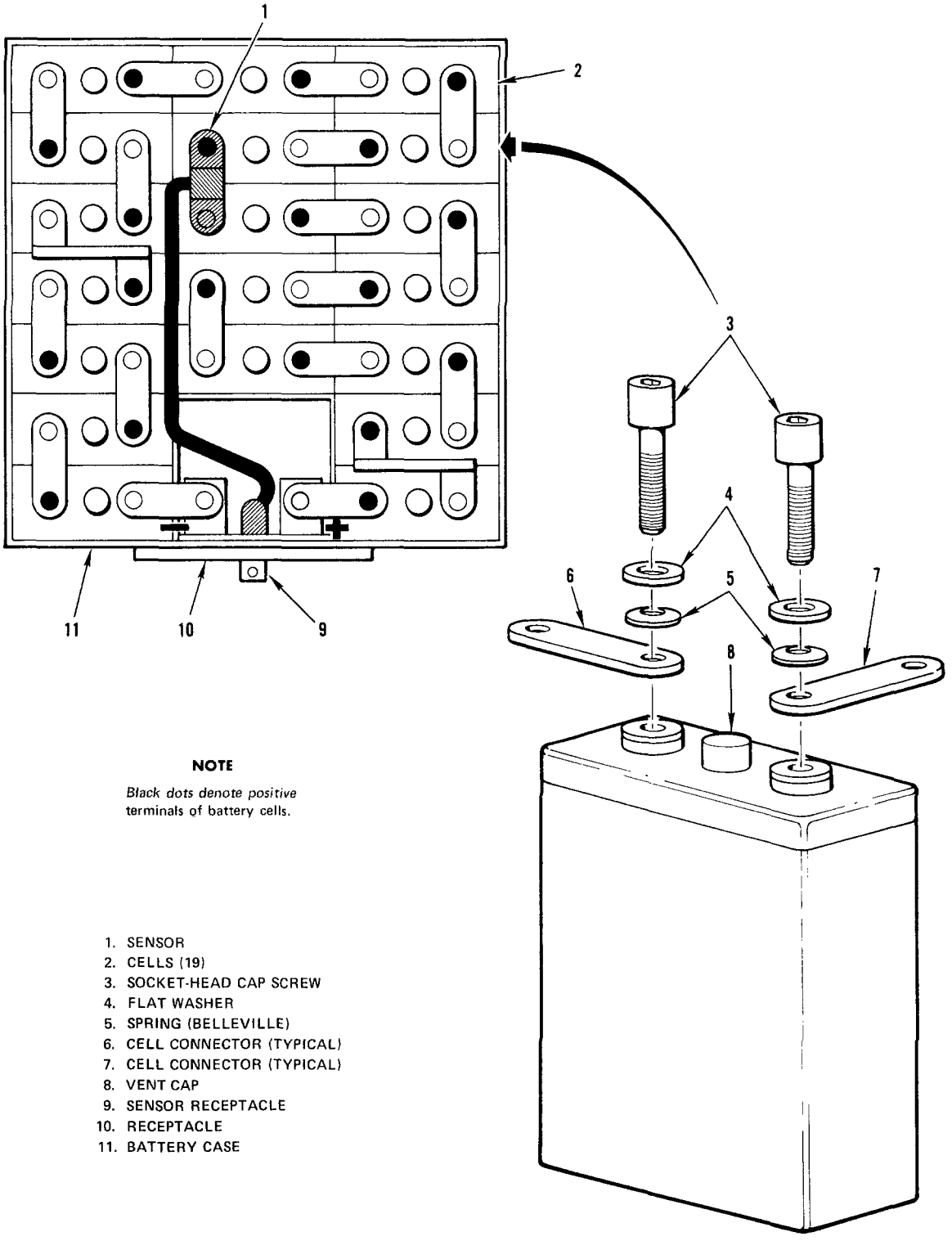

NOTE

Black dots denote positive terminals of battery cells.

1. SENSOR
2. CELLS (19)
3. SOCKET-HEAD CAP SCREW
4. FLAT WASHER
5. SPRING (BELLEVILLE)
6. CELL CONNECTOR (TYPICAL)
7. CELL CONNECTOR (TYPICAL)
8. VENT CAP
9. SENSOR RECEPTACLE
10. RECEPTACLE
11. BATTERY CASE

Battery Exploded View
Figure 24-30A-6

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10. Checking Battery Temperature Sensor Resistance

A. To check battery temperature sensor resistance, proceed as follows:

(1) Gain access to batteries.

(2) Remove electrical connector and battery temperature sensor plug from batteries.

(3) Using a Simpson 260 or equivalent multimeter, connect the negative lead of the multimeter to the negative terminal on the battery.

(4) Connect the positive lead of the multimeter to pin "A" of sensor electrical plug. If any voltage is present, replace sensor.

NOTE: It is mandatory that the battery is at ambient temperature before performing the resistance check; however, if time does not permit this, the battery cell interconnect may be removed and placed away from the heat that is generated by the battery. Enough time shall be allowed so that the battery interconnect (with sensor) be allowed to cool to ambient temperature.

(5) Disconnect multimeter leads and connect leads between pins "B" and "C" of battery temperature sensor receptacle. Check the following resistance measurement that is applicable to the ambient temperature of the battery:

Temperature	Resistance
-20oF	5.5 Meg ohms
0°F	3.5 Meg ohms
+32°F	1.08 Meg ohms
+75°F	300 thousand ohms
+120°F	102 thousand ohms
+150°F	51.1 thousand ohms
+170°F	33.2 thousand ohms
+190°F	22.1 thousand ohms

(6) Connect multimeter leads between pins "A" and "B" of sensor receptacle. Resistance should be 49.9 thousand ohms.

(7) Disconnect multimeter leads from receptacle.

(8) Install interconnect strap (if removed) torque screws to 35-50 inch-pounds.

(9) Install battery electrical plug and temperature sensor electrical plug.

(10) Check batteries for security and proper safeties.

(11) Perform operational check-out of battery temperature system.

(12) Install access doors.

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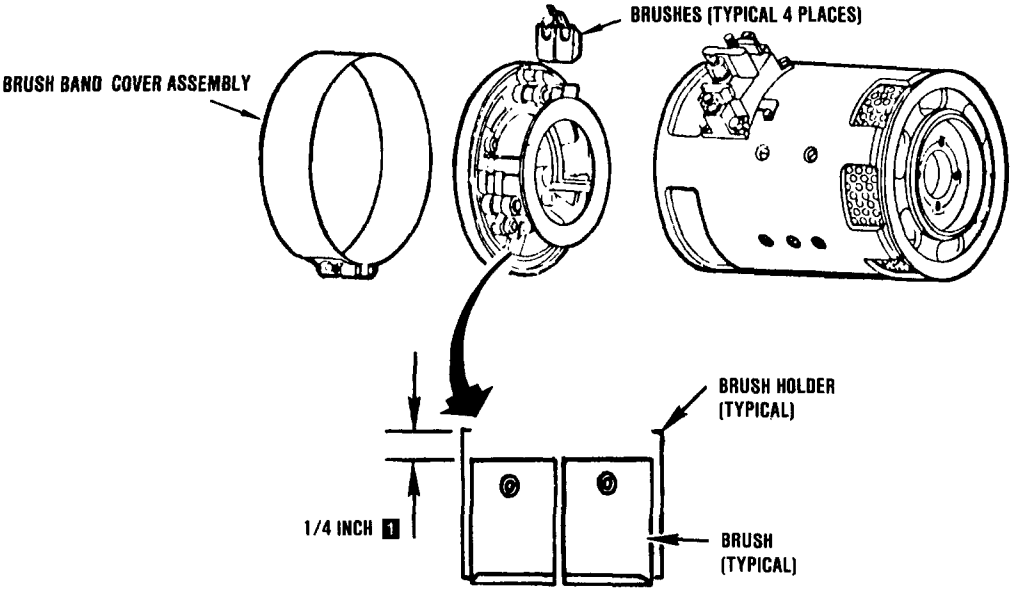

1 REPLACE BRUSHES WHEN TOP OF BRUSH RECEDES TO THIS DIMENSION.

Figure 24-30A-7 — G.E. Starter/Generator Brush Inspection

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11. DC Starter/Generator Brush Inspection

A. To inspect the starter/generator brushes, proceed as follows:

(1) Remove all electrical power from airplane.

(2) Open engine cowling.

(3) Loosen screw securing brush band cover assembly to generator. Lift screw (shank end with band retainer) from brush band.

CAUTION

Do not allow foreign material to enter brush area.

(4) Remove brush band cover assembly from generator.

(5) Inspect brushes (total of eight) as follows:

(a) Check the top of each brush for extension above the brush holders (see Figure 24-30A-7).

(b) If the top of the brush(es) is even (flush) with the top of the brush holder, continued inspections at 150-hour intervals is recommended.

(c) If the top of the brush recedes to 1/4" below the top surface of the brush holder, the brushes must be replaced.

(6) Position brush band cover assembly in place on generator and install band retainer and screw. Tighten and safety band retainer screw.

(7) Check complete installation for security, and close engine cowling.

12. DC Starter/Generator Brush Run-In

A. To perform the DC starter/generator brush run-in, proceed as follows:

EQUIPMENT/TOOLS

DC power source capable of providing up to 30 volts and 75 amperes. Rheostat approximately 5.5 amperes, 15 ohms.

(1) Place the starter/generator in a padded V-block to run in brushes.

(2) Connect the power source positive lead to starter/generator terminal B. Connect power source negative lead to terminal E.

(3) Connect rheostat across terminals A and B.

(4) Remove brush band cover assembly.

WARNING

A sudden high-speed start may cause serious personal injury. To control the start, make sure the rheostat is adjusted to the zero resistance point prior to motoring the unit.

(5) Run in brushes as follows: Apply power to starter/generator and set rheostat for a speed of 6000 rpm until brushes have a fit of 100 percent of the arc and at least 75 percent of the contact area.

(6) Blow loose carbon dust from starter/generator; install brush band cover assembly.

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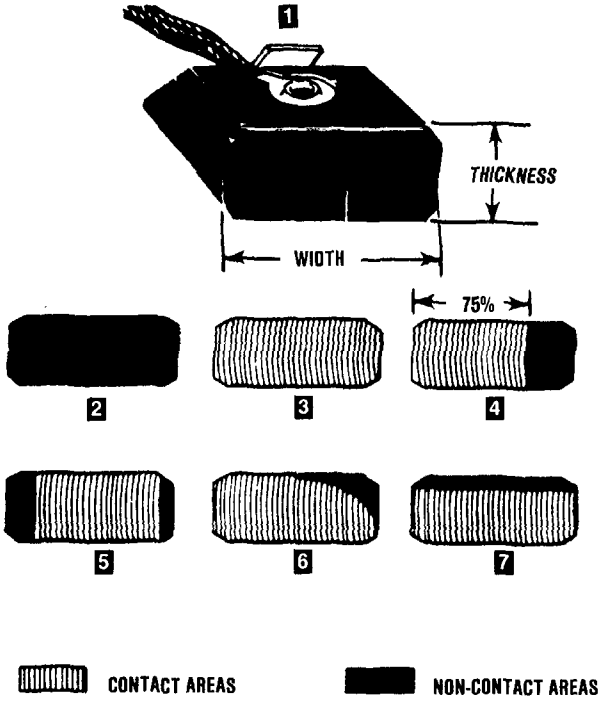

1 - BRUSH THICKNESS AND WIDTH

2 - NEW AND UNUSED BRUSH

3 - *BRUSH SHOWING 100 PERCENT CONTACT ACROSS* THICKNESS AND 100 PERCENT ACROSS WIDTH

4 5 - BRUSH SHOWING CONTACT 100 PERCENT ACROSS THICKNESS AND 75 PERCENT ACROSS WIDTH

6 - BARELY SATISFACTORY

7 - *INCORRECT BRUSH FIT*

Figure 24-30A-8 — Correct and Incorrect Brush Fit

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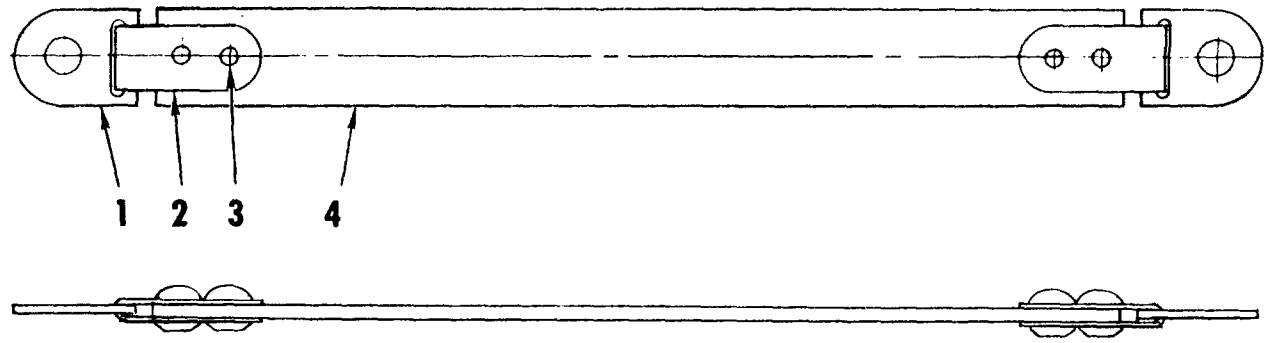
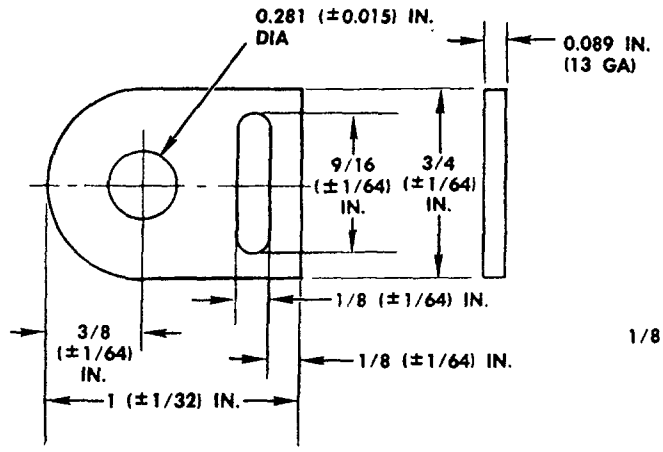
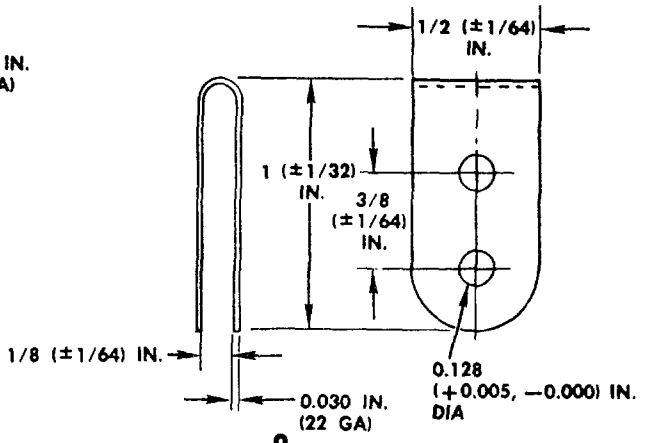




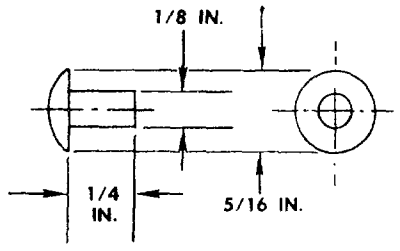
DETAIL 1

MATERIAL	CRS
FINISH	NICKEL PLATE
NO. REQD	2



DETAIL 2

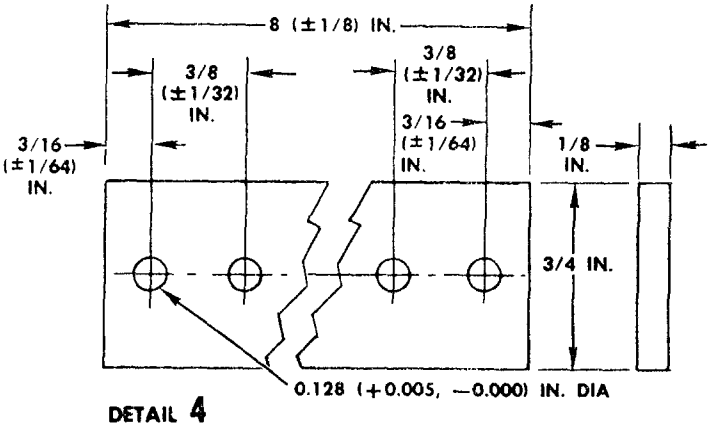
MATERIAL	CRS
FINISH	NICKEL PLATE
NO. REQD	2



DETAIL 3

AMER STD TRUSS HEAD RIVET

MATERIAL	STEEL
FINISH	NICKEL PLATE
NO. REQD	4



DETAIL 4

MATERIAL	WEB BELT—1000 LB TEST
FINISH	OLIVE DRAB
NO. REQD	1

T39D-2-54-79

Battery Cell Puller
Figure 24-30A-9

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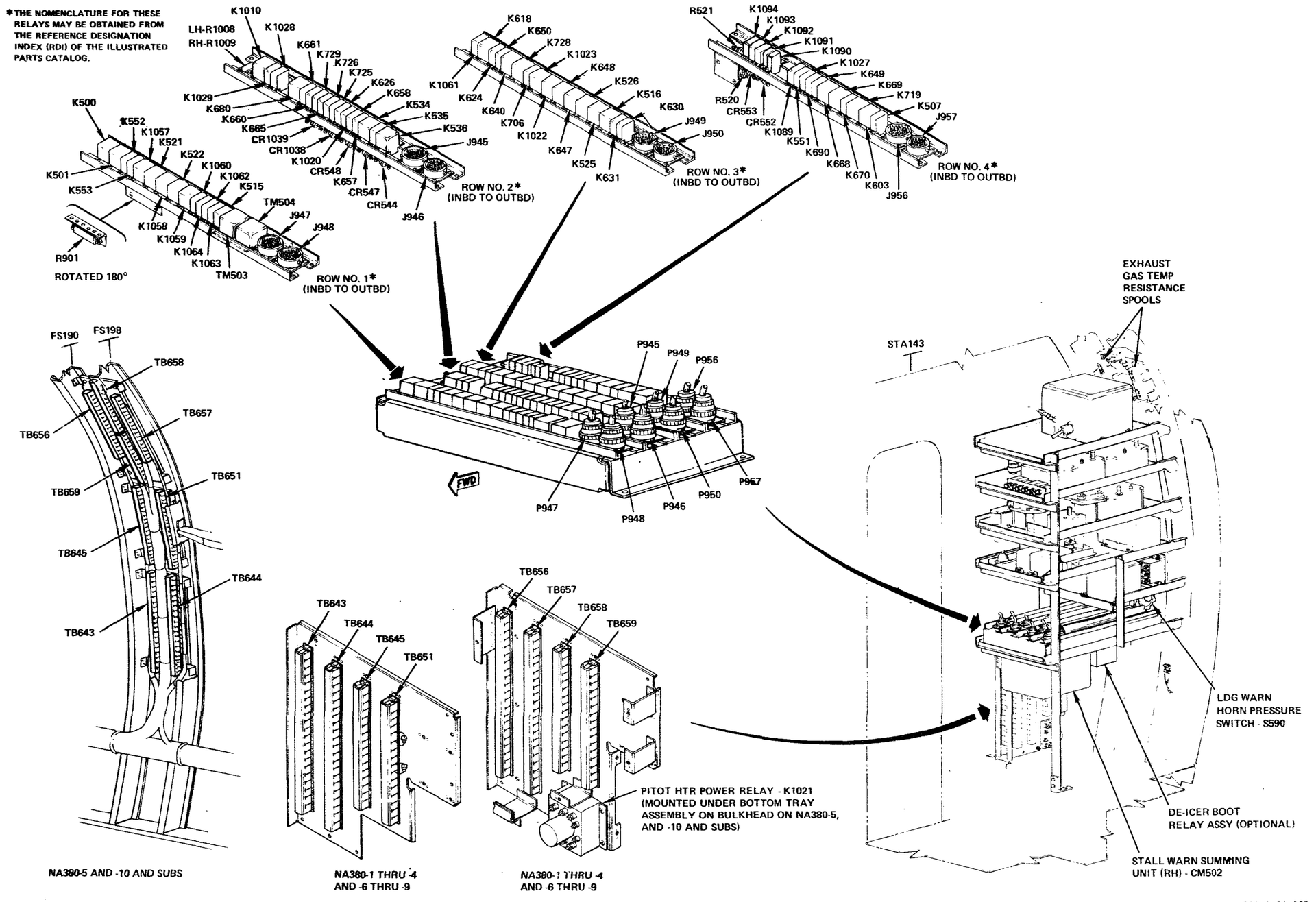

Electrical Equipment Rack
Figure 24-30A-10 (Sheet 1 of 2)

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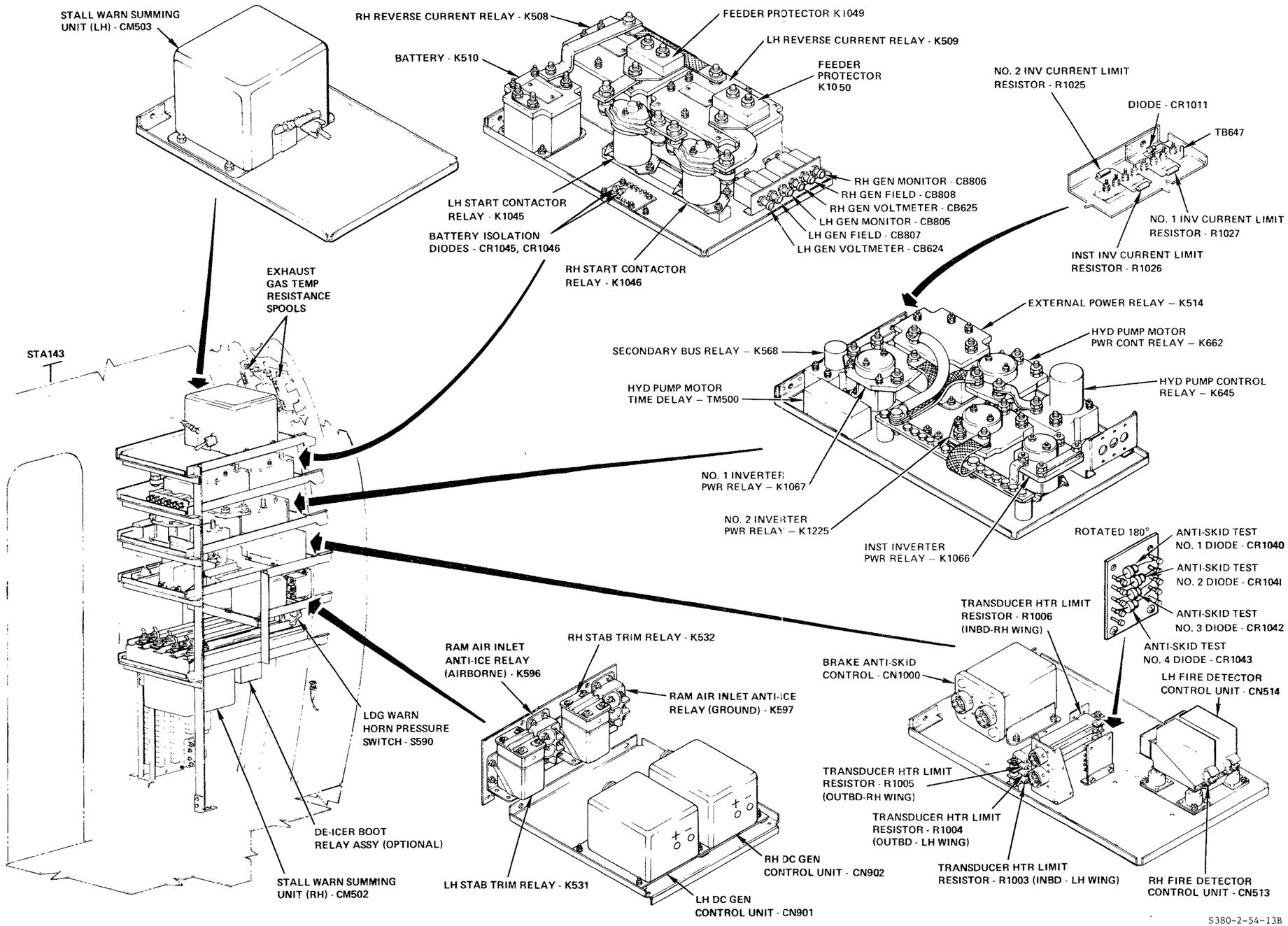
STALL WARN SUMMING UNIT (LH) - CM503
RH REVERSE CURRENT RELAY - K508
FEEDER PROTECTOR K1049
LH REVERSE CURRENT RELAY - K509
BATTERY - K510
FEEDER PROTECTOR K1050
NO. 2 INV CURRENT LIMIT RESISTOR - R1025
DIODE - CR1011
TB647
RH GEN MONITOR - CB806
RH GEN FIELD - CB808
RH GEN VOLTMETER - CB625
LH GEN MONITOR - CB805
LH GEN FIELD - CB807
LH GEN VOLTMETER - CB624
LH START CONTACTOR RELAY - K1045
BATTERY ISOLATION DIODES - CR1045, CR1046
RH START CONTACTOR RELAY - K1046
NO. 1 INV CURRENT LIMIT RESISTOR - R1027
INST INV CURRENT LIMIT RESISTOR - R1026
EXHAUST GAS TEMP RESISTANCE SPOOLS
STA143
EXTERNAL POWER RELAY – K514
HYD PUMP MOTOR PWR CONT RELAY – K662
SECONDARY BUS RELAY – K568
HYD PUMP CONTROL RELAY – K645
HYD PUMP MOTOR TIME DELAY – TM500
NO. 1 INVERTER PWR RELAY – K1067
NO. 2 INVERTER PWR RELAY – K1225
INST INVERTER PWR RELAY – K1066
ROTATED 180°
ANTI-SKID TEST NO. 1 DIODE - CR1040
ANTI-SKID TEST NO. 2 DIODE - CR1041
ANTI-SKID TEST NO. 3 DIODE - CR1042
ANTI-SKID TEST NO. 4 DIODE - CR1043
TRANSDUCER HTR LIMIT RESISTOR - R1006 (INBD-RH WING)
RH STAB TRIM RELAY - K532
RAM AIR INLET ANTI-ICE RELAY (AIRBORNE) - K596
RAM AIR INLET ANTI-ICE RELAY (GROUND) - K597
BRAKE ANTI-SKID CONTROL - CN1000
LH FIRE DETECTOR CONTROL UNIT - CN514
LDG WARN HORN PRESSURE SWITCH - S590
TRANSDUCER HTR LIMIT RESISTOR - R1005 (OUTBD-RH WING)
TRANSDUCER HTR LIMIT RESISTOR - R1004 (OUTBD - LH WING)
DE-ICER BOOT RELAY ASSY (OPTIONAL)
RH DC GEN CONTROL UNIT - CN902
TRANSDUCER HTR LIMIT RESISTOR - R1003 (INBD - LH WING)
STALL WARN SUMMING UNIT (RH) - CM502
LH STAB TRIM RELAY - K531
LH DC GEN CONTROL UNIT - CN901
RH FIRE DETECTOR CONTROL UNIT - CN513
S380-2-54-13B

Electrical Equipment Rack
Figure 24-30A-10 (Sheet 2 of 2)

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13 Removing Starter-Generator Inner Shaft

To remove the starter-generator inner shaft from the starter-generator, proceed as follows:

NOTE: The inner shaft cannot be procured as a separate item. It must be ordered as part of a supply kit P/N 3A228274G3 from General Electric; however, the friction cone and flat washer may be procured as separate items. See figure 24-30A-11 for part numbers.

(1) Remove the starter generator from the engine. Refer to "Removing DC Starter-Generator.

(2) Bend back tab on lockwasher securing locknut in place.

(3) Place locally fabricated spanner wrench (see figure 24-30A-11) over locknut and a deep socket and drive handle over the inner shaft coupling spines.

(4) While holding the inner shaft with the deep socket and drive handle, turn the spanner wrench and remove the locknut. If the inner shaft has been sheared, jar the locknut loose by hitting the spanner wrench handle with a mallet.

(5) Remove spring, flat washer, and lockwasher from the inner shaft.

(6) Remove inner shaft from starter-generator

(7) Remove friction cone from inner shaft. Using a 10 power magnifier, inspect the inner shaft and the friction cone as follows:

(a) Inspect inner shaft for cracks, corrosion, dents and abnormal wear.

(b) Inspect friction cone for cracks and corrosion. The small end of the friction cone shall not be worn to a knife edge or cracked or broken.

(8) Install friction cone on inner shaft. Check that 0.034 minimum/0.057 maximum dimension shown on Figure 24-30A-11 is obtained. If dimension is not obtained, exchange either inner shaft or friction cone for one that does provide the correct dimension.

NOTE: If inner shaft is being replaced perform the following step.

(9) Remove retainer ring from inner shaft, then remove coupling with its "O" rings and spacer.

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14 Installing Starter-Generator Inner Shaft

To install starter-generator inner shaft, refer to Figure 24-30-7 and proceed as follows:

(1) Clean all parts with naphtha, methyl ethyl ketone (MEK), xylene, or equivalent and air dry.

NOTE: If coupling has been removed perform steps 2 through 5.

(2) Check rotational play of coupling on end of inner shaft, less "O" rings and spacer. Rotational play shall not exceed 20 degrees.

(3) Apply a liberal amount of grease (MIL-G-21164C) to the drive splines of the inner shaft. Apply Plastilube No. 3 non-melting grease to the coupling splines of the inner shaft.

(4) Install new "O" ring packing in flanged end of coupling. Slide coupling, flanged end first on to drive end of inner shaft.

(5) Install new "O" rings (2) inside of coupling. Install spacer with beveled side of spacer next to "O" ring.

(6) Install retainer ring on end of inner shaft. Make certain that ring is locked in shaft groove.

(7) Install inner shaft into starter-generator.

(8) Install lockwasher, flatwasher, spring, and locknut on inner shaft. Torque locknut to 76 to 126 inch pounds, by turning spanner wrench on locknut, while holding the deep socket on the coupling splines.

(9) Bend tab of lockwasher into a notch on the locknut.

(10) Reinstall starter-generator on engine. Refer to Installing DC Starter-Generator.

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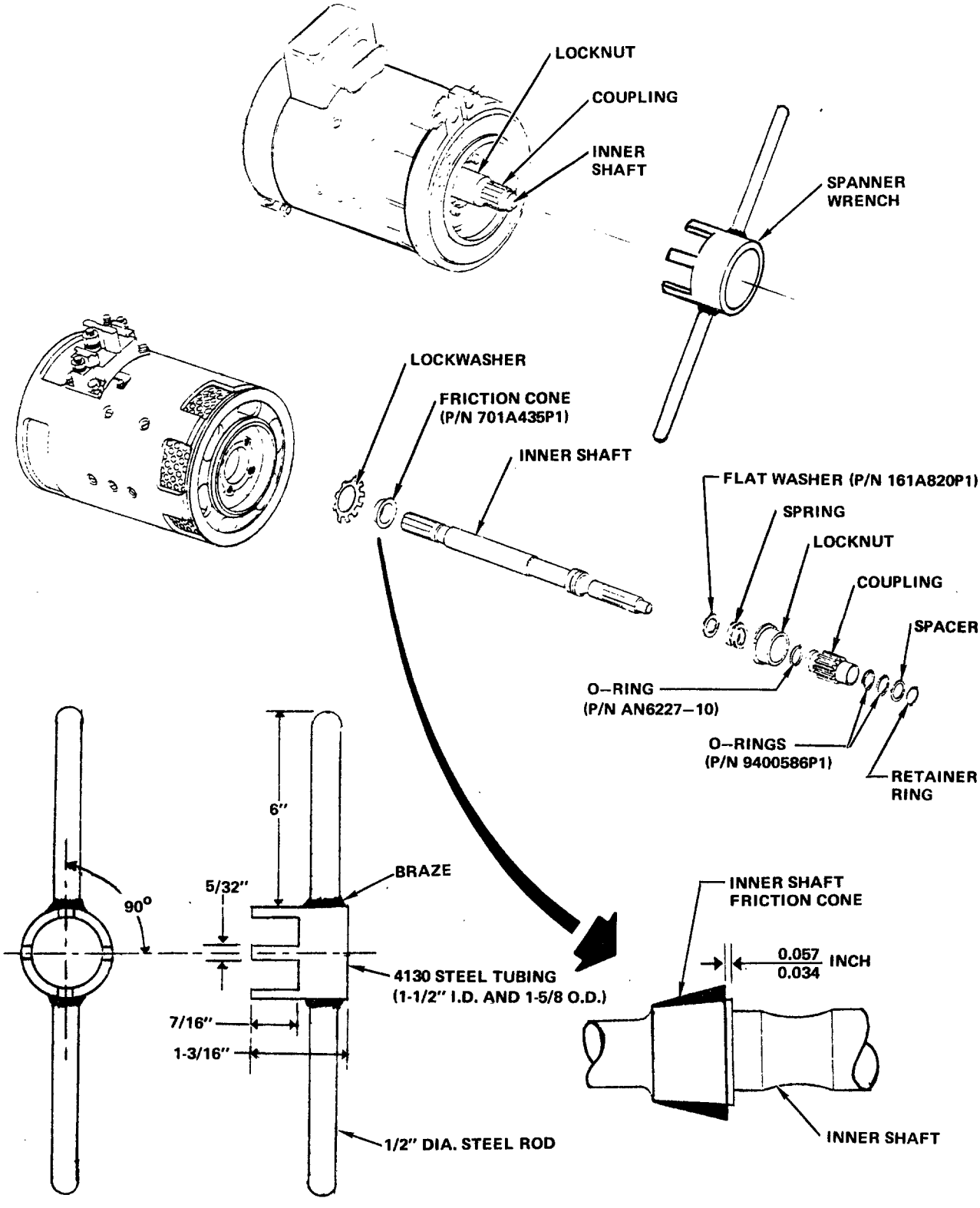

SPANNER WRENCH FABRICATION

Removing Starter - Generator Inner Shaft
Figure 24-30A-11

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WARNING: WHEN GROUND MAINTENANCE PROCEDURES REQUIRE THE REMOVAL OF PROTECTIVE COVERINGS (SUCH AS TAPE, RUBBER COVERS, ETC.), THESE COVERINGS MUST BE REINSTALLED BEFORE ANY ELECTRICAL POWER IS CONNECTED TO THE AIRPLANE. FAILURE TO OBSERVE THIS WARNING COULD RESULT IN A FIRE AND POSSIBLE LOSS OF THE AIRPLANE, CAUSED BY AN ACCIDENTAL SHORT CIRCUIT OF THE UNPROTECTED ELECTRICAL CONDUCTOR.

CAUTION: BEFORE REMOVING TERMINAL BLOCK COVERS TO MAKE IT EASIER TO TEST OR ADJUST THE ELECTRICAL SYSTEM, DISCONNECT ALL EXTERNAL POWER FROM THE AIRPLANE AND PLACE ALL AIRPLANE SWITCHES AT OFF OR AIRPLANE-PARKED POSITION.

NOTE: A precision 0- to 50-volt dc voltmeter, a constant-voltage type dc external power source of 28 volts dc, and a constant-current type dc external power source of 1000 amperes capacity are required to check and test the dc power electrical system.

- The dc generators and generator control units should be at operating temperature before the generator systems are adjusted.

1. Testing Battery System

NOTE: The specific gravity readings of the individual battery cells do not indicate the state-of-charge of a nickel-cadmium battery. The specific gravity remains the same, whether the battery is charged or discharged. The state of charge is determined by the amount of charging current. The batteries are considered charged when the charging current is less than 15 amperes for both batteries.

A. To test battery system, proceed as follows:

(1) Place all airplane switches at OFF or airplane-parked position.

(2) Connect a 28.5-volt, constant-voltage type dc external power source to airplane utility external power receptacle. Turn on external power source.

(3) Place "ELECTRIC MASTER" switch at ON.

(4) Make certain "BATTERY" switches are at OFF.

(5) Note dc external power source loadmeter reading.

(6) Place "BATTERY" switches at ON.

(7) Note dc external power source loadmeter reading.

(8) Compute the difference between loadmeter readings (step (7) minus step (5)).

NOTE: If the difference (step (8)) exceeds 15 amperes, the batteries are not operating at full capacity. They should remain on charge until the difference in loadmeter readings is less than 15 amperes.

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(9) Repeat steps (5) through (8) until difference in loadmeter readings is less than 15 amperes.

(10) Place "BATTERY" switches at OFF.

(11) Place "ELECTRIC MASTER" switch at OFF.

(12) Disconnect dc external power source from airplane.

2. Testing DC Generator System

A. To test dc generator system, proceed as follows:

(1) Start engines. (Refer to chapter 71.)

(2) Place "LH DC GEN" switch at OFF.

(3) Momentarily depress "PUSH RH GEN VOLT" button and note dc voltmeter. The meter should read 28.5 volts dc, the output voltage of the right dc generator system.

(4) Note left dc generator loadmeter. The meter should read 0.00 (0.0 percent).

(5) Note right dc generator loadmeter. The meter should read 0.30 (30 percent).

(6) Place "LH DC GEN" to ON.

(7) Place "RH DC GEN" switch at OFF.

(8) Note left dc generator loadmeter. The meter should read 0.30 (30 percent).

(9) Note right dc generator loadmeter. The meter should read 0.00 (0.0 percent).

(10) Momentarily depress "PUSH LH GEN VOLT" button and note dc voltmeter. The meter should read 28.5 volts dc, the output voltage of the left dc generator system.

(11) Place "RH DC GEN" switch to ON.

(12) Note both dc generator loadmeters. Each meter should read about 0.15 (15 percent).

NOTE: If loadmeters do not read the same (within one-half scale division), the electrical load is not equally shared between the dc generator systems. To equalize load between the dc generator systems, perform paralleling adjustments of generator control units.

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(13) While observing dc loadmeters, place "INVERTER" switch at No. 1, at OFF, and again at No. 1. Both loadmeter readings should rise, fall, then rise together, indicating that both dc generators remain operational and share the dc load.

NOTE: If the sudden increase in dc load causes one of the dc generators to trip off, a faulty or misadjusted generator system is indicated. Investigate and correct as required.

(14) Place "RH DC GEN" switch at OFF.

(15) Shut down left engine. (Refer to chapter 71.)

(16) While left engine is windmilling downward in speed, observe left dc generator loadmeter. The meter reading should indicate a rapid decrease in load (from 0.30) toward zero. Upon reaching zero, the meter should continue reading downward to a minus 0.1 reading, then rapidly return to zero.

NOTE: The minus 0.1 reading indicates proper operation of the reverse-current relay. The loadmeter reading must not exceed 0.1 in the reverse direction. If it does, or if the pointer fails to return to zero immediately after reading in the reverse direction, the reverse-current relay is faulty and should be replaced.

(17) Place "RH DC GEN" switch at ON.

(18) Note right dc generator loadmeter. The meter should read about 0.30 (30 percent), or full load.

(19) Shut down right engine.

(20) Repeat step (18) as applied to right dc generator loadmeter. Observe same requirements of operation.

3. Operational Check-out of Battery Temperature System

A. To perform an operational check-out of battery temperature system, proceed as follows:

(1) Start engines. (Refer to Chapter 71).

(2) Engage BATT TEMP IND circuit breaker.

(3) Depress BATT TEMP TEST switch and monitor battery temperature indicator. When indicator pointer reaches 120°F the amber caution should illuminate. Essential bus voltage should indicate approximately 26.5 volts dc.

(4) When battery temperature reaches 150°C, the BATT HOT warning light should be on and the batteries should be disconnected from the battery bus.

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NOTE: To ensure that the batteries are disconnected from the battery bus, a voltmeter should be connected across the battery relay (K510).

(5) Position "BATTERY" switches at EMERG. Battery should be connected to battery bus.

(6) Position "BATTERY" switches to ON and release "BATT TEMP TEST" switch.

(7) Check that BATT HOT warning light goes out when battery temperature pointer is below 150°F. Batteries should be connected to battery bus. Bus voltage should be approximately 26.5 volts dc.

(8) When indicator pointer is below 120°F, check that bus voltage is 28.5 volts dc or the voltage at which the system was adjusted.

(9) Shut down engine. (Refer to Chapter 71).

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INSPECTION CHECK

INSPECT — BATTERY GROUND CABLES/BOLTS (S/N 380-1 and Subs)

CONSUMABLES

- TT-L-32 Sealing Lacquer
- Aluminum Oxide Paper, or similar non-ferrous material
- Copper Shear Safetywire

1. Battery switch(es) ..... OFF
2. Electrical power ..... OFF
3. Remove access doors from each side of battery compartment.
4. Disconnect battery cable quick disconnects and battery temperature sensor plugs.
5. Temporarily insulate and stow cable quick disconnects.
6. Disconnect vent hoses from batteries and from sump jar.
7. Remove sump jar and loose vent hoses from airplane. Protect all openings.
8. Remove battery tie-down straps and rods.

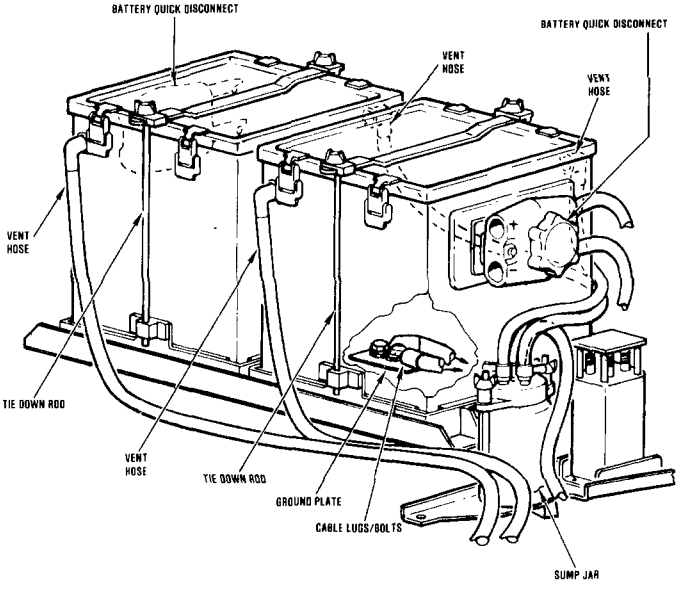
WARNING

Use care to avoid injury when lifting batteries. Each battery weighs 84 pounds.

CAUTION

To prevent an aft CG condition, an equivalent amount of ballast should be installed at each battery location after removal of battery. A tailstand may be used in lieu of ballast in battery area.

9. Remove batteries from airplane.
10. Remove landing light panel assembly.
11. Remove battery cable lugs from grounding plate.
12. Inspect battery cable lugs, bolts, washers, and grounding plate for evidence of arcing, pitting, or corrosion.
13. Clean affected parts with aluminum oxide paper to remove all evidence of arcing, pitting, or corrosion.
14. Install battery cable lugs to ground plate using 2 ea. AN6-5A bolts, AN961-616 washers, and MS21042L6 nuts.
15. Seal all exposed metal surfaces, cable lugs, and hardware with sealing lacquer to prevent corrosion.
16. Remove ballast weight in battery tie-down area (if used) and install batteries.
17. Connect battery tie-down straps and rods.
18. Install battery sump jar and hoses.
19. Remove any insulation material on battery quick disconnects and secure to battery. Safetywire quick disconnects with copper shear wire.
20. Connect battery temperature sensor plugs.
21. Install landing light assembly panel.
22. Install access doors on each side of battery compartment.
23. Perform operational test of electrical system using battery power.
24. Airplane ..... secure as desired



NEW

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EXTERNAL POWER

DESCRIPTION AND OPERATION

EXTERNAL POWER SYSTEM

The external power system consists of an engine start external power receptacle, a utility external power receptacle, an external power cutout relay, and associated relays and control circuitry (see Figures 24-40-1 and 24-40-2). Both the utility and engine start external power receptacles are on the left side of the fuselage, aft of the wing trailing edge. [1] The utility external power receptacle permits connection of an external DC power source to supply all DC-operated airplane systems, except the engine starting systems. The engine start external power receptacle provides for the connection of external DC power for use by the engine starting systems. On S/N 370-1 and Subs, the external power cutout relay is inside the electrical power distribution box at FS 333, and operates to connect the external power source to the paralleling bus. On S/N 380-1 and Subs, the external power cutout relay is on the electrical equipment rack at FS 150 and operates to connect the external power source to the essential bus. The relay also disconnects the external power source from the airplane paralleling (or essential) bus in case of external power source failure while the external power source remains plugged into the utility external power receptacle.

NOTE

S/N 370-1 and Subs and 380-1 and Subs each have the following DC electrical busses; battery bus, essential bus, secondary bus, and starter bus. In addition, S/N 370-2 thru -6, -8, and -9 have a paralleling bus.

The shell of the engine start external power receptacle is rectangular-shaped, to accept the rectangular plug of a constant-current type DC external power source. An arm, with a roller wheel attached, protrudes through a rectangular opening on the side of the receptacle shell. When external power is plugged into the receptacle, the body of the plug pushes against the roller and actuates the arm. The arm operates a microswitch, connecting DC power to the coil of the external start power monitor relay (see Figures 24-40-1 and 24-40-2). The receptacle has one small-diameter pin and two large-diameter pins. The large pins connect DC external power to the starter bus and the airplane structure (ground). The small pin connects to the starter-controller relays.

The shell of the utility external power receptacle is oval-shaped, to accept the oval plug of a constant-voltage type DC external power source. The receptacle has one small-diameter pin and two large-diameter pins. The large pins connect DC external power to the DC power distribution system paralleling (or essential) bus through contacts on the external power cutout relay and the airplane structure (ground). The small pin connects positive-polarity DC external power to the SW terminal on the external power cutout relay, through normally closed contacts on the starter power monitor relays.

On both external power receptacles, the small pin is shorter than the large pins, to permit the large pins to engage first when DC external power is plugged into the receptacle, and to allow the large pins to disengage last when DC external power is disconnected. This prevents the possibility of a fire caused by arcing at the large pins of the external power receptacle when flammable vapors or fumes are present in the area. The small pin of the utility external power receptacle energizes the external power cutout relay after the large pins are engaged, and de-energizes the relay before the large pins disengage.

The external power cutout relay consists of three relays inside one housing, as does a reverse-current relay. The voltage relay senses the polarity and voltage level of the external power source. The differential relay has three coils: one for sensing the differential voltage, one for sensing reverse current, and a biasing coil. The contactor relay connects the external power source to the paralleling (or essential) bus through the reverse-current coil.

When a DC external power source is plugged into the utility external power receptacle, the voltage relay closes, provided the polarity of the external power source is correct. When energized, the voltage relay connects external power to the paralleling (or essential) bus through the differential voltage coil. If no other source of power is connected to the paralleling bus, the external power cutout relay differential voltage coil picks up at about 20 VDC from the external power source. If airplane battery power or generator power is on the paralleling bus, the differential voltage coil energizes the differential relay when the external power source is 0.35 to 0.65 volt above the bus voltage. When energized, the differential relay contacts close and connect external power to the contactor relay coil. This energizes the contactor relay, closing it and connecting the external power source to the paralleling (or essential) bus through the reverse-current coil on the differential relay. The contactor relay also connects external power to the coil of the external power monitor relay coil. The external power cutout relay protects the airplane battery system or generator systems from possible overloading if the DC external power source fails or is turned off while the external source remains plugged into the utility external power receptacle. If this condition occurs, reverse-current flow through the external power cutout relay automatically forces the differential voltage relay contacts open when the reverse current exceeds 18 to 35 amperes. This de-energizes the contactor relay and disconnects the external power source from the airplane paralleling (or essential) bus.

[1] S/N 370-1 and Subs; on S/N 380-1 and Subs the external power receptacle is on the right side of the airplane at FS 150.

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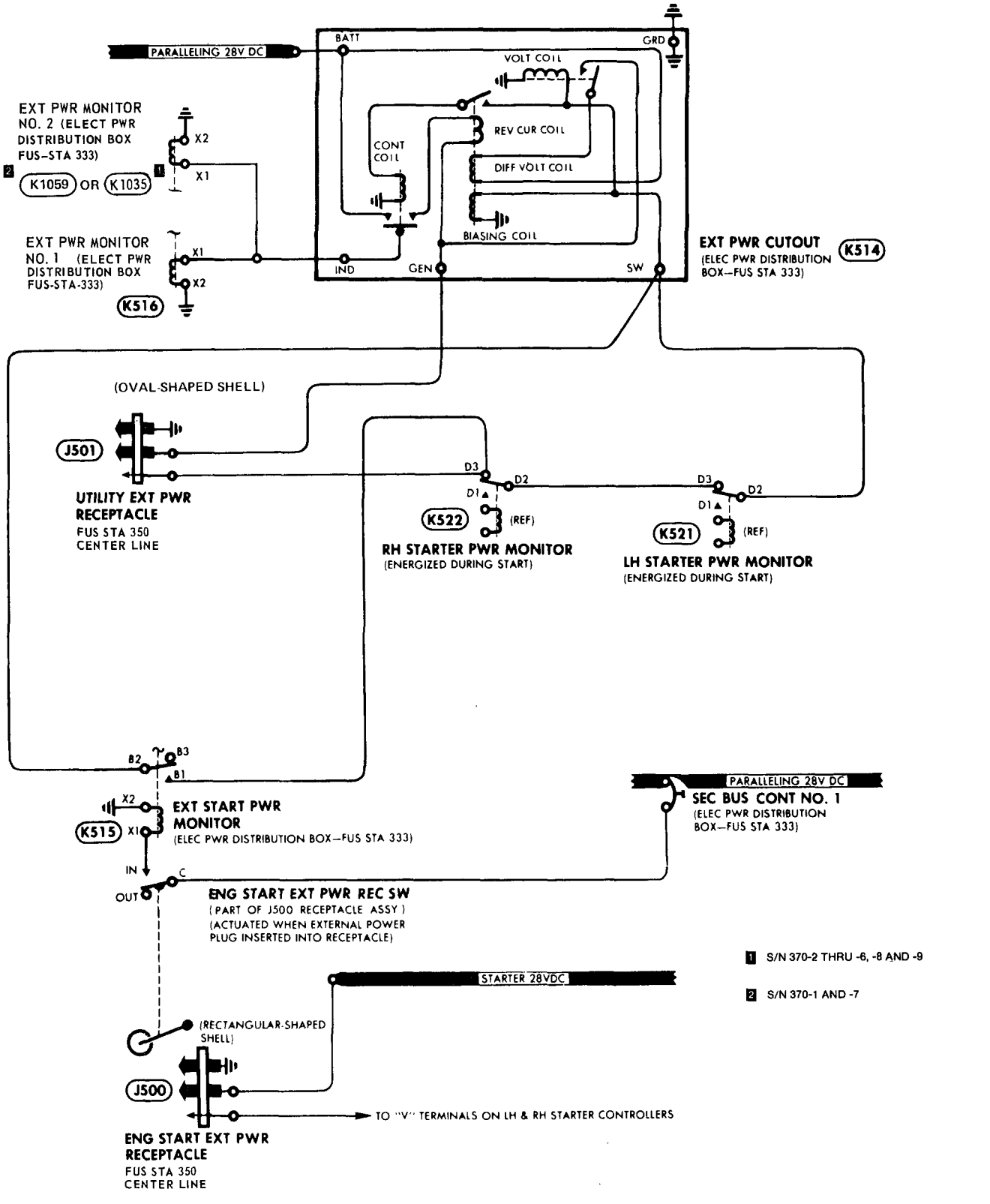

Figure 24-40-1 — External Power System Schematic — S/N 370-1 and Subs

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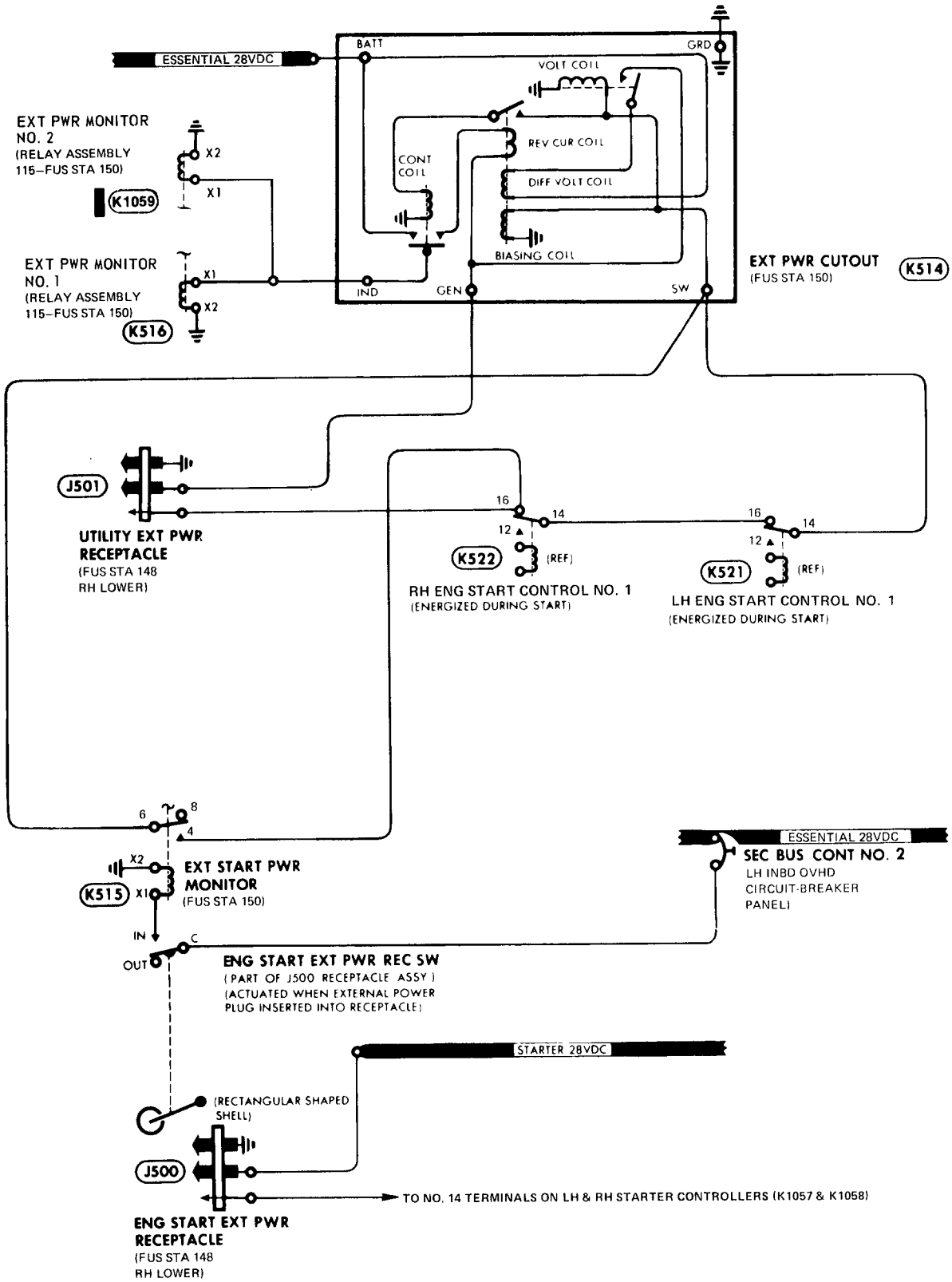

S380-2-54-4B

Figure 24-40-2 — External Power System Schematic — S/N 380-1 and Subs

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ELECTRICAL LOAD DISTRIBUTION SYSTEM

DESCRIPTION AND OPERATION

AC LOAD DISTRIBUTION SYSTEM

The AC power from the No. 1 and No. 2 inverter is distributed from the following electrical busses: the 115-volt AC primary (instrument), the 115-volt AC secondary, the 115-volt C-phase (SN 380-1 and Subs), and (through transformers) two 26-volt AC busses and four 5-volt indirect lighting busses. When the instrument inverter is operating (failure of the No. 1 and No. 2 inverters or during battery engine start), the power output is supplied only to the 115-volt primary bus and (through a transformer) one 26 VAC bus (see Figure 24-50-1). The remainder of the AC system is available to the windshield anti-ice system.

DC LOAD DISTRIBUTION SYSTEM — S/N 370-1 AND SUBS

The DC load distribution system (Figure 24-50-2) consists of five busses, and associated bus tie-in and control relays. The distribution system distributes generator system power, battery power, and DC external power to the busses through various bus tie-in combinations, according to the existing operating conditions.

During normal flight operation, all busses except the starter bus connect to the paralleling bus and are energized by the generator systems, through bus tie-in relays. If one generator system fails (or is turned off), the secondary busses remain connected to the essential bus. If both generator systems fail (or are turned off), all busses are automatically disconnected from the paralleling bus, and only the essential bus remains connected to the battery bus. This is to provide battery power on the essential bus, for the operation of equipment necessary for flight.

Circuit breakers rated at 140 amperes are installed in the circuit breaker box at station 333 of airplanes incorporating S/B 11. These larger breakers permit the breakers located in the cockpit to disengage first when an overload condition occurs. This permits the crew to quickly restore normal operation when the overload condition is corrected.

During ground maintenance (engines not running), when a DC external power source is plugged into the utility external power receptacle, all busses except the starter bus and battery bus are energized. If the ELEC MASTER and BATTERY switches are placed at ON, the external power source will also power the battery bus and charge the batteries.

During normal ground operation (engines running), when only one DC generator system is operating and no external power is connected to the airplane, the secondary bus is de-energized. The starter bus is energized by external power when a constant-current type DC external power source is plugged into the engine start external power receptacle. The starter bus can be energized by airplane battery system power, to make a self-sufficient engine start. During an emergency start, the batteries are connected in series to the starter bus, through contacts on the battery start relay.

When the airplane is parked and a DC external power source is plugged into the utility external power receptacle, the external power cutout relay (Figure 24-40-1) is automatically energized. The energized cutout relay connects external power to the paralleling bus and the coil of the external power monitor relay (see Figure 24-50-2). External power on the paralleling bus automatically energizes the essential bus relay, connecting the essential bus to external power on the paralleling bus. The energized external power monitor relay connects the paralleling bus to the coil of the secondary bus relay. This relay becomes energized and connects the secondary bus to the essential bus. Placing the ELEC MASTER and BATTERY switches at ON completes the ground connection for the battery relay, energizing the relay and connecting the battery bus to the paralleling bus. This connects the battery system in parallel with the DC external power source. All busses (except the starter bus) are then energized by external power on the paralleling bus.

When a constant-current type DC external power source is plugged into the engine start external power receptacle, the starter bus is energized by the external power source. The insertion of the external power plug into the engine start external power receptacle also mechanically actuates a microswitch which connects the external start power monitor relay coil (Figure 24-40-1) to the paralleling bus. The relay then becomes energized and its closed contacts place a short circuit across the contacts of the left and right starter power monitor relays, keeping the external power cutout relay energized during an engine start. This ensures DC external power (from the utility external power source) on the paralleling bus, for engine ignition during a normal engine start. After a normal engine start has been made, and generator system power is connected to the paralleling bus, the constant-current type external power source should be disconnected from the airplane receptacle. This de-energizes the starter bus and the external start power monitor relay. With the ELEC MASTER and BATTERY switches at ON, DC generaotr system power, battery system power, and DC external power (connected to the utility external power receptacle) are all connected in parallel on the paralleling bus. When the DC external power source is disconnected

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DESCRIPTION AND OPERATION (Continued)

DC LOAD DISTRIBUTION SYSTEM — S/N 370-1 AND SUBS (Cont'd)

from the utility external power receptacle, the external power cutout relay becomes de-energized, in turn de-energizing the external power monitor relay. The busses remain connected to the paralleling bus, because the left and right generator monitor relays are energized. If one DC generator system is turned off (or fails) while the airplane is on the ground, the secondary bus is automatically disconnected from the essential bus. If both DC generator systems are turned off (or fail), the secondary bus is disconnected from the essential bus and the essential and battery busses remain connected to the paralleling bus until the ELEC MASTER or BATTERY switch is placed at OFF.

If one DC generator is turned off (or fails) during flight, the remaining generator system will continue to supply the essential, battery, and secondary busses. When both generator systems are turned off (or fail), all busses are automatically disconnected from the paralleling bus. The essential bus relay, the battery relay and secondary bus relay are automatically de-energized by the generator monitor relays. The de-energized bus tie-in relays disconnect all busses from the paralleling bus. Under these conditions, the essential bus emergency relay coil ground connection is completed through the ELEC MASTER and BATTERY switches (at ON), and through contacts on the left and right generator monitor relays. This automatically connects battery system power to the essential bus only, for the operation of DC equipment necessary for flight.

When a self-sufficient engine start is made, the starter bus is energized by battery system power. During the starting sequence, the battery start relay (Figure 24-30A-2) is energized. When energized, the relay connects No. 2 battery in series with No. 1 battery to the starter bus. This connects 48 VDC to the starter bus during the starting sequence. When the engine reaches about 35% rpm, the starting system automatically drops out and the battery start relay is de-energized. This automatically disconnects No. 2 battery from the starter bus and reconnects it in parallel with No. 1 battery to the battery bus.

ESSENTIAL BUS CIRCUIT BREAKER OUT CAUTION LIGHT AND TEST SYSTEM — (S/N 370-1 AND SUBS CHANGED BY S/B 75-20)

The essential bus circuit breaker out caution light system provides a visual indication of an open 28 VDC essential bus circuit breaker located on the electrical power distribution box (FS 333). The test portion of the system provides a means to check the continuity of the circuit breaker out caution light system. The system consists of a switch, a switch actuator assembly, electrical solenoid, caution light, and a pushbutton test switch (Figure 24-50-4).

With 28 VDC electrical power applied to the airplane and in the event that a circuit breaker (located on the electrical power distribution box) should open, the plunger of that circuit breaker raises the switch actuator (flapper door) which in turn actuates switch S1032. With this switch actuated, 28 VDC electrical power is connected (from the essential bus) to caution light ESS BUS C/B OUT located on the caution warning indicator panel in the cockpit. The caution light will remain illuminated as long as the circuit breaker is open. The master caution light system is energized when the ESS BUS C/B OUT caution light comes on. Resetting the master caution light does not extinguish the ESS BUS C/B OUT caution light.

The essential bus circuit breaker out caution light system may be checked by actuating the ESS BUS C/B OUT TEST switch located on the electrical control panel. When 28 VDC electrical power is applied to the airplane and the ESS BUS C/B OUT TEST switch is depressed and held, a grounding circuit is provided to energize and actuate solenoid L100. The solenoid simulates an open circuit breaker by pushing the switch actuator (flapper door) to cause actuation of switch S1032 and illumination of ESS BUS C/B OUT caution light. Releasing ESS BUS C/B OUT TEST switch opens the grounding circuit allowing the solenoid to de-energize. With the solenoid de-energized, the spring loaded switch actuator (flapper door) causes the solenoid to retract, and causes switch S1032 to open breaking the circuit that illuminates the ESS BUS C/B OUT caution light.

DC LOAD DISTRIBUTION SYSTEM — S/N 380-1 AND SUBS

The DC load distribution system (Figure 24-50-3) consists of three busses, associated bus tie-in and control relays. The distribution system distributes generator system power, battery power, and DC external power to the busses through various bus tie-in combinations, according to the existing operating conditions.

During normal flight operation, the secondary and essential bus are energized by the generator system, through bus tie-in relays. If one generator system fails (or is turned off), the secondary busses remain connected to the essential bus. If both generator systems fail (or are turned off), the secondary bus is automatically disconnected from the essential bus. This is to provide battery power on the essential bus, for the operation of equipment necessary for flight.

During ground maintenance (engines not running), when a DC external power source is plugged into the utility external power receptacle, the essential and secondary busses are energized. If the ELEC MASTER and BATTERY switches are placed at ON, the external power source will also provide power to the batteries for a charging source. On S/N 380-1 and Subs changed by S/L 64 the circuit has been changed to allow the auxiliary power unit (APU) to energize the secondary bus control relay (K658). This allows the APU to power the non-essential electrical items connected to the secondary bus; i.e., portions of the lighting system, galley equipment, lavatory, etc.

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DESCRIPTION AND OPERATION (Continued)

DC LOAD DISTRIBUTION SYSTEM — S/N 380-1 AND SUBS (Cont'd)

During normal ground operation (engines running), when only one DC generator system is operating and no external power is connectd to the airplane, the secondary bus is de-energized. The starter bus is energized by external power when a constant-current type DC external power source is plugged into the engine start external power receptacle. The starter bus can be energized by airplane battery system power, to make a self-sufficient engine start. During a battery start, the batteries are connected in series to the starer bus, through contacts on the battery start relay.

When the airplane is parked and a DC external power source is plugged into the utility external power receptacle, the external power cutout relay (K514) is energized. With relay K514 energized, external power is connected to the essential bus and to the external power monitor relay (K516 and K1059).

Relay K516 is then energized routing power to the secondary bus control relay (K568). With relay K568 energized, the essential bus is connected to the secondary bus. Placing the No. 1 and/or No. 2 BATTERY switches at ON completes the ground circuit to the battery relay (K510) and one or both of the isolation relays (K512 and K1041) energizing the relays and connecting one and/or both batteries to the essential bus. Both busses are then energized by external power.

When a constant-current type DC external power source is plugged into the engine start external power receptacle, the starter bus is energized by the external power source. The insertion of the external power plug into the engine start external power receptacle also mechanically actuates a microswitch which connects the external start power monitor relay coil (Figure 24-40-2) to the essential bus. The relay then becomes energized and its closed contacts provide a holding which keeps the external power cutout relay energized during an engine start. This ensures DC external power (from the utility external power source) on the essential bus, for engine ignition during a normal engine start. After a normal engine start has been made, and generator system power is connected to the essential bus, the constant-current type external power source should be disconnected from the airplane receptacle. This de-energizes the starter bus and the external start power monitor relays. With the ELEC MASTER and BATTERY switches at ON, DC generator system power, battery system power, and DC external power (connected to the utility external power receptacle) are connected in parallel with the essential bus. When the DC external power source is disconnected from the utility external power receptacle, the external power cutout relay becomes de-energized, in turn de-energizing the external power monitor relays. The secondary bus remains connected to the essential bus, because the left and right generator monitor relays are energized. If one DC generator system is turned off (or fails) while the airplane is on the ground, the secondary bus is automatically disconnected from the essential bus. Battery power is then available to the essential bus. Battery power will remain on the essential bus until the ELEC MASTER or BATTERY switches are placed at OFF. If one DC generator is turned off (or fails) during flight, the operating generator will supply power to the essential and secondary busses and electrical power will also be available for charging the batteries. When both generator systems are turned off (or fail), the secondary bus is automatically disconnected from the essential bus. The secondary bus relay (K568) is de-energized by the generator monitor relays.

When a self-sufficient engine start is made, the starter bus is energized by battery system power. During the starting sequence, the battery start relay (K511) is energized. When energized, the relay connects No. 2 battery in series with No. 1 battery to the starter bus. This connects 48 VDC to the starter bus during the starting sequence. When the engine reaches about 42% rpm, the starting system automatically drops out and the battery start relay is de-energized. This automatically disconnects No. 2 battery from the starter bus and reconnects it in parallel with No. 1 battery to the battery bus.

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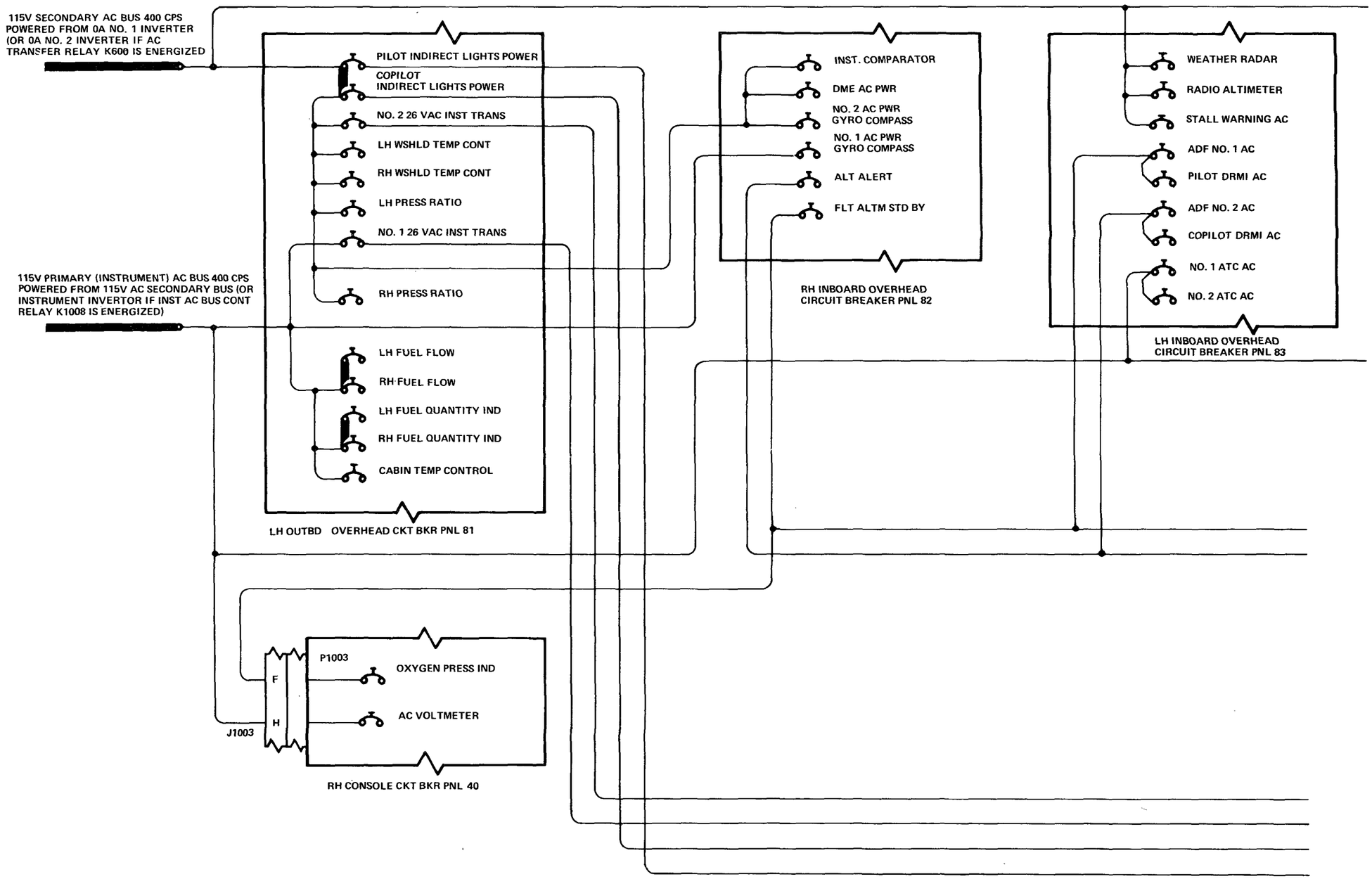
115V SECONDARY AC BUS 400 CPS POWERED FROM 0A NO. 1 INVERTER (OR 0A NO. 2 INVERTER IF AC TRANSFER RELAY K600 IS ENERGIZED
115V PRIMARY (INSTRUMENT) AC BUS 400 CPS POWERED FROM 115V AC SECONDARY BUS (OR INSTRUMENT INVERTOR IF INST AC BUS CONT RELAY K1008 IS ENERGIZED)
PILOT INDIRECT LIGHTS POWER
COPILOT INDIRECT LIGHTS POWER
NO. 2 26 VAC INST TRANS
LH WSHLD TEMP CONT
RH WSHLD TEMP CONT
LH PRESS RATIO
NO. 1 26 VAC INST TRANS
RH PRESS RATIO
LH FUEL FLOW
RH FUEL FLOW
LH FUEL QUANTITY IND
RH FUEL QUANTITY IND
CABIN TEMP CONTROL
LH OUTBD OVERHEAD CKT BKR PNL 81
INST. COMPARATOR
DME AC PWR
NO. 2 AC PWR GYRO COMPASS
NO. 1 AC PWR GYRO COMPASS
ALT ALERT
FLT ALTM STD BY
RH INBOARD OVERHEAD CIRCUIT BREAKER PNL 82
WEATHER RADAR
RADIO ALTIMETER
STALL WARNING AC
ADF NO. 1 AC
PILOT DRMI AC
ADF NO. 2 AC
COPILOT DRMI AC
NO. 1 ATC AC
NO. 2 ATC AC
LH INBOARD OVERHEAD CIRCUIT BREAKER PNL 83
P1003
F
H
J1003
OXYGEN PRESS IND
AC VOLTMETER
RH CONSOLE CKT BKR PNL 40

Figure 24-50-1 — AC Electrical Load Distribution System Schematic (Sheet 1 of 2)

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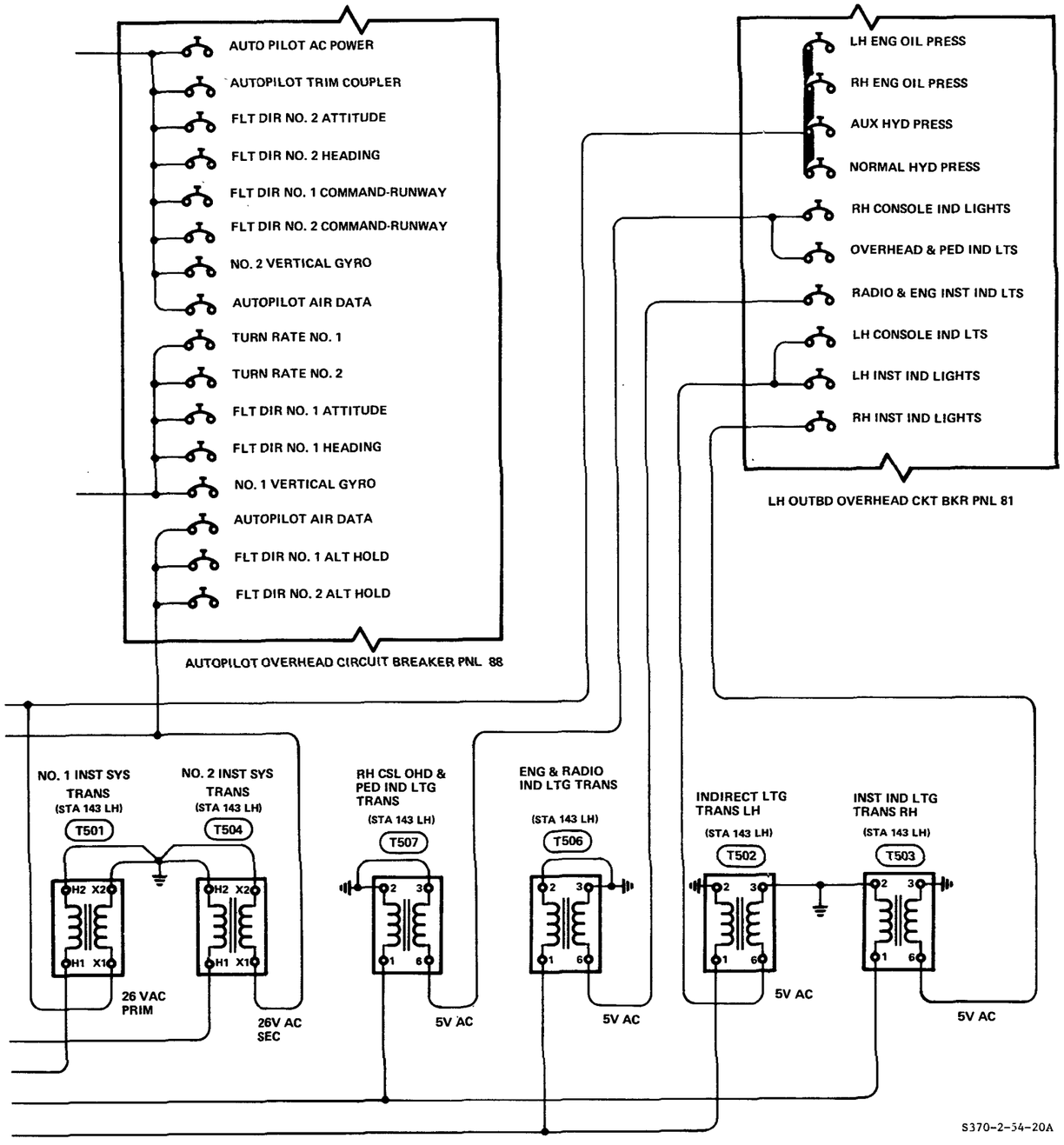
AUTO PILOT AC POWER
AUTOPILOT TRIM COUPLER
FLT DIR NO. 2 ATTITUDE
FLT DIR NO. 2 HEADING
FLT DIR NO. 1 COMMAND-RUNWAY
FLT DIR NO. 2 COMMAND-RUNWAY
NO. 2 VERTICAL GYRO
AUTOPILOT AIR DATA
TURN RATE NO. 1
TURN RATE NO. 2
FLT DIR NO. 1 ATTITUDE
FLT DIR NO. 1 HEADING
NO. 1 VERTICAL GYRO
AUTOPILOT AIR DATA
FLT DIR NO. 1 ALT HOLD
FLT DIR NO. 2 ALT HOLD
AUTOPILOT OVERHEAD CIRCUIT BREAKER PNL 88
LH ENG OIL PRESS
RH ENG OIL PRESS
AUX HYD PRESS
NORMAL HYD PRESS
RH CONSOLE IND LIGHTS
OVERHEAD & PED IND LTS
RADIO & ENG INST IND LTS
LH CONSOLE IND LTS
LH INST IND LIGHTS
RH INST IND LIGHTS
LH OUTBD OVERHEAD CKT BKR PNL 81
NO. 1 INST SYS TRANS (STA 143 LH)
T501
NO. 2 INST SYS TRANS (STA 143 LH)
T504
H2 X2
H1 X1
26 VAC PRIM
26V AC SEC
RH CSL OHD & PED IND LTG TRANS (STA 143 LH)
T507
2 3
1 6
5V AC
ENG & RADIO IND LTG TRANS (STA 143 LH)
T506
5V AC
INDIRECT LTG TRANS LH (STA 143 LH)
T502
5V AC
INST IND LTG TRANS RH (STA 143 LH)
T503
5V AC
S370-2-54-20A

Figure 24-50-1 — AC Electrical Load Distribution System Schematic (Sheet 2 of 2)

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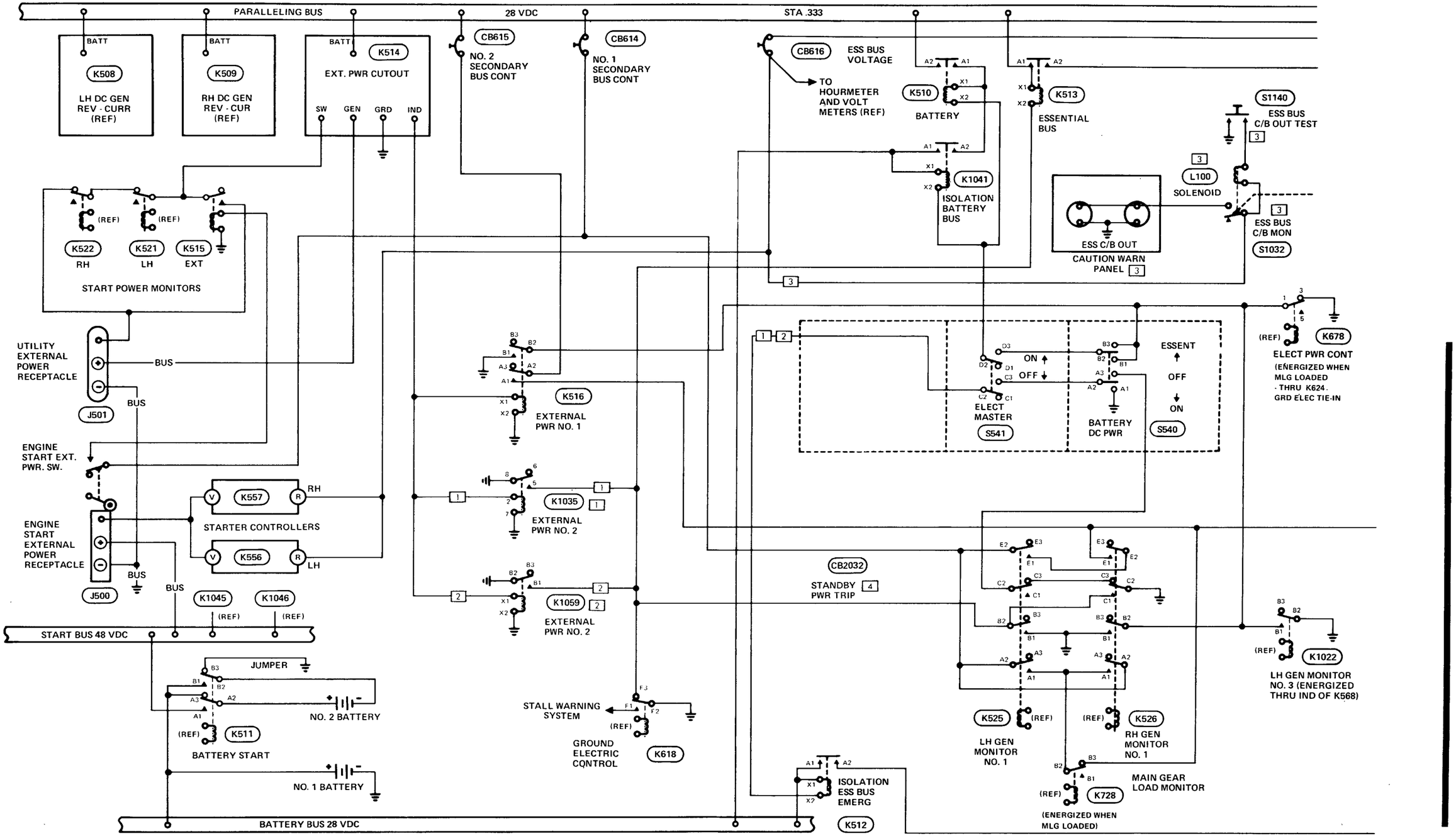

Figure 24-50-2 — DC Electrical Load Distribution System Schematic, S/N 370-1 and Subs (Sheet 1 of 2)

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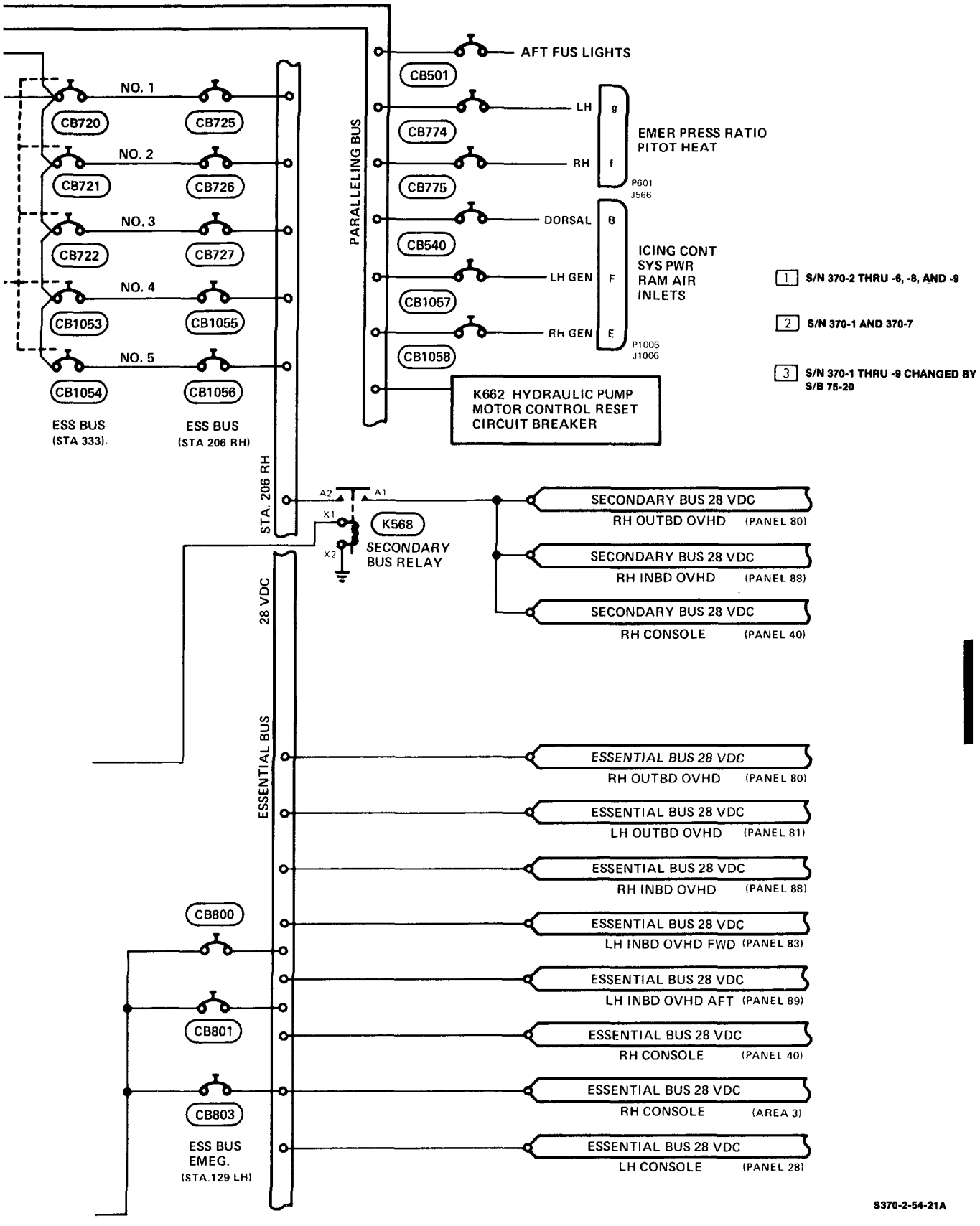

Figure 24-50-2 — DC Electrical Load Distribution System Schematic, S/N 370-1 and Subs (Sheet 2 of 2)

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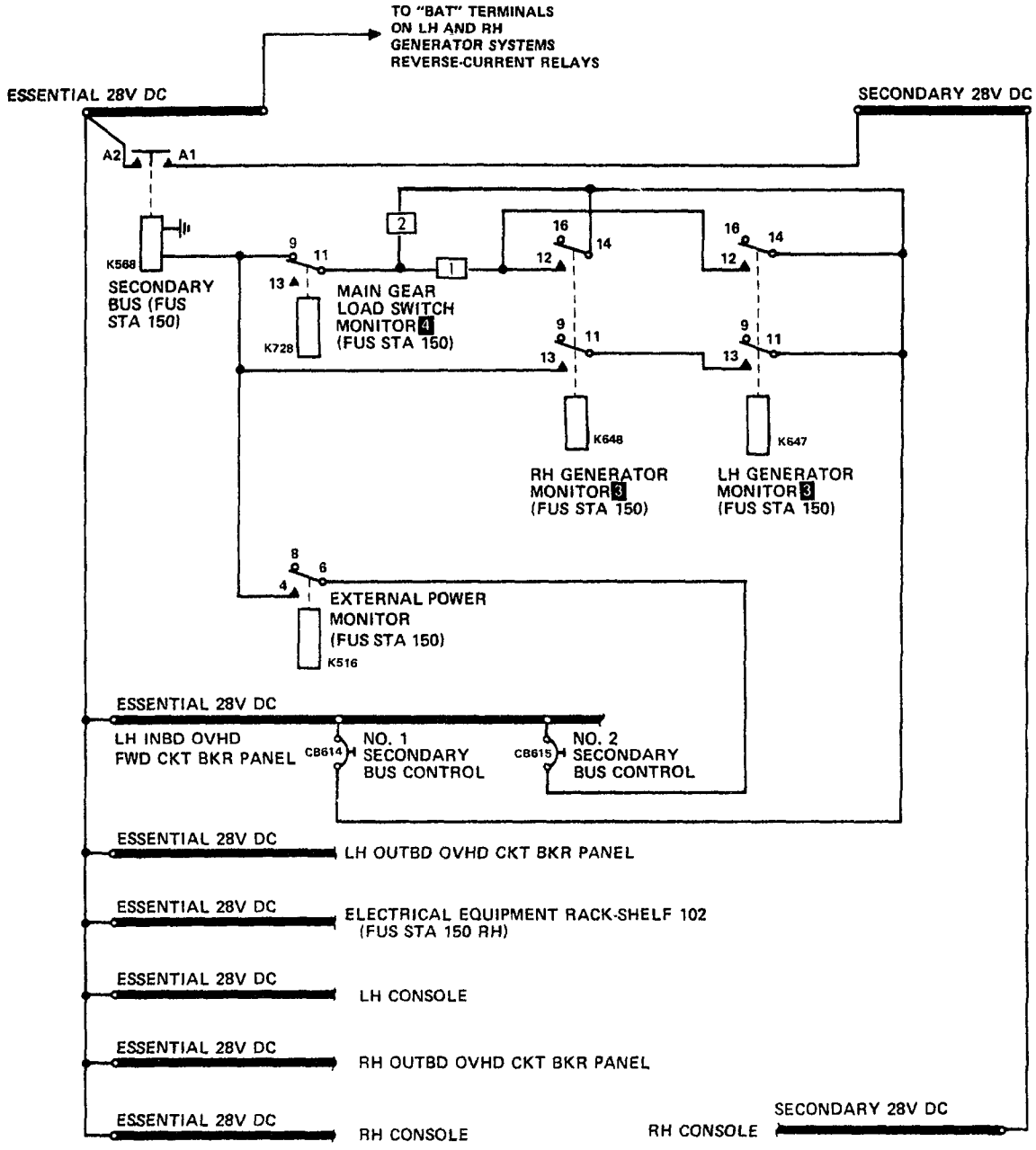

[1] NA-380 AIRPLANES NOT CHANGED BY SERVICE LETTER 64

[2] NA-380 AIRPLANES CHANGED BY SERVICE LETTER 64

[3] ENERGIZED WHEN GENERATOR IS OPERATING

[4] ENERGIZED WHEN MAIN LANDING GEAR SWITCHES ARE IN LOADED POSITION

NOTE

Starter bus is shown on External Power and Battery System Schematic.

380-2-54-5A

Figure 24-50-3 — DC Electrical Load Distribution System Schematic, S/N 380-1 and Subs

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ELECTRICAL LOAD DISTRIBUTION - MAINTENANCE PRACTICES

1. Check-out of Essential Bus Circuit Breaker Out Caution Light System - NA-370 Airplanes Changed by Service Bulletin 75-20

A. Checking Caution Light System Circuitry

(1) Place all airplane switches at OFF or airplane-parked position.

(2) Connect a 28.5-volt, constant-voltage type dc external power source to airplane utility external power receptacle and turn on external power source.

(3) Depress and hold ESS BUS C/B OUT TEST switch located on co-pilot's inboard instrument panel.

(4) Check that ESS BUS C/B OUT caution light is illuminated. (The master caution light is also illuminated.)

(5) Release test switch. Check that ESS BUS C/B OUT caution light goes out. Reset master caution light system.

(6) Disconnect external power source from airplane.

B. Checking Switch Actuator Mechanism

(1) Place all airplane switches at OFF or airplane-parked position.

(2) Connect a 28.5-volt constant-voltage type dc external power source to airplane utility external power receptacle and turn on external power source.

(3) Remove trim panels or couch (as required) to gain access to electrical power distribution box located in aft end of passenger cabin (station 333).

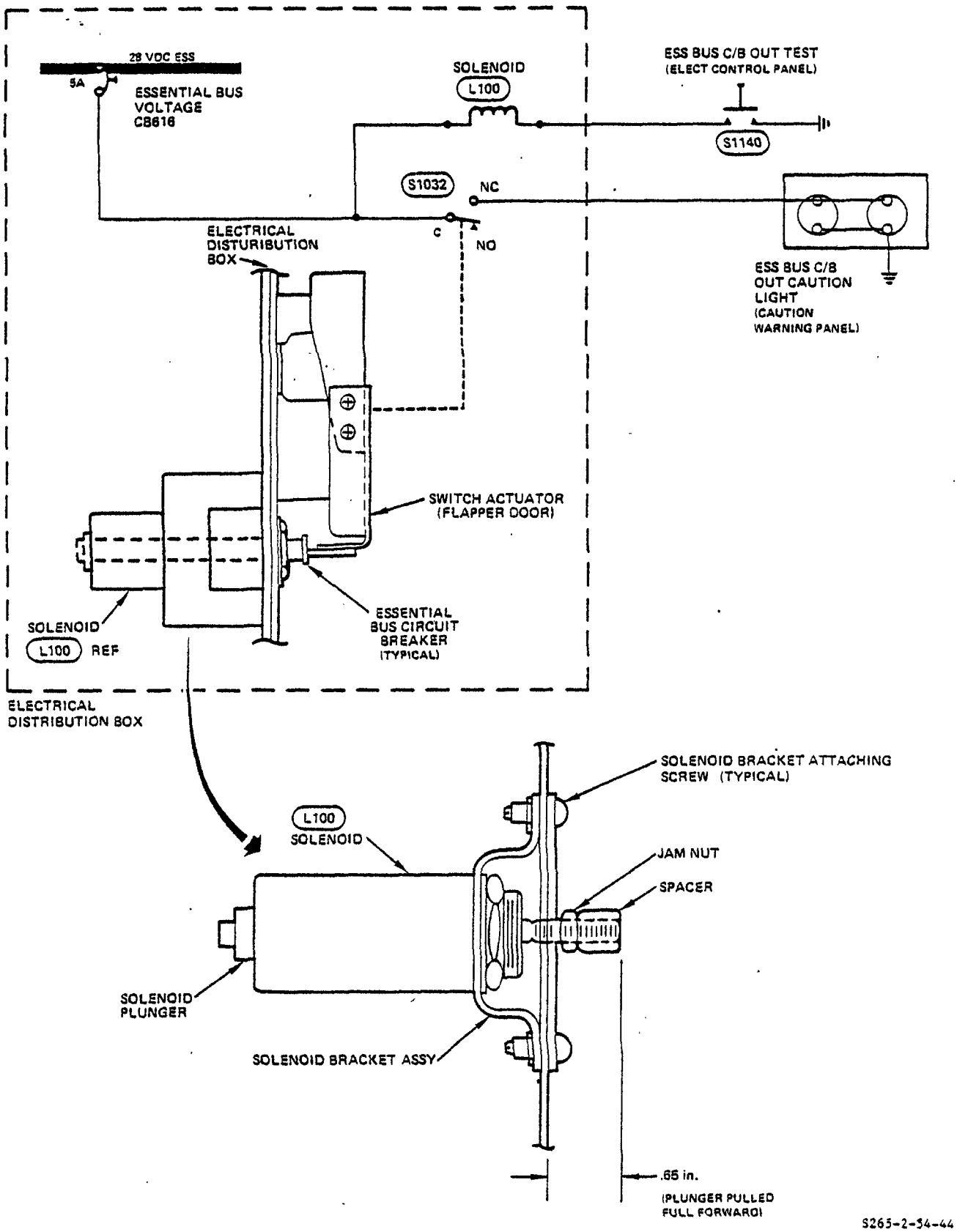
(4) Visually check gap between each of the five essential bus circuit breaker buttons and the circuit breaker out switch actuator assembly.

(5) Manually pull out the circuit breaker button that has the largest gap as determined in step (4).

(6) Check that ESS BUS C/B OUT light on cockpit caution light panel is illuminated. (Master caution light should also be illuminated.)

(7) Depress circuit breaker button (selected in step 5) slightly (3/64-inch minimum) and check that ESS BUS C/B OUT caution light remains illuminated. (Actuator door must contact button.)

(8) Reset circuit breaker and lower actuator door. Check that ESS BUS C/B OUT caution light goes off.



Essential Bus Circuit Breaker Out Caution Light
and Test System
Figure 24-50-4

(9) Replace trim panels or couch as required.

(10) Reset master caution light.

(11) Disconnect external power source from airplane.

2. Essential Bus 28V DC Circuit Breaker Functional Check - (NA-370-1 Thru -9 Not Changed by Service Bulletin No. 11)

A. This procedure consists of making a functional check of the 100 ampere, 28 volt dc essential bus circuit breakers located in the cabin and cockpit. Steps (1), (2), (3), (5), (6) and (7) may be used to perform this check with the circuit breakers installed in airplane. Steps (1), (2), (4), (5), (6) and (7) may be used when the circuit breakers are removed from the airplane. To perform the circuit breaker functional check, proceed as follows:

(1) Gain access to fuselage station 333 electrical power distribution box and remove copilot's seat.

(2) Remove trim panel from forward side of fuselage station 143 adjacent to essential circuit breakers panel. Remove panel from bulkhead.

(3) To perform check with circuit breakers installed in the airplane, proceed as follows:

(a) Pull (open) the No. 2, 3, 4 and 5 circuit breakers at the fuselage station 143 and 333 (see figure 24-50-5).

NOTE: A 105 ampere load may be applied to the aircraft bus system by utilizing a load bank or using the aircraft electrical equipment.

(b) Connect load bank to essential bus (if used).

(c) Connect external 28V DC electrical power to airplane and adjust load bank or turn on the necessary electrical equipment to obtain a 105 ampere load on the essential bus.

NOTE: If the external power unit contains an ammeter and it is known to be accurate, it may be used to monitor the 105 ampere load.

(d) Allow the load to remain on both No. 1 circuit breakers for one hour. The circuit breakers must remain engaged during the one hour period.

(e) Subsequent to the one hour period, and with the 105 ampere load applied to the circuit breakers, measure the voltage drop across both No. 1 circuit breakers (see figure 24-50-5). If voltage drop exceeds 0.5 volts, replace the applicable circuit breaker. Replacement circuit breakers shall be dated July 8, 1974 or later.

(f) Close the No. 2 circuit breakers at fuselage stations 143 and 333; then open the No. 1 circuit breakers and repeat steps (d) and (e).

(g) Repeat steps (d) through (f) on the No. 3, 4, and 5 circuit breakers.

(h) Remove electrical power from airplane.

(i) Check bus bar and electrical cable attachment to circuit breakers for security. Torque any loose bolt(s) to 50-60 inch-pounds.

CAUTION: TO PREVENT DAMAGE TO CIRCUIT BREAKERS, SUPPORT THE CIRCUIT BREAKER LUGS PRIOR TO TORQUING BOLTS.

(4) To perform a functional check of all 28 volt dc essential circuit breakers when removed from the airplane, proceed as follows:

(a) Remove bolts securing circuit breaker lugs to bus bar and electrical cables at fuselage stations 143 and 333.

CAUTION: TO PREVENT DAMAGE TO CIRCUIT BREAKERS, SUPPORT THE CIRCUIT BREAKER LUGS PRIOR TO REMOVING BOLTS.

- DO NOT DROP HARDWARE IN POWER DISTRIBUTION BOX.

(b) Mark circuit breaker location, then remove screws securing circuit breakers to panel, then remove circuit breakers from airplane.

(c) Using a suitable bus bar or electrical cable capable of carrying at least 105 amperes, connect all circuit breakers in series (see figure 24-50-5).

(d) Connect a 28 VDC power supply (minimum rating of 105 amperes) to one end and a 105 ampere load to the other end of the circuit breaker string.

(e) Check that all circuit breakers are engaged, and turn on the power supply. Adjust load to obtain 105 amperes.

(f) Allow the load to remain on the circuit breaker string for one hour. All circuit breakers must remain engaged during the one hour period.

(g) Subsequent to the one hour period, and with the 105 ampere load applied to the circuit breaker string, measure the voltage drop across each circuit breaker. If any voltage drop exceeds 0.5 volts, replace applicable circuit breaker. Replacement circuit breakers shall be dated July 8, 1974 or later.

(h) Remove electrical power from circuit breaker string and remove circuit breakers from test set up.

(i) Replace circuit breakers in original locations and install supporting screws.

(j) Position bus bar and electrical cables on circuit breaker lugs. Install and torque bolts to 50-60 inch-pounds.

CAUTION: TO PREVENT DAMAGE TO CIRCUIT BREAKERS, SUPPORT CIRCUIT BREAKER LUGS PRIOR TO TORQUING BOLTS.

- ENSURE THAT NO BOLTS HAVE BEEN INADVERTENTLY MISSED DURING THE TORQUING PROCEDURE.

(5) Check and remove any foreign material from circuit breaker panels.

(6) Check all components for broken or loose electrical wires, and component security.

(7) Install panel covers and applicable trim panels and copilot's seat.

3. Removal and Installation of Essential Bus Circuit Breaker Out Test Solenoid (NA-370 Airplanes Changed by Service Bulletin 75-20)

A. To remove solenoid, proceed as follows:

(1) Remove all external electrical power from airplane and disconnect both batteries.

(2) Remove trim panels or couch as required to gain access to electrical power distribution box located at station 333.

(3) Open relay box.

(4) Disconnect two wires, at splice points, from rear of L100 solenoid. Tag wires for later identification.

(5) Raise and support switch actuator (circuit breaker cover) and remove two screws securing solenoid bracket and remove solenoid and bracket assembly.

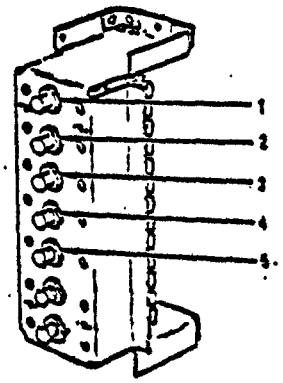
(6) Remove solenoid mounting nut and separate solenoid from bracket.

(7) Using two wrenches, loosen spacer jam nut on plunger. Remove spacer and jam nut from solenoid plunger. Retain for reinstallation on new solenoid.

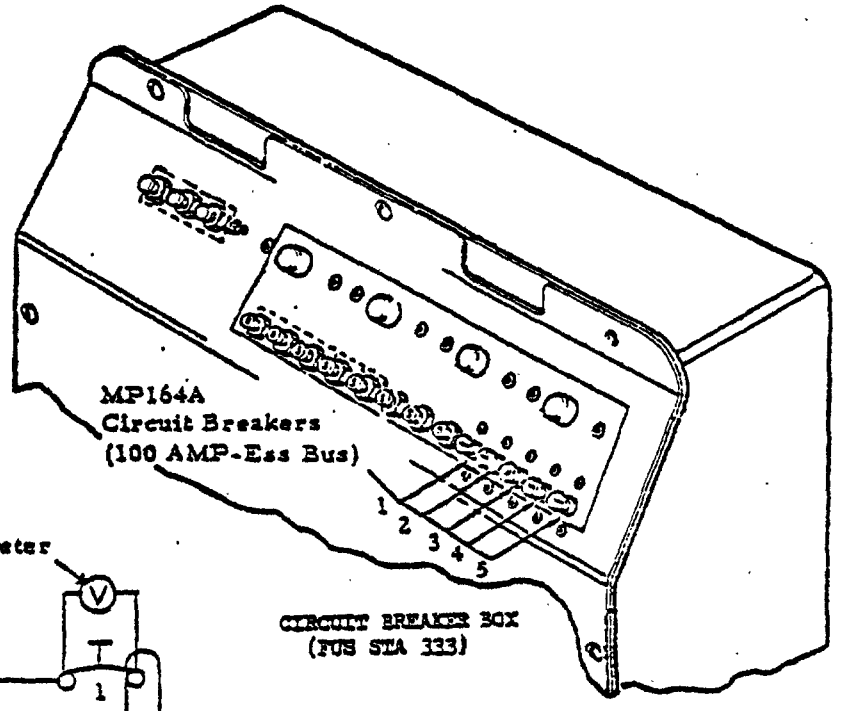
B. To install solenoid, proceed as follows:

(1) Install bracket on solenoid as shown in figure 24-50-4.

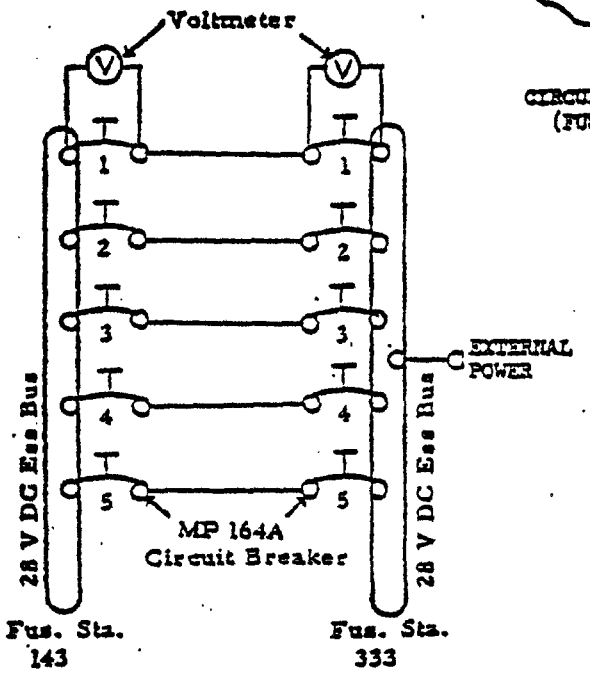
(2) Check mounting nut for security, then install assembly between circuit breakers with two solenoid bracket attaching screws. Lift switch actuator (cover) to gain access to screw locations.

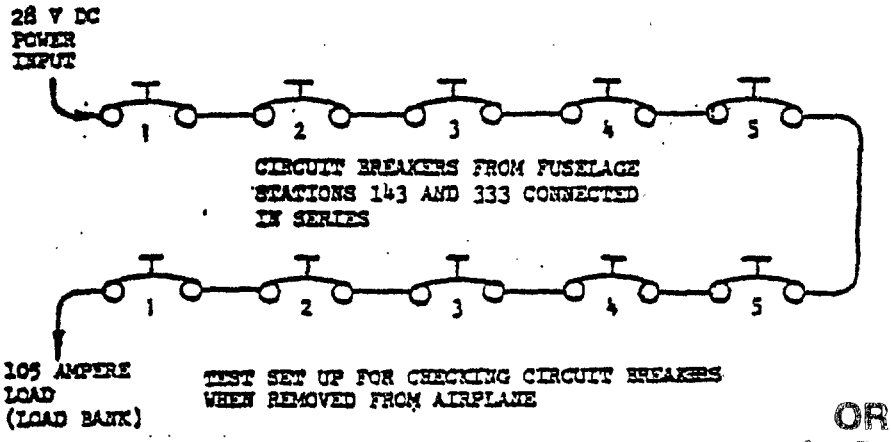


CIRCUIT BREAKER
PANEL (FUS STA 143)



CIRCUIT BREAKER BOX
(FUS STA 333)





TEST SET UP FOR CHECKING CIRCUIT BREAKERS
WHEN REMOVED FROM AIRPLANE

ORIGINAL
As Received By
ATP

Essential Bus 28VDC Circuit Breakers
Figure 24-50-5

(3) Splice wires (tagged in solenoid removal procedure) to solenoid stub wires.

NOTE: Do not remove jam nut and spacer from solenoid plunger, since plunger may then fall out of solenoid.

(4) Using two wrenches, loosen jam nut on solenoid plunger and adjust plunger spacer so that end of spacer is 0.65-inch from mounting plate when plunger is pulled fully forward. See figure 24-50-4.

(5) Check interior of relay box for any debris. Remove debris.

(6) Close relay box.

(7) Check switch actuator mechanism (refer to paragraph 1B).

(8) Replace trim panels or couch as required.

(9) Reconnect batteries.

(10) Check caution light system circuitry (refer to paragraph 1A).

CHAPTER 25

EQUIPMENT/ FURNISHINGS

CHAPTER 25

EQUIPMENT/FURNISHINGS

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CHAPTER 25

EQUIPMENT/FURNISHINGS

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		3/4	May 15/81
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25-20-00		201	May 31/84
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*The asterisk indicates pages revised, added, or deleted by the current revision.

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GENERAL

CLEANING/PAINTING

AIRPLANE INTERIOR

The airplane interior should be frequently cleaned to maintain its appearance and to minimize deterioration.

CAUTION

- ***Do not use abrasive cleaners on any interior surface. Abrasive cleaners may damage leathers, painted surfaces, fabrics, and plastic surfaces.***
- ***Avoid using cleaners that contain organic chemicals, particularly ketones, nitro compounds, esters, aromatic amines, and hydrocarbons.***
- ***Avoid using materials such as fire extinguisher fluids, de-icing fluids, hydraulic fluids, or paint thinners as interior cleaners.***

Most interior surfaces can be cleaned by dusting, vacuuming, and/or wiping with a mild soap and water mixture.

If an interior surface is heavily soiled or stained, consult the manufacturer for a suggested cleaning agent and procedure.

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FLIGHT COMPARTMENT - DESCRIPTION AND OPERATION

1. Pilot's and Copilot's Seats (See figure 25-10-1.)

Two crew seats are provided in the pilot's compartment. Each seat is equipped with a quick-disconnect combination lap belt and shoulder harness; two armrests, which are raised upward and aft to a stowed position; removable seat and back cushions; and adjustment levers for vertical as well as fore-and-aft movement. A combination map pocket and flight report holder is provided in the pilot's seat back. Each seat may be individually adjusted manually by two levers underneath the seat on the inboard side. Fore-and-aft adjustment is accomplished by aft movement of the lever that extends upward from the floor. Vertical adjustment is accomplished by raising the horizontal lever and holding it in that position until the proper adjustment is made. Releasing the lever allows the locking pin to enter an adjustment hole. Both vertical and horizontal adjustments are in 1/2-inch increments with a full travel of 5 inches.

2. Safety Belts and Inertia Reels

A. Pilot's and Copilot's Safety Belts

The safety belt assembly consists of a lap belt, a crotch strap, and a shoulder harness. The lap belt and crotch strap are attached separately to the seat. Each strap has an attachment fitting which is inserted into a single quick-release buckle attached to one side of the seat belt. Turning the face of the buckle approximately one eight of a turn in either direction will immediately release all plug-in connections. In addition, there is an alternate release at the top of the buckle for the shoulder straps only. This permits extended forward movement of the head and shoulders without releasing the complete harness assembly.

B. Inertia Reels

A dual inertia reel for the shoulder harness straps is provided within the bulkhead immediately aft of each crew seat. No manual locking device is provided for the shoulder straps, as the automatic locking device within the inertia reel will lock at a maximum forward acceleration of 1.4 G and will function with the reel in any position. When the shoulder straps are released from the belt buckle, they will retract, eliminating the hazard of loose straps in the pilot's compartment.

The action of the inertia reel may be checked by grasping the loose ends of the shoulder straps and then pulling quickly. The harness should lock up. When the load is eased, the harness should rewind.

3. Foot and Knee Heater Pads

Electric heating elements inside rubber pads are installed under the carpet in the pilot's and copilot's area. The knee heater pads are installed on the inboard side of the circuit breaker console panel under the trim panels. Operation of the heater pads is automatic when the "FOOT HEATERS" circuit breaker on the upper aft right-

hand circuit-breaker panel is engaged, and the 28-volt dc essential bus is energized. A thermal switch, mounted on the fuselage frame at right station 143, controls operation of the heating pads. This switch closes at 28°F, completing the electrical circuit to the heating pads. When the temperature at the switch increases to 43°F, the switch opens, shutting off the current to the heating pads. The 28-volt dc essential bus provides the power to heat the pads. (See figure 25-10-2.)

NOTE: The foot heaters are optional equipment for the cabin area.

4. Foot-warmer Blower-Heaters - Optional Equipment

An optional foot-warmer blower-heater may be installed underneath the pilot's and copilot's seats on a portion of the seat assemblies. The blower provides a flow of air that passes a dc-powered heating unit, causing heated air to flow from the heater outlet. Each blower-heater is controlled by a "FOOT WARMER" switch on each side of the cockpit, just above the respective consoles. When the switch is placed at BLOWER, only the blower operates. When the switch is placed at HEATER, both the blower and the heater operate.

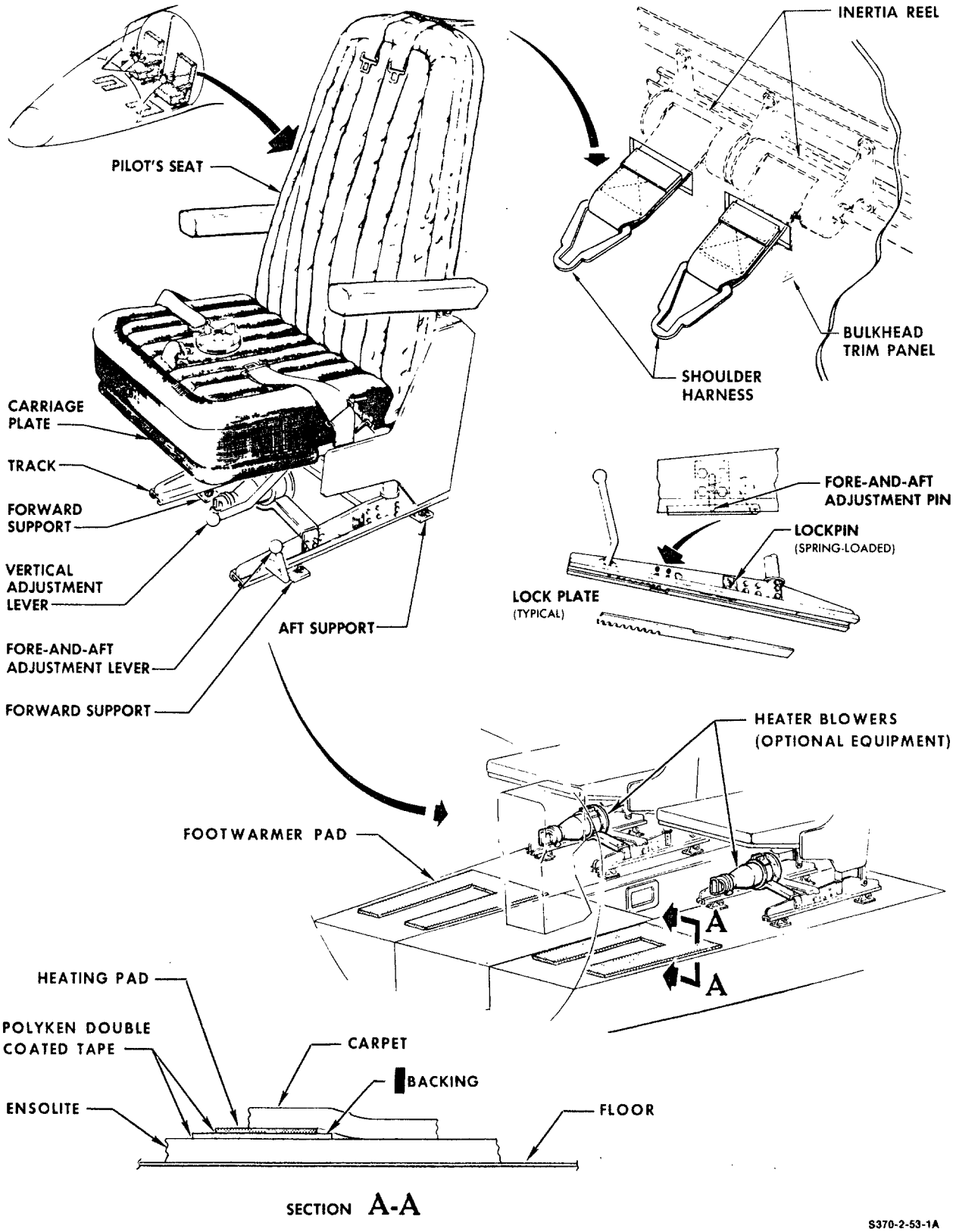
The blower is a 26- to 30-volt dc motor-driven unit that operates at 5700 rpm. Maximum starting current for the blower is 4.0 amperes. Normal operating current is 0.45 ampere.

The heater is a Calrod type with a maximum power consumption of 425 watts (±10%) at 27 volts dc. Maximum starting current for the heater is 20.0 amperes. Normal operating current is 15.7 amperes.

The blower-heater is protected from overheating by a fusible link that will open the heater circuit when the temperature goes to 246.1°C (±27.8°C) [475°F(±50°F)].

5. Manual and Automatic Controlled Shoulder Harness - Optional Equipment

A manual and automatic shoulder harness inertia reel is mounted on the bulkhead immediately aft of the crew seat. A control lever, located on the cockpit sidewall, is placarded with aft AUTOMATIC and forward MANUAL LOCK positions. The AUTOMATIC position normally allows freedom of movement with the reel maintaining spring tension on the harness, but in emergency the reel will lock automatically on forward impact. The MANUAL LOCK position allows locking the reel with spring takeup of any slack. With the control lever in AUTOMATIC, if the reel automatically locks, it can be unlocked by moving the lever to MANUAL LOCK and back to AUTOMATIC. If a malfunction jams the reel in the locked condition preventing freedom of movement, the shoulder harness can be independently disconnected from the lap belt buckle by depressing the tab at the top of the buckle.



SECTION A-A

S370-2-53-1A

Flight Compartment Equipment Furnishings
Figure 25-10-1

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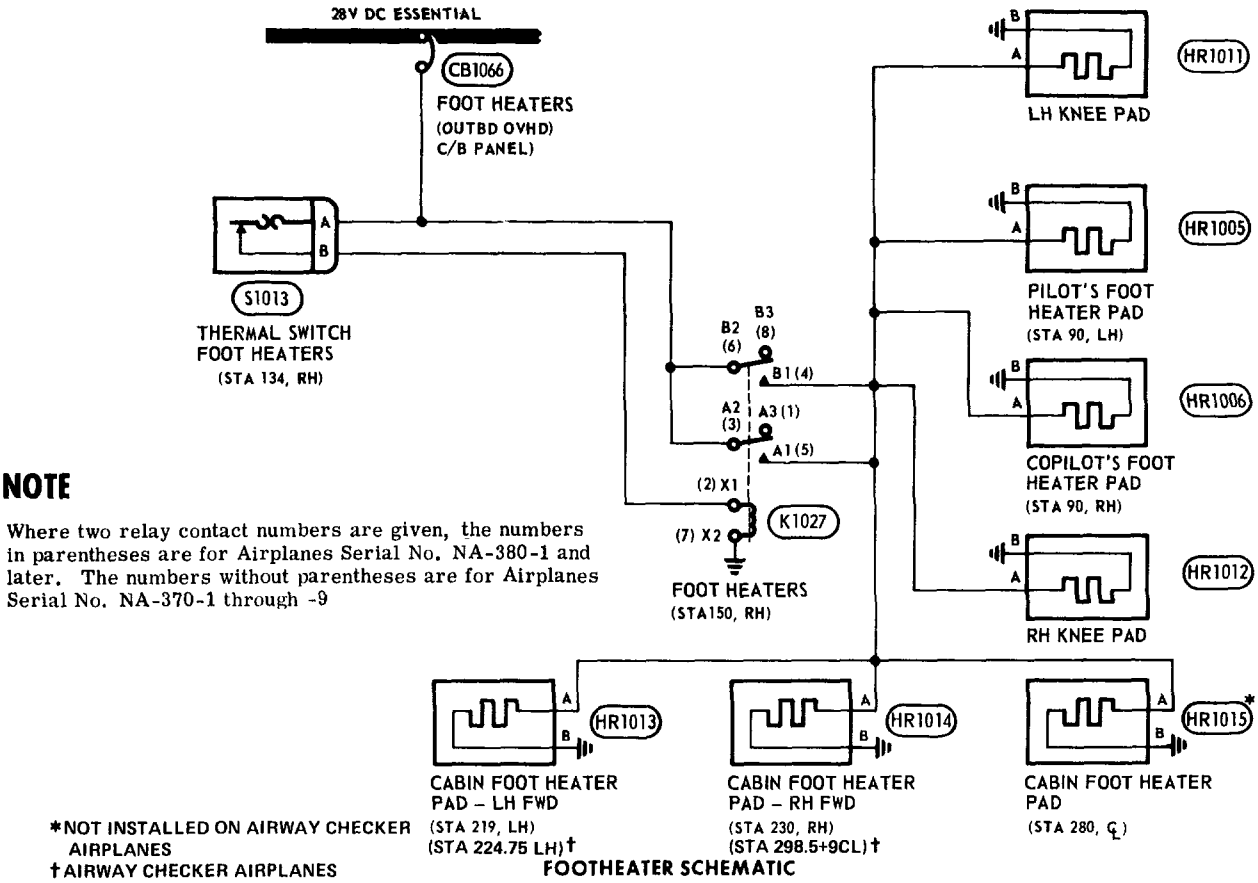
FLIGHT COMPARTMENT - TROUBLE SHOOTING

1. Trouble Shooting Foot Heaters

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. HEATING PADS DO NOT OPERATE.		
(1) Thermal switch not operating.	Attach jumper wire between "A" and "B" terminals of switch and check for heating of pads.	Replace thermal switch.
(2) Foot heaters relay inoperative.	With jumper wire in place at thermal switch, check for continuity between A1 and A2, and B1 and B2, of relay.	Replace relay.
(3) Heating pads inoperative.	With jumper wire in place at thermal switch, check for continuity between terminals "A" and "B" on heating pad.	Replace heating pad.
(4) Faulty wiring.	With jumper in place at thermal switch, perform continuity check of wiring.	Repair faulty wiring.

2. Trouble Shooting Foot-warmer Blower-Heater

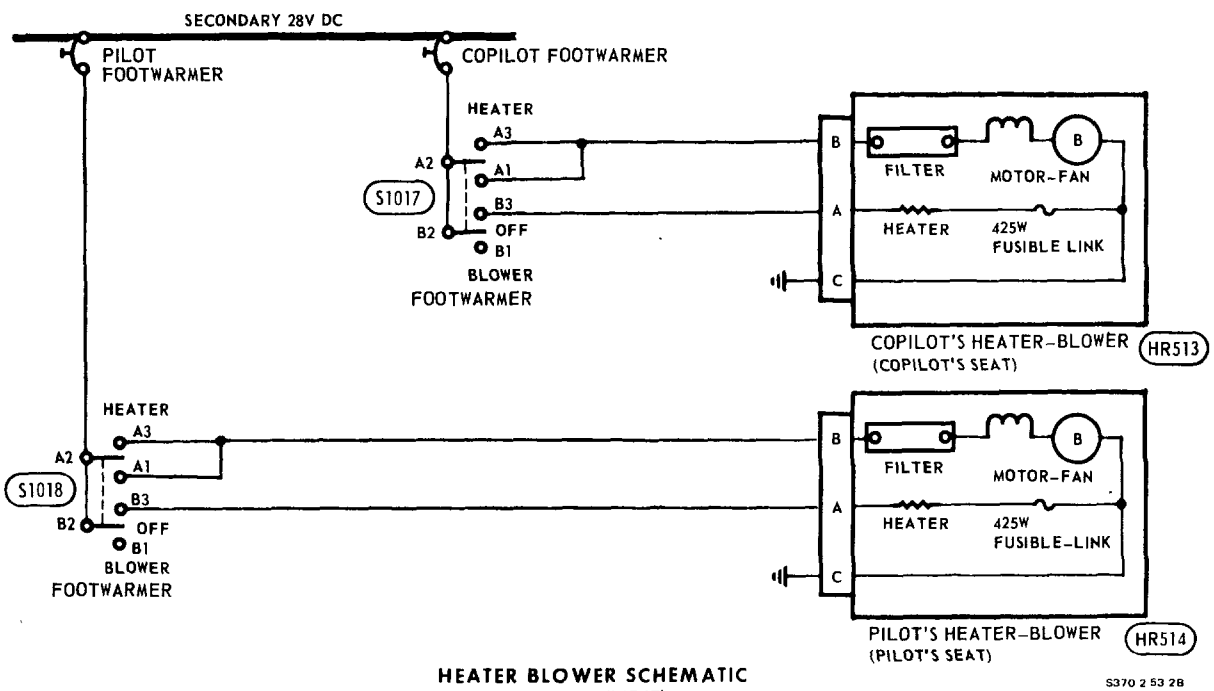
PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. FOOT-WARMER BLOWER-HEATER WILL NOT PROVIDE WARM AIRFLOW.		
(1) Blower-heater overheat fusible link open.	Check for operation of heater and blower. (Refer to paragraph 10.) If only blower functions, heater circuit is open.	Replace blower-heater.
(2) Wiring within blower motor defective.	Check operation of blower. (Refer to paragraph 10.)	Replace blower-heater.



FOOTHEATER SCHEMATIC

NOTE

Where two relay contact numbers are given, the numbers in parentheses are for Airplanes Serial No. NA-380-1 and later. The numbers without parentheses are for Airplanes Serial No. NA-370-1 through -9



HEATER BLOWER SCHEMATIC
(OPTIONAL EQUIPMENT)

Foot Heaters and Heater Blowers Electrical Schematic
Figure 25-10-2

FLIGHT COMPARTMENT - MAINTENANCE PRACTICES

CAUTION: USE CARE WHEN REMOVING AND INSTALLING COCKPIT AND CABIN SEATS TO ENSURE THAT THE LEFT AND RIGHT COCKPIT KNEE HEATER PADS AND THE COCKPIT AND CABIN FOOT HEATER PADS ARE NOT DAMAGED. UNDETECTED DAMAGE TO THE ELECTRIC HEATING PADS MAY RESULT IN BURNED PADS THAT CAN EMIT PUNGENT ODORS.

1. Removal of Seat

A. See figure 25-20-1 for parts identification, and proceed as follows:

(1) Raise lockpins and lift seat fore-and-aft adjustment handle to release seat lock plates.

(2) Slide seat forward enough to clear aft supports and release adjustment handle; then lift seat from forward supports.

NOTE: With the seat removed from the supports, the lock plates are free to slide out of the tracks. To retain lock plates in tracks, raise seat fore-and-aft adjustment handle, position lock plates midway in track, and release handle.

2. Installation of Seat

A. It is recommended that an assistant aid in the following installation procedure.

(1) If the lock plates have been removed, slide them into each track. Ensure that adjustment positions on lock plates are in proper relation to adjustment pin in track.

(2) With lockpins up, align tracks with forward supports. Raise seat fore-and-aft adjustment handle, and slide seat forward on supports.

(3) Align tracks with rear supports. Raise adjustment handle, and slide seat back to engage aft supports.

(4) Engage lockpins in lock plates, and check seat fore-and-aft adjustments through full range of travel.

3. Removal of Safety Belt and Inertia Reel

A. Remove safety belt and inertia reel as follows:

(1) Remove seat cushion from seat pan.

(2) Remove crotch strap by removing the two screws, nuts, and washers on forward bottom of seat pan.

(3) Unsnap lap belt from fittings at rear and on each side of seat pan.

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(4) Detach bulkhead trim panel, behind seat.

(5) Remove the six attaching bolts securing reel assembly to bulkhead fitting.

(6) Pull shoulder harness straps through guide on top of seat back.

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4. Installation of Safety Belt and Inertia Reel

A. Install safety belt and inertia reel as follows:

(1) Position inertia reel on bulkhead fitting and install six attaching bolts.

(2) Thread harness through bulkhead trim panel and through guides on seat back. Seucre trim panel to bulkhead.

(3) Thread crotch strap through seat pan and seat cushion.

(4) Secure crotch strap to seat pan with two screws, nuts, and washers.

(5) Attach seat belts to fittings on each side of seat pan.

5. Removal of Foot Heaters

A. Remove foot heaters as follows:

(1) Disengage "FOOT HEATERS" circuit breaker.

(2) Lift up carpet in area of heating pad and disconnect electrical leads from pad.

(3) Peel up heater pad. (The pad is fastened down with double-coated tape to a heater backup sheet which is in turn taped down to the Ensolite padding. Do not remove backup sheet from padding.)

6. Installation of Foot Heaters

A. Install foot heaters as follows:

(1) Lift up carpet in area and fasten heater pad to heater backup sheet, using double-coated tape.

(2) Ensure that "FOOT HEATER" circuit breaker is disengaged; then connect electrical leads to heater pad. (See figure 25-10-2.)

(3) Replace carpet.

6A. Removal of Knee Heater Pads

A. Remove knee heater pad(s) as follows:

(1) Disengage "FOOT HEATERS" circuit breaker.

(2) Remove trim panels from applicable side of console.

(3) Disconnect electrical plug to heater pad(s).

(4) Remove heater pad(s).

6B. Installation of Knee Heater Pads

A. Install knee heater pad(s) as follows:

(1) Connect electrical plug to heater pad(s).

(2) Install heater pad(s) to side of console.

(3) Perform operational check-out of system.

(4) Install trim panel on side of console.

7. Deleted. Information moved to table 601.

8. Removal of Foot-warmer Blower-Heater

A. Remove foot-warmer blower heater as follows:

(1) Make sure "FOOT WARMER" SWITCH is at OFF.

(2) Remove seat.

(3) Remove nuts, screws, and washers from clamp holding blow-heater.

(4) Lift blower-heater away from clamp.

9. Installation of Foot-warmer Blower-Heater

A. Install foot-warmer blower-heater as follows:

(1) Invert seat; then position blower-heater in clamp.

(2) Install screws, washers, and nuts. Tighten nuts sufficiently to hold blower-heater, but loose enough to allow for blower-heater unit rotation.

(3) Install seat.

(4) Rotate blower-heater unit until clamp holding wiring is at its closest point to blower-heater plug receptacle.

(5) Tighten nuts on mounting clamp.

(6) Install blower-heater plug to receptacle.

(7) Check operation of blower-heater. (Refer to paragraph 10.)

10. Deleted. Information moved to table 602.

11. Portable Fire Extinguisher (Cabin and/or Cockpit) Agent Container - Weight Check

A. Weigh container as follows:

(1) Remove portable fire extinguisher container from airplane.

(2) Weigh container and check that weight agrees with total weight stenciled on container.

(3) Install serviceable container in airplane.

12. Portable Fire Extinguish (Cabin and/or Cockpit) Agent Container - Hydrostatic Test

A. To perform hydrostatic test of portable fire bottle container, proceed as follows:

(1) Remove portable fire extinguisher container from airplane.

(2) Hydrostatic test container (approved agency).

(3) Install container after hydrostatic test (or replacement container).

13. Deleted. Information moved to Inspect - Cockpit Heater Pads.

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FLIGHT COMPARTMENT

INSPECTION/CHECK

OPERATIONAL CHECK — COCKPIT FLOOR OR KNEE HEATER PADS

Equipment/Tools

- 28 VDC Electrical Power Supply
- Electrical Jumper Wire

Consumables

- None

Procedure

Table 601. Operational Check — Cockpit Floor or Knee Heater Pads

PROCEDURE	NORMAL INDICATION
NOTE ***Bypassing the thermal switch permits checking the operation of the foot heater and knee heater pads regardless of outside or cabin air temperature.***	
1. FOOT HEATER circuit breaker ..... out	
2. Electrical power ..... off	
3. Connect jumper wire between terminals "A" and "B" on thermal switch.	
4. Electrical power ..... on	
5. FOOT HEATER circuit breaker ..... in	Heater pads ..... heat
6. Airplane ..... secure as desired	

OPERATIONAL CHECK — FOOT-WARMER BLOWER HEATER

Equipment/Tools

- 28 VDC Electrical Power Supply

Consumables

- None

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INSPECTION/CHECK (Continued)

Procedures (Cont'd)

Table 602. Operational Check — Foot-Warmer Blower-Heater (Cont'd)

PROCEDURE	NORMAL INDICATION
1. Electrical power ..... on	
2. FOOT WARMER sw ..... BLOWER	Check for flow of cool air from blower-heater outlet.
3. FOOT WARMER sw ..... Heater and wait 5 minutes	Check for flow of warm air from blower-heater outlet.
CAUTION *Set FOOT WARMER sw to OFF if blower does not operate when heating unit heats up. The heater will overheat and cause separation of the overheat fusible link.*	
4. FOOT WARMER sw ..... off	
5. Repeat steps 2 thru 4 using other FOOT WARMER sw and blower-heater.	
6. Airplane ..... secure as desired	A. Blower ceases to rotate. B. Heater cools to ambient temperature.

INSPECT — COCKPIT FLOOR HEATER PADS

Equipment/Tools

- 28 VDC Electrical Power Supply
- Digital Multimeter, Fluke 8022 or Equivalent

Consumables

- Tape, Double-Coated

Procedure

1. FOOT HEATER circuit breaker ..... out
2. Electrical power ..... off
3. Lift up carpet in area of heating pad(s).
4. Disconnect electrical leads from cockpit floor heater pad(s).

NOTE

Heater pad is secured with double-coated tape to a heater backup sheet that is taped to the Ensolite padding. Do not remove backup sheet from padding.

5. Peel up heater pad(s).
6. Visually inspect heater pads for the following:
   A. Loose or broken wires.
   B. Darkened or charred areas.
   C. Delamination, cuts, or tears.
7. Check resistance of pad(s) for resistance tolerances. (Refer to table 603 for resistance tolerances.)

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INSPECTION/CHECK (Continued)

INSPECT — COCKPIT FLOOR HEATER PADS (Cont'd)

8. Replace pad(s) if defects are found and/or if resistance is outside resistance tolerances.
9. Secure heater pads to heater backup sheets with tape.
10. Connect electrical leads to heater pad(s).
11. Install carpeting.
12. Perform operational check of cockpit floor heater pad system.
13. Airplane ..... secure as desired

INSPECT — COCKPIT KNEE HEATER PADS

Equipment/Tools

- 28 VDC Electrical Power supply
- Digital Multimeter, Fluke 8022 or Equivalent.

Consumables

- None

Procedure

1. FOOT HEATER circuit breaker ..... out
2. Electrical power ..... off
3. Remove trim from side of console.
4. Disconnect electrical leads from knee heater pad(s).
5. Remove heater pad(s).
6. Visually inspect heater pads for the following:
   A. Loose or broken wires.
   B. Darkened or charred areas.
   C. Delamination, cuts, or tears.
7. Check resistance of pad(s) for resistance tolerances. (Refer to table 603 for resistance tolerances.)
8. Replace pad(s) if defects are found and/or if resistance is outside resistance tolerances.
9. Install heater pad(s).
10. Connect electrical leads to heater pad(s).
11. Install side trim panels.
12. Perform operational check of cabin floor heater pad system.
13. Airplane ..... secure as desired

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INSPECTION/CHECK (Continued)

INSPECT — COCKPIT KNEE HEATER PADS (Cont'd)

Table 603. Cockpit and Knee Heater Pad Resistance Tolerances

Manufacturer / Electrofilm P/N	Outside Dimensions (In.)	Resistance (Ohms)	
		Min	Max
103169	19.0x22.0 (With Notch for control column)	18.51	22.62
112422-1	14.0x18.0 (Cockpit side panel)	17.82	21.77
112422-2	14.0x18.0 (Cockpit side panel)	17.82	21.77
Goodyear P/N			
4056-2256-2	19.0x22.0 (With notch for [A] control column)	12.25(±5%)	

NOTE

[A] Dimensions do not include backing.

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PASSENGER COMPARTMENTS

MAINTENANCE PRACTICES

REPLACEMENT/INSPECTION — CABIN FLOOR HEATER PADS

Equipment/Tools

- 28 VDC Electrical Power Supply
- Digital Multimeter, Fluke 8022 or Equivalent

Consumables

- Tape, Double coated

Removable Procedures

1. FOOT HEATERS circuit breaker ..... out
2. Remove cabin seats.
3. Roll up carpet in cabin to expose heater pads.
4. Disconnect electrical leads from pads.

NOTE

Heater pad is secured with double-coated tape to a heater backup sheet that is taped to the Ensolite padding. Do not remove backup sheet from padding.

5. Peel up heater pad.

Inspection Procedures

1. Visually inspect heater pads for the following:
   A. Loose or broken wires.
   B. Darkened or charred areas.
   C. Delamination, cuts, or tears.
2. Check resistance of pad for resistance tolerances. (Refer to table 201 for cabin pad resistance tolerances.)
3. Replace heater pads as required.

Installation Procedures

1. Secure heating pad to heater backup sheet with tape.
2. Connect electrical leads to heater pad(s).
3. Perform an operational check of cabin floor heater pads.
4. Install carpet and cabin seats.
5. Airplane ..... secure as desired.

OPERATIONAL CHECK — CABIN FLOOR HEATER PADS

Equipment/Tools

- 28 VDC Electrical Power Supply
- Electrical Jumper Wire

Consumables

- None

Procedure

See table 202. Operational Check — Cabin Floor Heater Pads.

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MAINTENANCE PRACTICES (Continued)

REPLACEMENT/INSPECTION — FOOT HEATERS (Cont'd)

Table 201. Cabin Heater Pad Resistance Tolerances

Manufacturer	Outside Dimensions (In.)	Resistance (Ohms)	
Electrofilm P/N		Min	Max
104509	15.0x24.0	16.45	20.10
104510	13.5x40.0	11.00	13.45
108301	19.25x24.0	7.20	9.80
108302	7.5x24.0	18.80	23.60
Goodyear P/N			
4056-2256-1	15.0x24.0 A	10.90(±5%)	

Table 202. Operational Check — Cabin Floor Heater Pads

PROCEDURE	NORMAL INDICATION
NOTE ***Bypassing the thermal switch permits checking the operation of the foot heater and knee heater pads regardless of outside or cabin air temperature.***	
1. Electrical power ..... off	
2. FOOT HEATER circuit breaker ..... out	
3. Connect jumper wire between terminals "A" and "B" on thermal switch.	
4. Electrical power ..... on	
5. FOOT HEATER circuit breaker ..... in	Heater pads ..... heat
6. Airplane ..... secure as desired	

NOTE

A Dimensions do not include backing.

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LAVATORIES

MAINTENANCE PRACTICES

INSPECTION — LAVATORY WASTE RECEPTACLES

1. Inspect all lavatory paper and linen waste receptacle enclosure access doors for proper operation, fit, sealing, and latching for the containment of possible trash fires.
2. Inspect all "No Smoking" decals for readability and security.
3. Correct any defects found during this inspection.

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CHAPTER

26

FIRE PROTECTION

CHAPTER 26

FIRE PROTECTION

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CHAPTER 26

FIRE PROTECTION

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*The asterisk indicates pages revised, added, or deleted by the current revision.

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FIRE PROTECTION - DESCRIPTION AND OPERATION

1. General

Fire protection provisions include a detecting and indicating system and an electrically controlled fire-extinguishing system for each engine nacelle. The detecting system uses sensing elements so positioned that high temperatures resulting from a fire in the engine nacelle area are detected. Two "T" handles incorporating red fire-warning lights notify the crew of a fire or overheat condition in the respective nacelle.

Two similar, but independent, electrically controlled fire-extinguishing systems are provided to control fires within the engine nacelles. Each system is armed by pulling the respective "FIRE PULL" handle. The contents of each agent container are discharged by positioning the fire-extinguisher selector switch to EXT NO. 1 or EXT NO. 2.

Airplanes with an APU installed, refer to Chapter 49 for Fire Extinguisher and Detection Systems.

FIRE DETECTION SYSTEM - DESCRIPTION AND OPERATION

1. Description and Operation

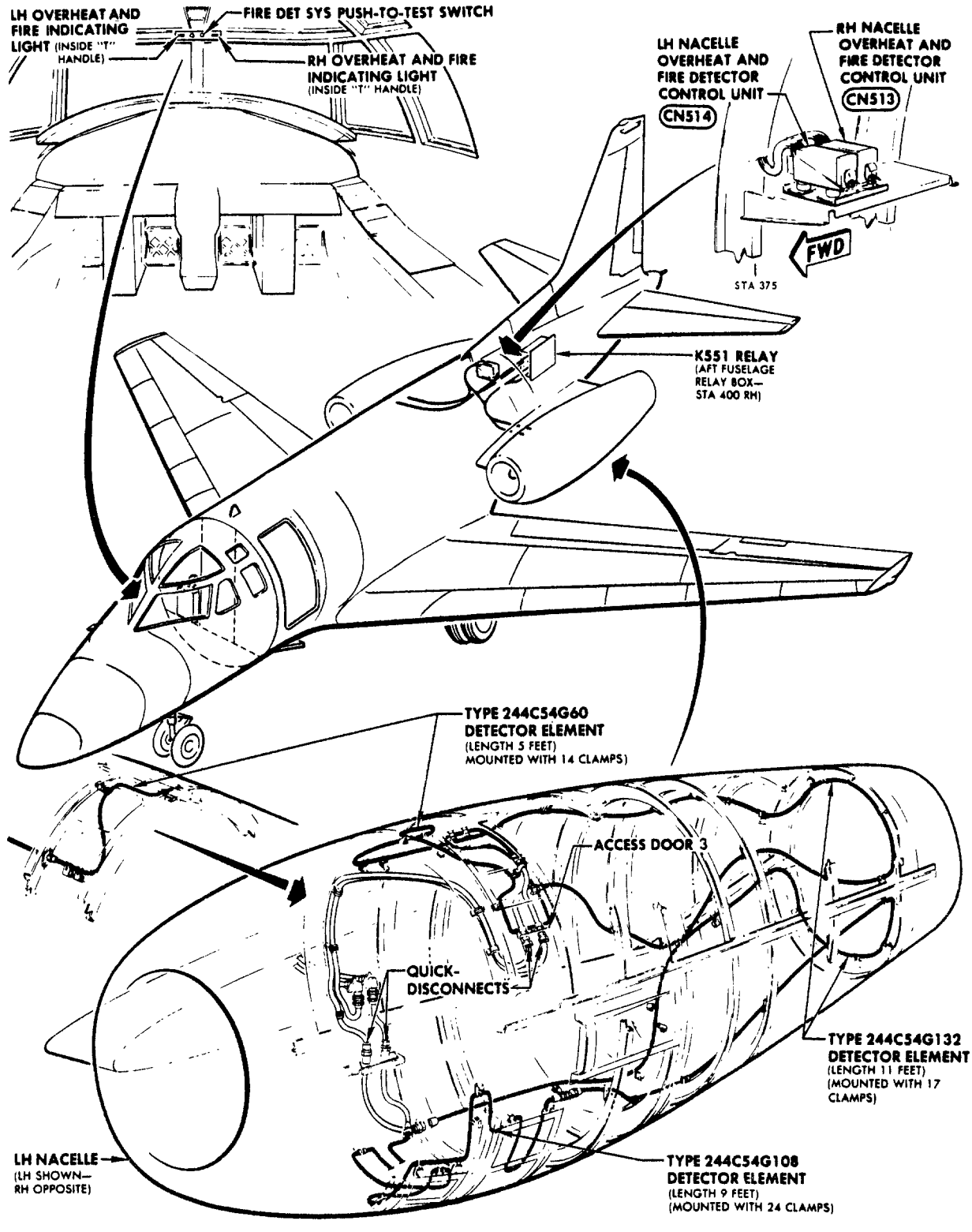
A. General

The engine overheat and fire detection system consists of two independently operated overheat and fire detection circuits. Each detection circuit consists of a control unit, detector elements, and a warning indicator light. (See figures 26-10-1 or 26-10-1A.) Common to both detection circuits are a push-to-test switch and a fire detector test relay. Each detection circuit is connected as an unbalanced Wheatstone bridge circuit. (See figures 26-10-2 or 26-10-2A.) Two legs of the bridge circuit (R_1 and R_2) are fixed resistors to known value and are inside the control unit. The third leg is a fixed resistor on the mounting bracket. The fourth leg consists of the fixed resistor (R_3) inside the control unit, shunted by the engine nacelle detector elements of semiconductor design. The detector elements function as variable resistors that vary inversely with the engine nacelle temperature. Bridge circuit operation is sensed by a polarized or sense relay which is connected across the bridge circuit inside the control unit and its sensitivity is controlled by the value of the fixed resistor on the mounting bracket. Its value is determined by the control unit manufacturer for proper circuit operation. When an overheat or fire condition occurs within the engine nacelle, the ambient temperature in that area rises and causes the resistance of the detector element to decrease. The bridge circuit operating condition passes through a balanced condition into an unbalanced condition, which is opposite to normal operation, and causes the warning indicator light to come on.

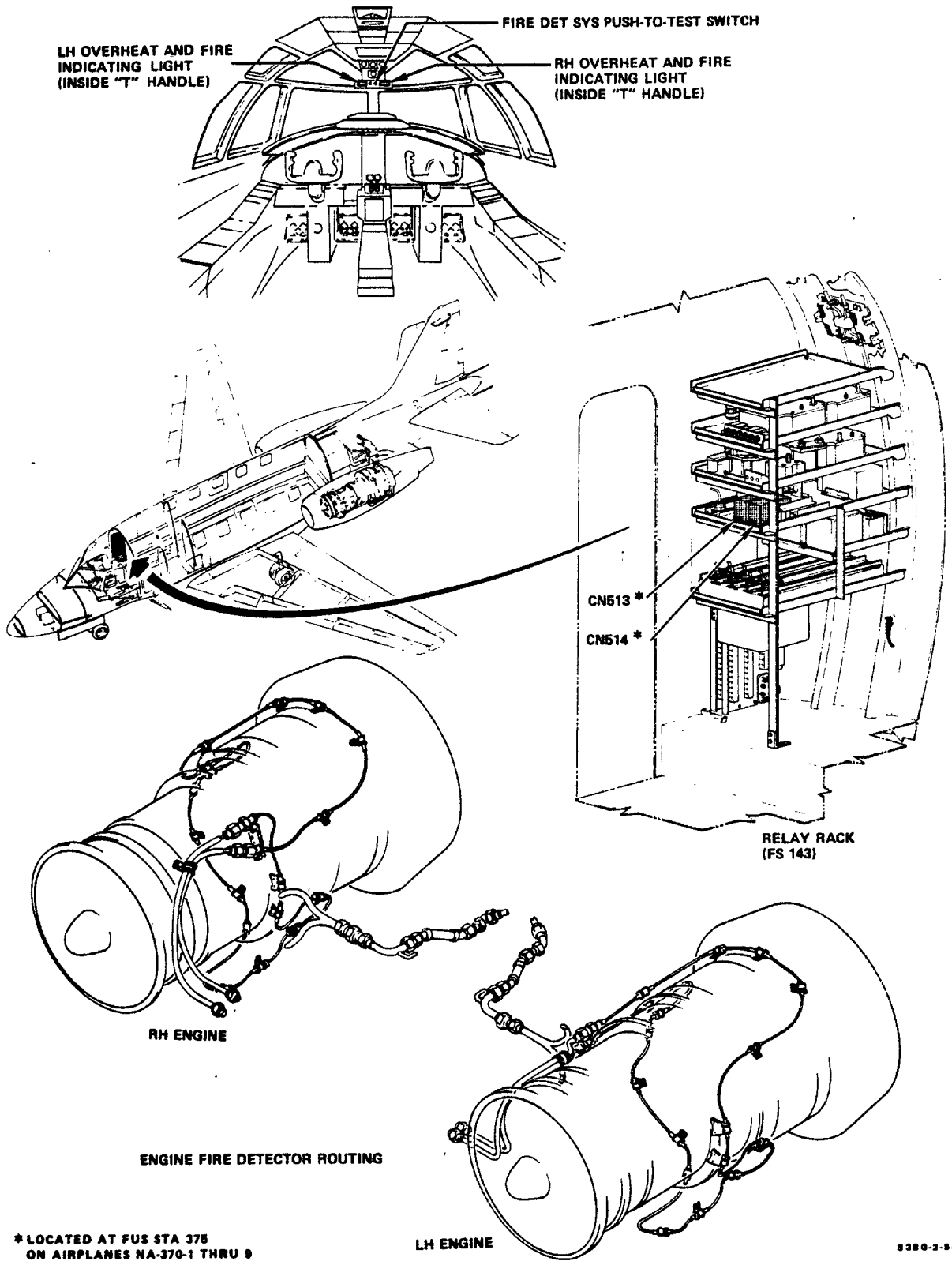
If the overheat or fire condition is remedied, the system condition reverses, the circuit returns to normal operation, and the warning indicator light is turned off. The detector circuit is then capable of indicating a subsequent overheat or fire condition should such a condition occur.

B. Detector Elements

Each detector element is an Inconel tube containing a single wire imbedded in a semiconductive ceramic material. The resistance of the semiconductor material varies inversely with nacelle temperature and with the physical



Engine Overheat and Fire Detection System Equipment Location - Airplanes Serial No. NA-370-2 Through -6, -8 and -9
Figure 26-10-1


FIRE DET SYS PUSH-TO-TEST SWITCH
LH OVERHEAT AND FIRE INDICATING LIGHT (INSIDE "T" HANDLE)
RH OVERHEAT AND FIRE INDICATING LIGHT (INSIDE "T" HANDLE)
CN513 *
CN514 *
RELAY RACK (FS 143)
RH ENGINE
LH ENGINE
ENGINE FIRE DETECTOR ROUTING
* LOCATED AT FUS STA 375 ON AIRPLANES NA-370-1 THRU 9
3380-2-54-8A

Engine Overheat and Fire Detection System Equipment Location - Airplanes Serial No. NA-370-1, -7, and NA-380-1 and Later
Figure 26-10-1A

length of the heated portion of the detector element. Therefore, an overheat or fire condition indication can result from an infinite number of engine nacelle temperature and heated element length combinations.

C. Control Units

The two hermetically sealed control units contain a Wheatstone bridge circuit, a polarized or sense relay, and a slave relay. The polarized or sense relay senses bridge circuit operation. The slave relay is energized by the operation of the polarized or sense relay. When energized, the slave relay turns on the system warning indicator light. An electrical connector on the bottom of the control unit plugs into an electrical connector on the mounting bracket, to connect with the airplane wiring harness and system detector elements.

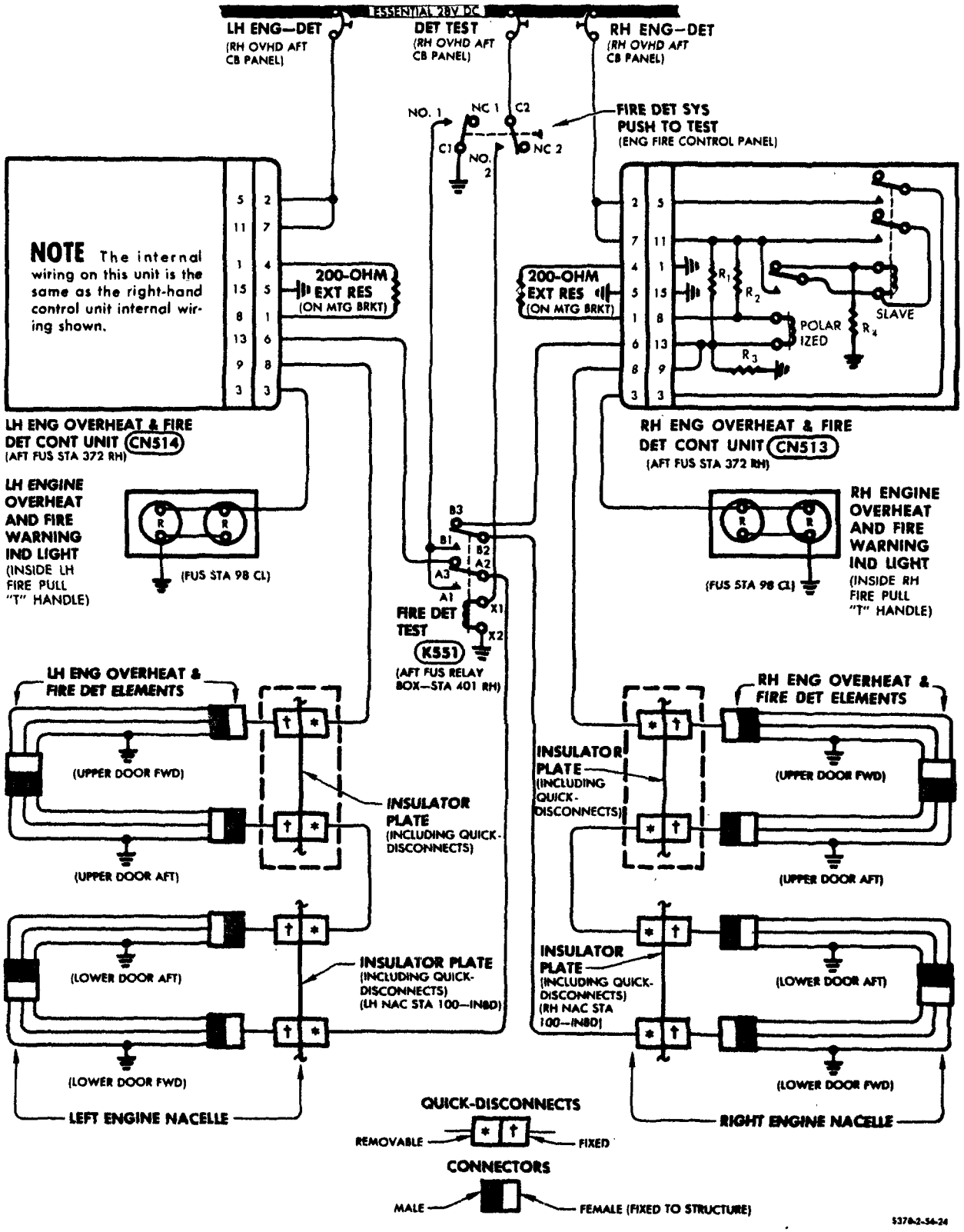
D. Push-to-Test Switch

The fire detection push-to-test switch on the fire-extinguishing panel checks the continuity of the circuits and the proper control unit operation of both detection systems. The test switch can be operated any time the 28-volt dc essential bus is energized and the "DET TEST" circuit breaker is pushed in.

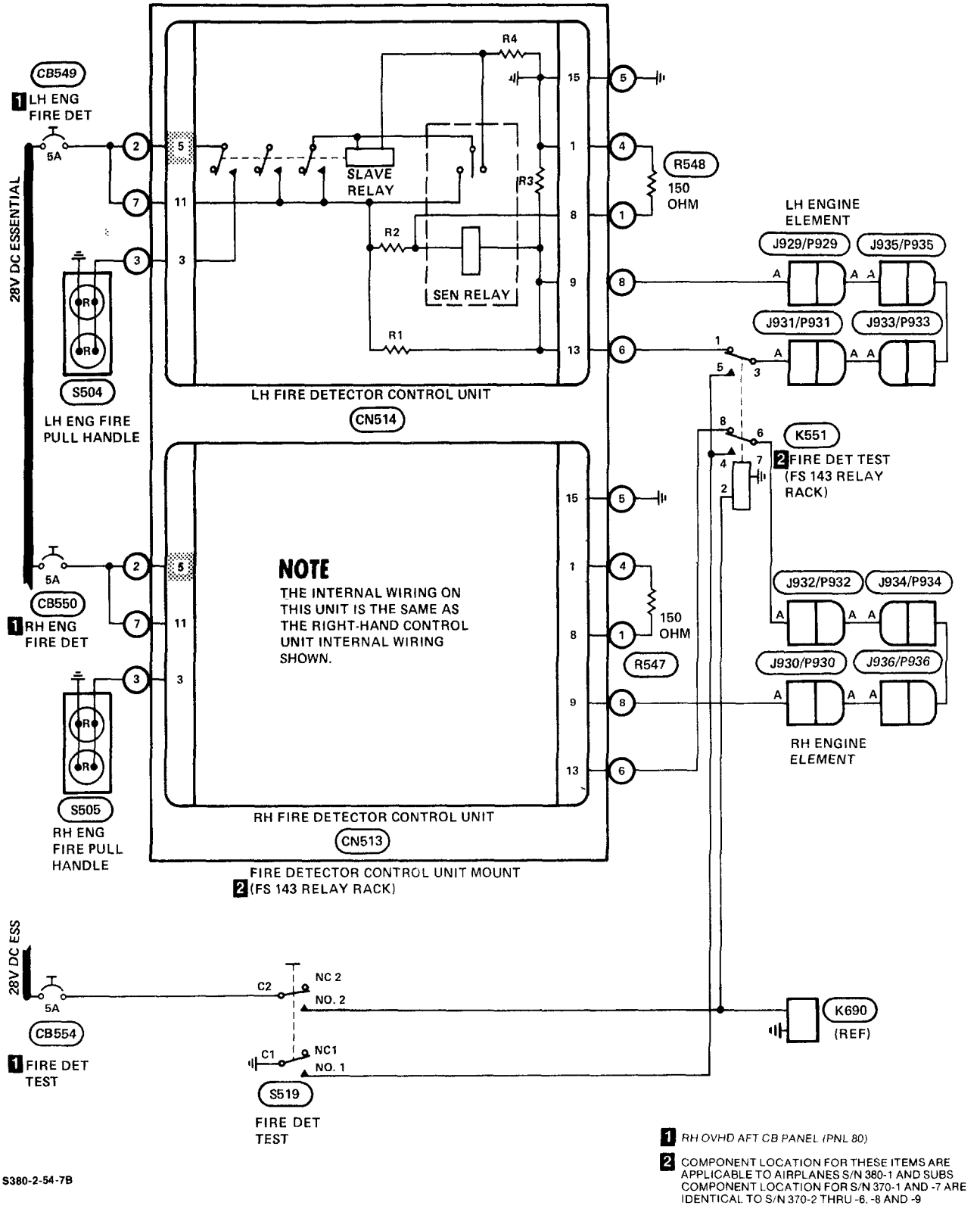
E. Overheat- and Fire-warning Indicator Lights

The warning indicator lights are inside the "FIRE PULL" handles mounted on the windshield bow. Each handle contains lights connected in parallel. The lights come on whenever the associated detection circuit control unit is tripped by an engine overheat or fire condition or when the "FIRE DET SYS PUSH TO TEST" switch is depressed.

ORIGINAL
As Received By
ATP



Engine Overheat and Fire Detection System Schematic - Airplanes Serial No. NA-370-2 Through -6, -8 and -9
Figure 26-10-2



Engine Overheat and Fire Detection System Schematic
S/N 370-1, -7 and S/N 380-1 And Subs
Figure 26-10-2A

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FIRE DETECTION SYSTEM - TROUBLE SHOOTING

1. Trouble Shooting Engine Overheat and Fire Detection System

If the engine overheat and fire detection system does not work, the cause of the trouble must be determined and eliminated before the airplane is flown again. Do not use a torch or soldering iron to test operation of a detector element. If a detector element is not cause of trouble, check circuits in airplane carefully until cause is definitely determined. Do not overlook possibility that an overheat condition may have existed. If any system unit is replaced to remedy a system fault, make a complete operational check-out of system.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. WARNING LIGHT FAILS TO LIGHT UP DURING OPERATIONAL CHECK-OUT.		
(1) Low batteries or electrical power supply off.	Measure voltage applied to warning lights.	Charge batteries or turn on electrical power.
(2) Faulty lamps.	Remove lamps. Apply power to system. Depress "FIRE DET SYS PUSH TO TEST" switch and hold. Check for 28-volt dc between center contacts (for lamps) and ground. If voltage is present, lamps are defective.	Replace faulty lamps.
(3) Defective lamp base.	With power applied to system and "FIRE DET SYS PUSH TO TEST" switch depressed, check for 28-volt dc between base contacts and ground. If voltage is present, base is defective.	Replace defective base.
(4) Faulty test switch.	Turn off and disconnect power from airplane. Make continuity check on "FIRE DET SYS PUSH TO TEST" switch in normal and test positions.	Replace test switch.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
(5) Loose connections or broken wire in system.	Make detailed continuity check of circuit. Visually check system wiring for loose connections or broken wire.	Tighten connections or repair broken wire.
(6) Faulty engine overheat and fire detector control unit.	Make applicable engine nacelle control unit trip resistance test.	Replace faulty engine overheat and fire detector control unit.
(7) Faulty K551 relay. (See figure 26-10-1 or 26-10-1A.)	Disconnect both engine overheat and fire detector control unit plugs. Pull "LH ENG - DET" and "RH ENG - DET" circuit breakers. Push in "DET TEST" circuit breaker. Depress "FIRE DET SYS PUSH TO TEST" switch and hold; check for 28 VDC at K551 relay coil. If voltage is present, K551 relay is faulty.	Replace faulty K551 relay.
B. WARNING LIGHT COMES ON DIMLY DURING TEST.		
(1) One faulty lamp in light.	Remove light cover. Remove lamps. Check each lamp for continuity.	Replace faulty lamp.
C. WARNING LIGHT COMES ON WHEN ENGINE IS NOT RUNNING OR WHILE ENGINE IS RUNNING AT NORMAL TEMPERATURES.		
(1) Broken wire grounding system.	With external electrical power turned off, make visual and continuity check for broken wire.	Repair or replace broken wire.
(2) Defective detector element.	Make applicable detector element total resistance test.	Replace defective detector element.
(3) Defective quick-disconnects.	Check quick-disconnects for moisture or metal chips.	Dry out quick-disconnects or clean out metal chips.
(4) Defective engine overheat and fire detector control unit.	Make applicable engine nacelle control unit trip resistance test.	Replace defective overheat and fire detector control unit.

FIRE DETECTION SYSTEM - MAINTENANCE PRACTICES

WARNING: CARE MUST BE TAKEN WHEN MAINTENANCE IS DONE ON THE ENGINE OVERHEAT AND FIRE DETECTION SYSTEM. IMPROPER HANDLING AND INSTALLATION OF THE DETECTOR ELEMENTS MAY RESULT IN A FALSE WARNING.

NOTE: Do not try to straighten acceptable dents or kinks, because stresses will be set up in the detector element, creating potential failure.

- Any crushed section resulting in a decrease of element diameter to 0.060 inch is cause for rejection.
- Any abrasion of the element 0.005 inch or more deep is cause for rejection.
- Green oxide on surface of detector element must be maintained throughout the element length; however, since the oxide is present for heat transfer purposes only, slight scratches or marring of the oxide can be tolerated.

1. Replacement of Detector Elements - Airplanes NA-370-2 Through -6, -8 and -9

A. Removal

(1) Place all switches in OFF position and disconnect all external power from airplane.

(2) Remove lower or upper engine nacelle door.

(3) Observe routing of detector element on mounting brackets. The replacement detector element must be installed in the same position.

(4) Remove safety wire and disconnect both connections.

(5) Unfasten detector element clamps and remove detector element.

(6) Install dust covers on open connectors.

B. Installation

(1) Place all switches in OFF position and disconnect external electrical power from airplane.

NOTE: All connectors must be clean and dry and have straight center pins before connection. Dust caps should be removed only just before connection is to be made.

(2) Insert female connector into plate, making sure connector hexnut is properly engaged into hex flange to prevent element from twisting.

(3) Thread hex jam nut into connector which projects through plate and torque 40 to 50 inch-pounds (50 inch- pounds maximum). When torquing hex jam nut to female end connectors that are not supported by connector joint plates, use a wrench on end connector hexnut to prevent element from twisting.

(4) Safety-wire jam nut to connector hexnut or to special safety-wire hole provided in plate.

(5) Route detector element from this end to the first element clamp according to figure 26-10-1, taking care not to bend element excessively (1/2 - inch minimum radius) or damage element in any other way.

(6) Pinch clamp open and insert element in clamp. Release clamp. Make sure element is centered in spring clamp.

(7) Proceed as in steps (5) and (6) until element has been routed through all clamps.

NOTE: Minor adjustments may be necessary to permit correct mating of detector element male end fitting.

(8) Mate the two fittings and torque swivel nut from 50 to 70 inch-pounds.

(9) Safety-wire swivel nut to support bracket or to special safety-wire hole provided in structure.

(10) Make a complete operational check of engine overheat and fire detection system.

1A. Replacement of Detector Elements - Airplanes NA-370-1, -7, and NA-380-1 and Subsequent

A. Removal

(1) Place all switches in OFF position and disconnect external power from airplane.

(2) Open engine access doors.

(3) Observe routing of detector element on mounting brackets. The replacement detector element must be installed in the same position.

(4) Disconnect flex leads to pylon and detector element plugs.

CAUTION: ON 380-4 THRU 380-53 AIRPLANES NOT CHANGED BY SSL#39, ENSURE THAT BOTH FITTINGS ARE HELD ON EACH SIDE OF THE PYLON DISCONNECT POINT WHEN REMOVING DETECTOR ELEMENTS.

(5) Remove detector element clamps and remove detector element from engine.

(6) Install dust covers on all open connectors.

B. Installation

(1) Remove electrical power from airplane.

NOTE: Connectors must be clean and dry and have straight pins before connection. Dust caps should be removed just prior to connecting cables.

(2) Route fire detector element as shown in figure 26-10-1A. Maintain 3/8" clearance around all components and make all bends (minimum bend radius 1/2 inch) by hand.

(3) Install detector element clamps. Clamp couplings to engine. Use proper clamps as depicted in the Illustrated Parts Catalog.

(3A) Using a volt/ohm meter (Simpson 260 or equivalent) set to read ohms, connect negative (-) lead to airframe ground and positive (+) lead to fire detector element coupling. Check that meter reads one ohm or less. Remove volt/ohm meter leads.

(4) Install flex leads to pylon and detector elements.

CAUTION: WHEN TORQUING CONNECTORS TO DETECTORS, USE A WRENCH ON CONNECTOR FLATS TO PREVENT DETECTOR CABLE FROM TWISTING.

- ON 380-4 THRU 380-53 AIRPLANES NOT CHANGED BY SSL#39, ENSURE THAT BOTH FITTINGS ARE HELD ON EACH SIDE OF THE PYLON DISCONNECT POINT WHEN REMOVING DETECTOR ELEMENTS.

(5) Perform an operational check-out of engine overheat and fire detection system.

2. Replacement of Control Units

A. Removal

(1) Place all airplane switches at OFF and disconnect all external electrical power from airplane.

(2) Break safety wire that locks knurled to retainer bracket on control unit.

(3) Loosen and swing knurled nut outward and downward, until nut clears retainer bracket.

(4) Slide control unit inboard on mounting bracket, until connector on back of control unit becomes unplugged from receptacle on mounting bracket.

(5) Remove control unit from mounting bracket.

B. Installation

(1) Place all airplane switches at OFF and disconnect all external electrical power from airplane.

(2) Place control unit on mounting bracket and align pins of connector with receptacle on bracket.

(3) Slide control unit toward mounting bracket until control unit connector pins are fully engaged with receptacle. Apply pressure to both control unit and back of mounting bracket to ensure proper mating of connector with receptacle.

(4) Swing knurled nut upward into position over retainer bracket, and finger-tighten retainer nut.

(5) Safety-wire knurled nut to retainer bracket on control unit.

(6) Make an operational check-out of system.

WARNING: ON AIRPLANES NA-370-2 THROUGH -6, -8 AND -9, BE SURE TO RECONNECT FIRE WARNING SYSTEM AFTER REMOVAL AND REINSTALLATION OF UPPER AND LOWER ENGINE COWLING.

1. Testing Fire Detection System

NOTE: Make checks in the order given. When a result does not conform to the procedure, make corrections before going on. If the check does not meet requirements, refer to 26-10-1.

A. Operational Check-out of Fire Detection System

(1) Place all switches at OFF position.

(2) Place battery and electrical master switches at ON.

(3) Push in "DET TEST," "RH ENG - DET," and "LH ENG - DET" circuit breakers.

(4) Press "FIRE DET SYS PUSH TO TEST" switch. Right and left "FIRE PULL" overheat and fire-warning indicator lights should illuminate within one second after switch is pushed. Make sure both lights in each assembly illuminate. Release test switch. Both lights should go out within 5 seconds.

(5) Place battery and electrical master switches to OFF.

B. Left or Right Engine Nacelle Detector Element Total Resistance Test

CAUTION: DO NOT ATTEMPT TO PERFORM THIS TEST IF ENGINE NACELLE AMBIENT TEMPERATURE OR TEMPERATURE OF ANY DETECTOR ELEMENT EXCEEDS 100°F. FAILURE TO OBSERVE THIS CAUTION WILL RESULT IN PERMANENT DAMAGE TO DETECTOR ELEMENT.

DO NOT ATTEMPT TO PERFORM THIS TEST DURING AN ENGINE RUN-UP OR IMMEDIATELY AFTER AN ENGINE SHUTDOWN, BECAUSE ENGINE NACELLE AMBIENT TEMPERATURE WILL HAVE EXCEEDED 100°F.

(1) Engine nacelle detector element resistance test - NA-380 airplanes

(a) Disconnect airplane wiring harness from control unit mounting bracket terminals "6" and "8".

(b) Connect the two disconnected wires together.

(c) Connect one lead of a multimeter to the two connected wires.

(d) Connect other lead of multimeter to airplane structure. Select a clean metal surface which is not anodized or painted.

(e) Check resistance with multimeter. The resistance should not be less than 500,000 ohms.

(f) Disconnect multimeter leads from airplane wiring harness.

(g) Reconnect airplane wiring harness to control unit mounting bracket terminals "6" and "8".

(h) Open RH and LH engine cowling. Using a volt/ohm meter (Simpson 260 or equivalent) set to read ohms, connect negative (-) lead to airframe ground and positive (+) lead to fire detector element coupling. Check that meter reads one ohm or less. Remove volt/ohm meter leads.

NOTE: If resistance in step (h) is greater than specified, check coupling bracket that attaches to engine for corrosion or any substance that could cause a faulty ground condition.

(i) On RH engine, disconnect fire detector element from one coupling. Apply external power to airplane.

(j) Using a suitable jumper wire, attach one end of jumper wire to inner conductor of receptacle, and the other end of jumper wire to ground. Check that the right hand "FIRE PULL" handle lights illuminate. Remove jumper from receptacle inner conductor and insert in center conductor of fire detector element. Check that right hand "FIRE PULL" handle lights illuminate. Remove jumper wire.

(k) Connect and tighten fire detector element to coupling element disconnected in step (i). Lockwire Coupling.

(l) On LH engine, disconnect fire detector element from one coupling.

(m) Using a suitable jumper wire, attach one end of jumper wire to inner conductor of receptacle, and the other end of jumper wire to ground. Check that the left hand "FIRE PULL" handle lights illuminate. Remove jumper from receptacle inner conductor and insert in center conductor of fire detector element. Check that left hand "FIRE PULL" handle lights illuminate. Remove jumper wire.

(n) Connect and tighten fire detector element to coupling (element disconnected in step (l).

(o) Repeat steps (a) through (n) for other control unit.

(p) Close engine cowling and remove electrical power from airplane. Make an operational check-out of system. (Refer to paragraph A.)

EDISON SYSTEM COMPONENTS	PART NO.	VOLTAGE BREAKDOWN TEST (1000V DC)	TOTAL RES. COND TO GRD MIN OHMS		CENTER WIRE RES. MAX OHMS
			72°F	100°F	
Detector element	244C54G132		250K	125K	6.60
Detector element	244C54G108		250K	125K	5.40
Detector element	244C54G60		250K	125K	3.00
Male connector	42232				0.05
Female connector	42233				0.05
Elbow	41493				0.05
Plug assembly	872195*	Must pass			0.05
Socket assembly	872196*	Must pass			0.05
Wiring	RSS-101-16				MIL-W-25038

*Walter Kidde part number.

(2) Engine nacelle detector element resistance test, NA-370 airplanes with APU installed.

(a) Electrical power - OFF.

(b) Open fire detector access door on nacelle. (See figure 26-10-1).

(c) Disconnect aft fire detector disconnect through access door.

(d) Connect multimeter lead to plug and the other lead to receptacle (mounted on bracket) at disconnected quick-disconnect.

(e) Record resistance reading (maximum 25 ohms).

(f) Disconnect multimeter lead at plug and connect to ground. (Clean metal surface which is not anodized or painted on engine or pylon structure).

(g) Open lower nacelle door and disconnect forward fire detector quick-disconnect on door bracket. (See figure 26-10-1).

(h) Record insulation resistance reading of fire detector elements. (Minimum 500K at 100°F or under).

NOTE: If total resistance is less than that stated in step (h), make an individual resistance test of each component and wiring. (Refer to the table for individual resistance values.) Replace faulty component or wiring, and repeat test procedure.

(i) Disconnect multimeter leads.

(j) Connect both connectors and close access door and lower nacelle door.

(k) Make operational check-out of system. (Refer to paragraph A.)

C. Left or Right Engine Nacelle Control Unit Trip Resistance Test - Airplanes Serial No. NA-370-2 Through -6, -8 and -9

NOTE: The following test requires a resistance decade box and a 28-volt dc external power source.

(1) Disconnect all external power from airplane.

(2) Disconnect plug P663 from aft fuselage relay box. (See figure 26-10-1.)

(3) Connect one lead of decade box to pin "A" of P663 for left nacelle test.

(4) Connect other lead of decade box to a convenient ground (airplane structure which is clean, and not anodized or painted).

(5) Push in "DET TEST", "RH ENG - DET", and "LH ENG - DET" circuit breakers.

(6) Connect a 28-volt dc external power source to airplane.

(7) Adjust decade box to approximately 400 ohms.

(8) Slowly decrease decade box resistance (by small amounts) until left overheat and fire-warning light inside "T" handle on windshield bow comes on. The light should come on when the decade box resistance is between 180 and 220 ohms.

(9) Slowly increase decade box resistance until warning light goes out. The light should go out when the decade box resistance is at or below 300 ohms.

NOTE: If warning light does not go out within limits specified in step (9), check resistance of external resistor connected to mounting bracket terminals "1" and "4". The resistor should measure 200 ohms (±1 percent). If resistor is not correct, remove and replace resistor; then repeat test. If resistor value is correct, remove and replace control unit. Repeat test.

(10) Disconnect decade box lead from pin "A" of P663 and reconnect lead to pin "C" for right nacelle test.

(11) Repeat steps (7) through (9).

(12) Disconnect decade box from P663 and ground.

(13) Disconnect external power from airplane.

(14) Reconnect plug P663 to aft fuselage relay box. Safety-wire plug.

(15) Make an operational check-out of system. (Refer to paragraph A.)

D. Left or Right Engine Control Unit Trip Resistance Test - Airplanes Serial No. NA-370-1, -7 and NA-380-1 and Later

NOTE: The following test requires a resistance decade box and a 28-volt dc external power source.

(1) Remove all electrical power from airplane.

(2) Disconnect airplane wiring harness to control unit mounting bracket terminal "6".

(3) Connect leads of decade box between terminal "6" and a clean ground.

(4) Apply 28 volts dc electrical power to airplane.

(5) Adjust decade box to approximately 400 ohms.

(6) Slowly decrease decade box resistance until fire-warning light inside "T" handle comes on. The light should come on when the decade box resistance is between 135 and 225 ohms.

(7) Slowly increase decade box resistance until warning light goes out. The light should go out when the decade box resistance is at or below 225 ohms.

NOTE: If warning light operation is not within limits, measure resistance of R547 or R548 external resistor. The resistor should measure 150 ohms (±1 percent). If value of resistor is not correct, remove and replace resistor; then, repeat test. If value of resistor is correct, replace control unit and repeat test.

(8) Perform test for opposite engine if required.

(9) Remove electrical power from airplane.

(10) Disconnect decade box and connect airplane wiring harness to terminal "6" of control unit mount bracket.

(11) Perform an operational check-out of system. (Refer to paragraph A.)

FIRE-EXTINGUISHING SYSTEM - DESCRIPTION AND OPERATION

1. Description and Operation

A. General

A high-rate-of-discharge fixed fire extinguishing system is provided for each engine pod. Two pressurized extinguishing-agent containers provide monobromotrifluoromethane, which can be directed from each container to either engine pod as required. The discharge lines from each agent container meet at a double-check "T" valve to make a single discharge line. (See figure 26-20-1 or 26-20-1A.) The one-way check valve prevents one agent container discharging into the other container line and, thereby possibly depleting the contents of one container into the other. For airplanes equipped with APU, refer to Chapter 49.

The system is controlled from the cockpit by the use of manual handles plus an electrical selector switch. Two discharge indicator disks, mounted externally on the fuselage, indicate either manual discharge of the agent containers, or automatic discharge overboard due to a thermal condition. A pressure gage, mounted on each container, and visible from the main wheel well, indicates pressure in the container.

A discharge valve is attached to the bottom of each agent container. The cartridge in the extinguisher No. 1 or extinguisher No. 2 discharge valve is ignited by 28-volt dc power when a "FIRE PULL" handle is pulled and the fire-extinguisher selector is actuated to EXT NO. 1 or EXT NO. 2. A direction valve then routes the extinguishing agent to the proper engine, according to which "FIRE PULL" handle is pulled. When a cartridge is fired, the contents of one agent container are discharged by nitrogen pressure and forced through delivery lines and nozzles into the selected engine pod. (See figure 26-20-1 or 26-20-1A.)

NOTE: On airplanes NA-370-2 through -6, -8 and -9, the sealed fire wall, between the forward and aft compartments, does not allow any extinguishing agent into the aft compartment. Since no combustible fluid or material is contained in the aft compartment, an extinguishing agent is unnecessary.

B. Engine "FIRE PULL" Handles

Two mechanically interconnected "FIRE PULL" handles, one for each engine, are on the fire-extinguisher control panel, on the windshield bow. During a fire or overheat condition, indicated by a warning light in the respective "FIRE PULL" handle, immediately pull handle aft. When the right-hand "FIRE PULL" handle is pulled, the emergency bleed-air shutoff valve closes, and the extinguisher system direction valve is energized to allow the extinguishing agent to be routed to the right-hand engine. In addition, the dc generator is taken off the line. When the "FIRE PULL" handle is returned to its original position, those items previously turned off, except the dc generator, are reinstated, unless the engine master switch is turned off. The dc generator is reinstated by placing the generator switch to RESET, then to ON* or at ON†. If the second "FIRE PULL" handle is pulled, the first handle automatically retracts to its original position.

*NA-370-2 through -6, -8 and -9
†NA-370-1, -7, NA-380-1 and Later

C. Fire-extinguishing Selector Switch

The fire-extinguisher selector switch, mounted in the center of the extinguisher control panel and powered by the dc essential bus, has three positions: EXT NO. 1, EXT NO. 2, and a center, unmarked, off position. When the selector switch is momentarily positioned to either EXT NO. 1 or EXT NO. 2 and has been armed by pulling the "FIRE PULL" handle, the extinguishing agent from one agent container is discharged to the engine pod selected by the handle. After the initial agent container has been expended, the fire-extinguisher selector switch may, if necessary, be momentarily positioned to the other extinguisher to discharge the second agent container.

D. Agent Containers

Two agent containers for the fire-extinguisher system are installed aft of the main wheel well area between fuselage stations 298 and 307. (See figure 26-20-1 or 26-20-1A.) Each agent container has a gage which indicates the pressure in the container. A discharge valve containing a cartridge is mounted on the lower part of each agent container. The cartridge, when fired, discharges the container contents into the ducting toward the engine pods. A fitting is provided in each container to accommodate lines to the thermal discharge indicators installed in the fuselage exterior.

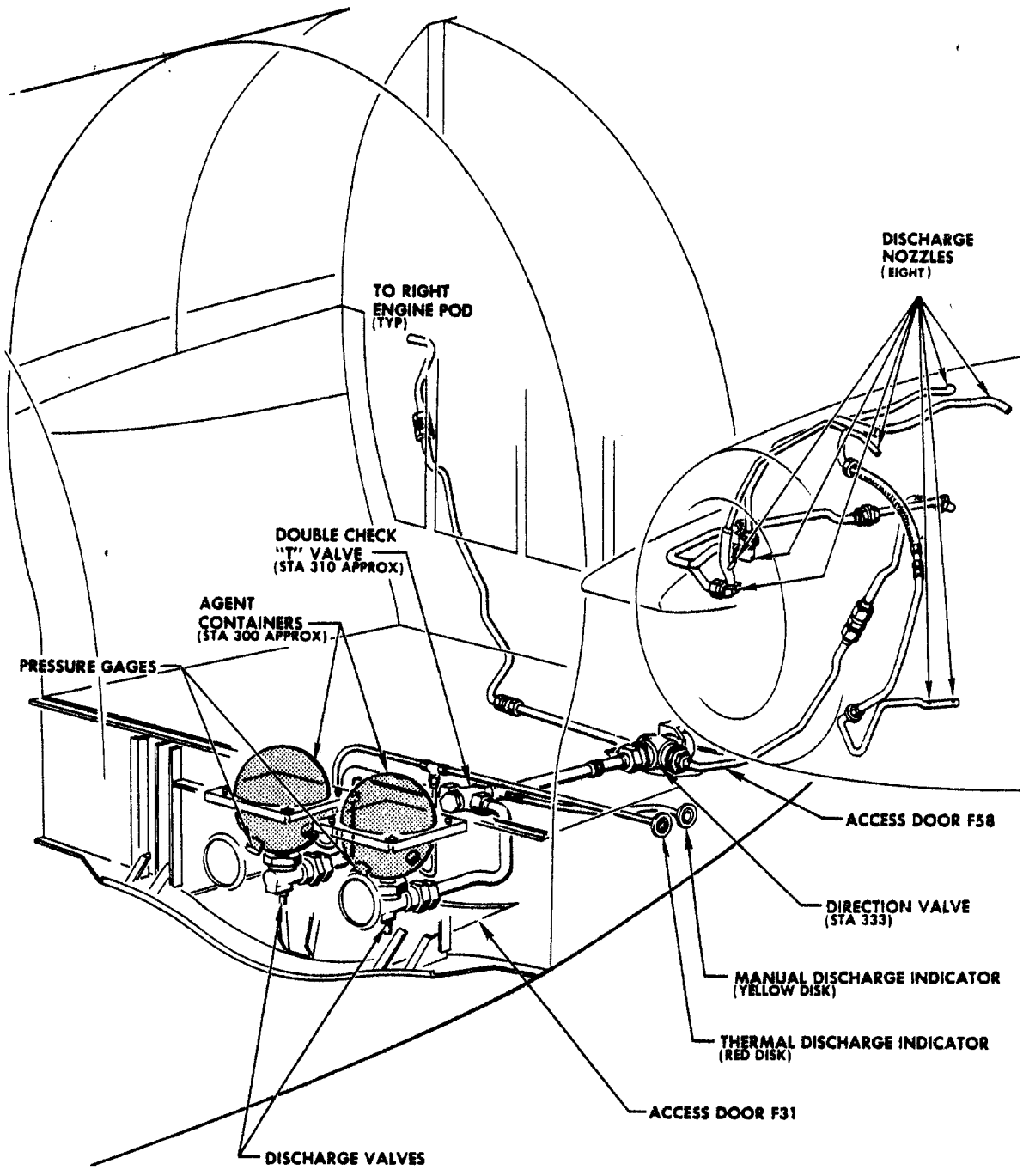
E. Direction Valve

Downstream from the double-check "T" valve, the line connects to a direction valve. (See figure 26-20-1 or 26-20-1A.) The valve has two exit ports: a normally open port connected with the left engine pod fire-extinguishing discharge line, and a normally closed port connected with the discharge line to the right engine pod. When the right-hand engine "FIRE PULL" handle is pulled, an electrical circuit is completed to the direction valve, energizing the solenoid. The fire-extinguishing agent discharge is then directed to the right-hand engine pod.

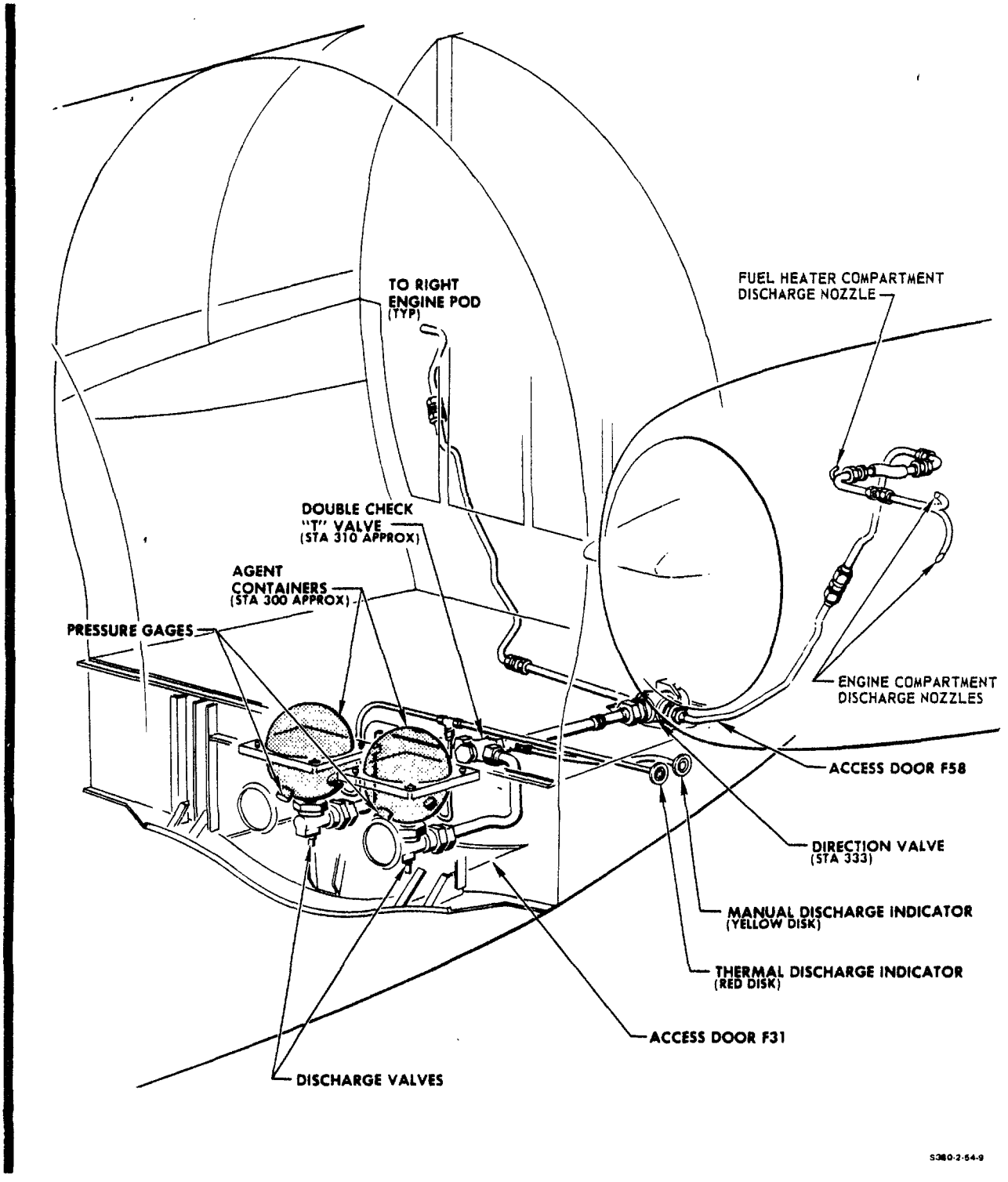
F. Discharge Indicators

Two fire-extinghishing system discharge indicating disks are mounted on the left side of the fuselage, aft of the wing. The yellow disk in the aft discharge indicator is connected by a 1/4-inch line to the fire-extinguisher discharge line between the double-check "T" valve and the direction valve. When either container is discharged, a limited flow will be directed to the yellow disk, blowing it out. A later check of the agent container pressure gages will show whether one or both containers have been discharged.

The red disk in the forward discharge indicator is connected by a 1/4-inch line to both agent containers. When the agent containers have been overheated excessively, the internal pressure will cause the fusible safety outlet plug to discharge. The agent flow will be directed to the red disk, blowing it out. A check of the agent container pressure gages will show whether one or both containers were discharged.

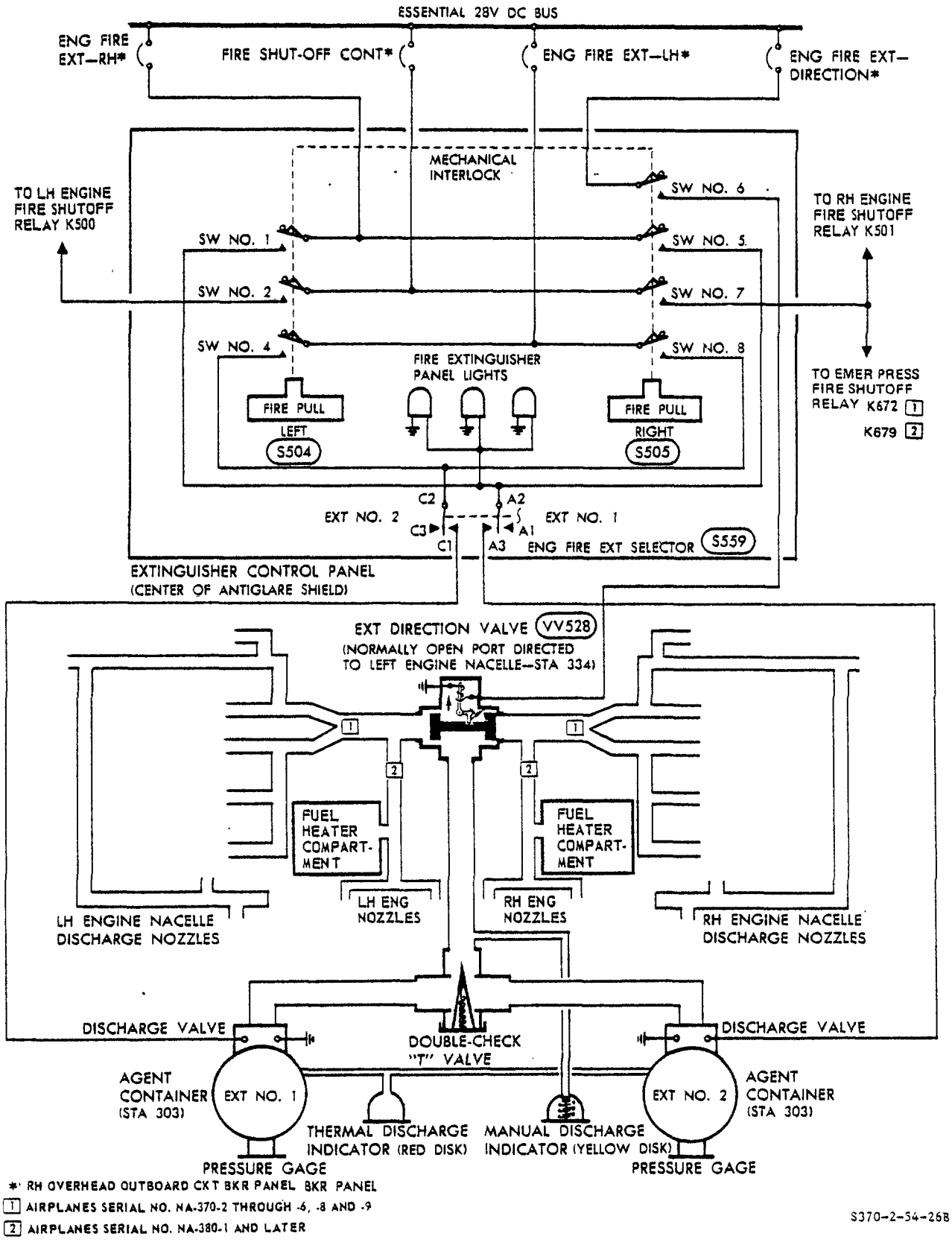


Engine Fire-extinguishing System - NA-370-2 Through -6, -8 and -9
Figure 26-20-1



Engine Fire-extinguishing System - NA-370-1, -7, and NA-380-1 and Later

Figure 26-20-1A



Engine Fire-extinguishing System Electrical Schematic
Figure 26-20-2

FIRE-EXTINGUISHING SYSTEM - MAINTENANCE PRACTICES

1. Servicing Fire-extinguishing System

WARNING: MONOBROMOTRIFLUOROMETHANE (CF_3Br) IS NONTOXIC AND CLASSED AS NON-POISONOUS. HOWEVER, IT CAN BE HARMFUL AND SHOULD BE HANDLED WITH CARE. DO NOT BREATHE VAPOR OR PERMIT LIQUID TO COME IN CONTACT WITH SKIN OR CLOTHING. IF LEAKAGE IN A CLOSED SPACE IS SUSPECTED, THIS SPACE SHOULD NOT BE ENTERED UNTIL ALL VAPOR HAS BEEN DISSIPATED AND THE AREA IS WELL VENTILATED. USE A HALIDE DETECTOR (PRES-O-LITE, OR EQUIVALENT) TO ENSURE THAT AREA IS SAFE.

A. To service engine fire-extinguishing system, proceed as follows:

(1) Remove discharged agent containers and discharge valves as soon as possible after containers have been discharged.

(2) With dry compressed air or nitrogen, purge fire-extinguishing lines.

(3) Install serviceable agent containers and discharge valves.

(4) Replace indicator disks as required.

REMOVAL AND INSTALLATION

REPLACEMENT OF AGENT CONTAINERS

CAUTION

Do not drop pressurized containers.

Removal:

1. Remove access aft of main wheel wells.
2. Disconnect electrical wires from discharge valve and immediately connect a shorting wire between cartridge positive terminal and ground stud (to prevent detonation by static electricity).
3. Disconnect system discharge line and thermal discharge indicator line.
4. Remove four mounting bolts and washers.
5. Remove agent container from airplane.
6. Cap all open lines and cover electrical wire lugs.

Installation:

1. Weigh container (along with its discharge valve and cartridge) to ensure that weight agrees with total weight stenciled on decal attached to container within ±0.25 pound. The total weight of the container assembly should be approximately 14 pounds.

WARNING

Do not attempt to remove fusible safety outlet plug from agent container. Removal can cause injury due to high pressure in container.

2. On Kidde agent containers, remove (shipping) thread protection from container safety port. Apply Permatex No. 1 to threads of thermal discharge connector P/N 800036 and install connector on container. Torque to 50 inch-pounds.
3. Place agent container in position in airplane and install four mounting bolts with two washers for each mounting bolt.
4. Connect system discharge line and thermal discharge indicator line. On HTL Ind. container, valve may be rotated to align with discharge line. Do not loosen fitting that secures valve to container. After rotating, safety wire cartridge to 1¼-inch hex nut.
5. Remove shorting wire, touch positive lead to airframe ground, then connect electrical wires to discharge valve.
6. Install access.

REPLACEMENT OF DISCHARGE VALVE — KIDDE BOTTLE INSTALLATION

Removal:

WARNING

Take care that no electrical power is applied at discharge valve terminal when valve is removed from agent container. The cartridge in the discharge valve contains a metal slug that is fired with lethal force.

WARNING

Do not attempt to remove fusible safety outlet plug from agent container. Removal of fusible safety outlet plug can cause injury to personnel due to high pressure in container.

1. Remove fire extinguisher agent container (with discharge valve) Refer to Replacement of Agent Containers.
2. Note positioning of valve assembly and mark for installation in same position.
3. Using strap wrench remove discharge valve from container by loosening swivel nut on valve body.

Installation:

1. Apply grease (Specification MIL-G-7711A or 994 Regal Starfak Premium 2, a Texaco product) to cartridge threads, and install gasket, contact, strainer, and cartridge in discharge valve assembly. Torque cartridge to 150-200 inch-pounds.

WARNING

Take care that no electrical power is applied at discharge valve terminal when valve is removed from agent container. The cartridge in the discharge valve contains a metal slug that is fired with lethal force.

2. Assemble discharge valve assembly to container as follows:
   A. Check that O-ring packing and disk retainer are in place.
   B. Apply small amount of grease (Specification MIL-G-7711A or 994 Regal Starfak Premium 2, a Texaco product) to container neck threads.
   C. Position valve and align with mark made during removal. (Valve should point 180° from pressure gauge to align discharge valve fitting to discharge line.) Tighten swivel nut with a strap wrench to a torque of approximately 75 foot-pounds.

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REMOVAL AND INSTALLATION (Continued)

Installation (Cont'd)

3. Install container in airplane (refer to Replacement of Agent Containers).

REPLACEMENT OF DIRECTION VALVE

Removal:

1. Remove access 2. (Refer to chapter 6, Figure 6-00-3 or 6-00-4.)
2. Disconnect electrical wires from valve.
3. Disconnect engine fire-extinguisher discharge lines from valve. Remove valve.
4. Cap all open lines and cover electrical wire lugs.

Installation:

1. Position direction valve so that normally open port connects to line from left engine pod. Connect engine fire-extinguisher discharge lines to direction valve.
2. Connect electrical wires to direction valve.
3. Perform system obstruction and purge check of direction valve.
4. Install access.

REPLACEMENT OF FIRE-EXTINGUISHING DISCHARGE INDICATOR DISKS

NOTE

The fire-extinguishing discharge indicator disks are not normally removed. If the thermal discharge indicator or the manual discharge indicator has been removed, it usually indicates discharge of an agent container. The agent containers should be checked and, if depleted, replaced and the cause of the discharge corrected before the indicator disk or disks are replaced.

Replacement of Thermal Discharge Indicator Disk (Red)

1. Place red disk in position in thermal discharge indicator body.
2. Install retaining ring.

Replacement of Manual Discharge Indicator Disk (Yellow)

1. Place O-ring packing in position in manual discharge indicator body.
2. Install piston against packing.
3. Place spring on stem of piston.
4. Depress spring and install washer. Hold in place with lock ring.
5. Place yellow disk in position against lock ring and install retaining ring.

REPLACEMENT OF FIRE-EXTINGUISHING CARTRIDGES — KIDDE BOTTLE INSTALLATION

Removal:

WARNING

Take care that no electrical power is applied at discharge valve terminal when valve is removed from agent container. The cartridge in the discharge valve contains a metal slug that is fired with lethal force.

WARNING

Do not attempt to remove fusible safety outlet plug from agent container. Removal of fusible safety outlet plug can cause injury to personnel due to high pressure in container.

1. Remove fire extinguisher agent container (with discharge valve). Refer to Replacement of Agent Containers.
2. Note position of valve assembly and mark for reinstallation in same position.
3. Using a strap wrench, remove discharge valve from container by loosening swivel nut on valve body.
4. Inspect interior of discharge valve for corrosion.
5. Remove and safely dispose of old discharge cartridge.
6. Remove cartridge contact and inspect for corrosion. Discard gasket.

Installation:

1. Apply grease (Specification MIL-G-7711A or 994 Regal Starfak Premium 2, a Texaco product) to threads of new cartridge, and install gasket, contact, strainer, and new cartridge in discharge valve assembly. Torque cartridge to 150-200 inch-pounds.

WARNING

Take care that no electrical power is applied at discharge valve terminal. The cartridge contains a metal slug that is fired with lethal force.

2. Assemble discharge valve assembly to container as follows:
   A. Check that O-ring packing and disk retainer are in place.

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REMOVAL AND INSTALLATION (Continued)

Installation (Cont'd)

B. Apply small amount of grease (Specification MIL-G-7711A or 994 Regal Starfak Premium 2, a Texaco product) to container neck threads.

C. Position valve and align with mark made during removal. Tighten swivel nut with a strap wrench to a torque of 75 foot-pounds.

3. Install agent container. Refer to Replacement of Agent Containers.

REPLACEMENT OF FIRE-EXTINGUISHING CARTRIDGES — HTL INDUSTRIES INC. BOTTLE INSTALLATION

WARNING

To prevent possible injury to personnel, ensure that no electrical power is applied at cartridge electrical (positive) terminal.

WARNING

To prevent discharge of bottle, do not remove discharge valve securing nut.

Removal:

1. Remove access aft of main wheel wells. Refer to chapter 6.
2. Remove positive ground (-) electrical wires from cartridge and immediately connect a shorting wire between positive terminal and ground stud (to prevent detonation by static electricity).
3. Remove lockwire from cartridge.
4. Using two wrenches (one to remove cartridge and one to hold discharge valve securing nut), remove and safely dispose of old cartridge.

Installation:

1. Install O-ring seal (AN6290-5) on new cartridge; then, using two wrenches (one to hold discharge valve securing nut and one to install replacement cartridge) install cartridge and torque to 90-100 inch-pounds. Safety wire cartridge to 1¼-inch hex nut.

CAUTION

To prevent possible discharge of bottle, touch positive lead to airframe ground before connecting to cartridge.

2. Remove shorting wire and connect positive ground (-) electrical leads to cartridge.
3. Check completed installation for security. Install access door.

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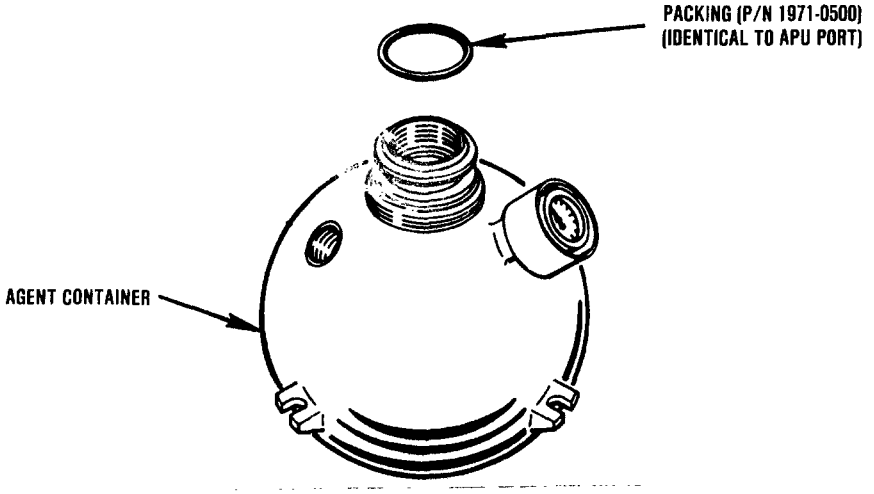

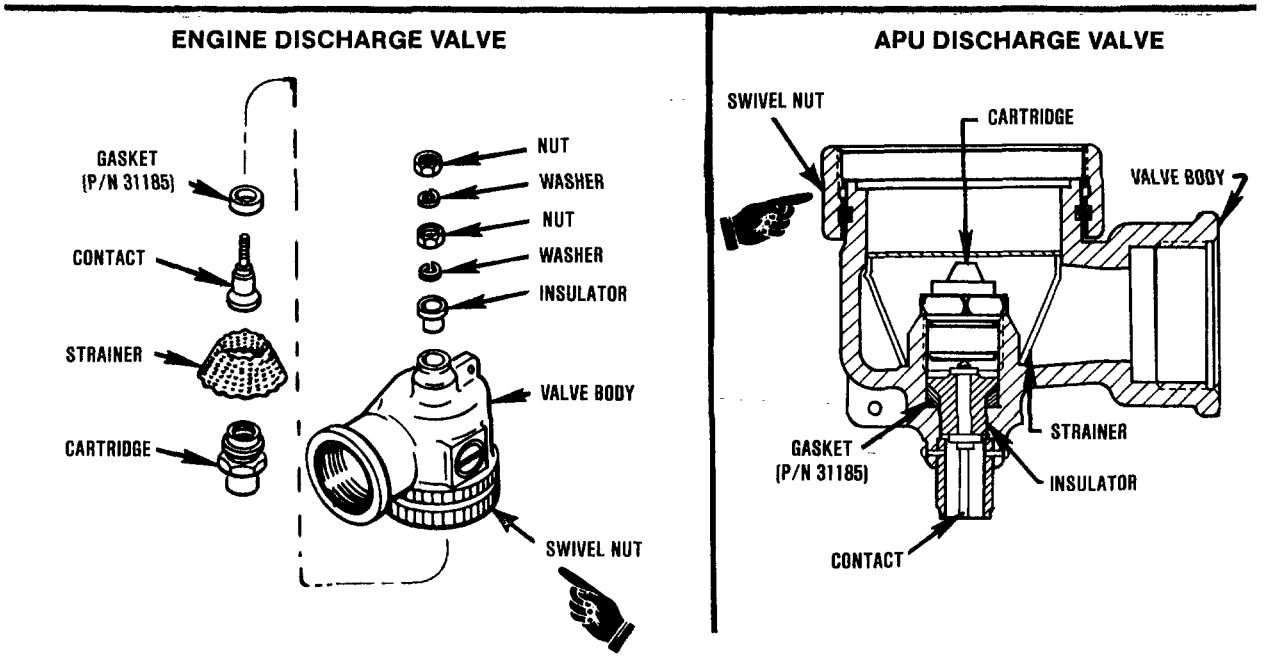

Figure 26-20-5 — Fire-extinguishing Cartridge — Kidde Bottle Installation

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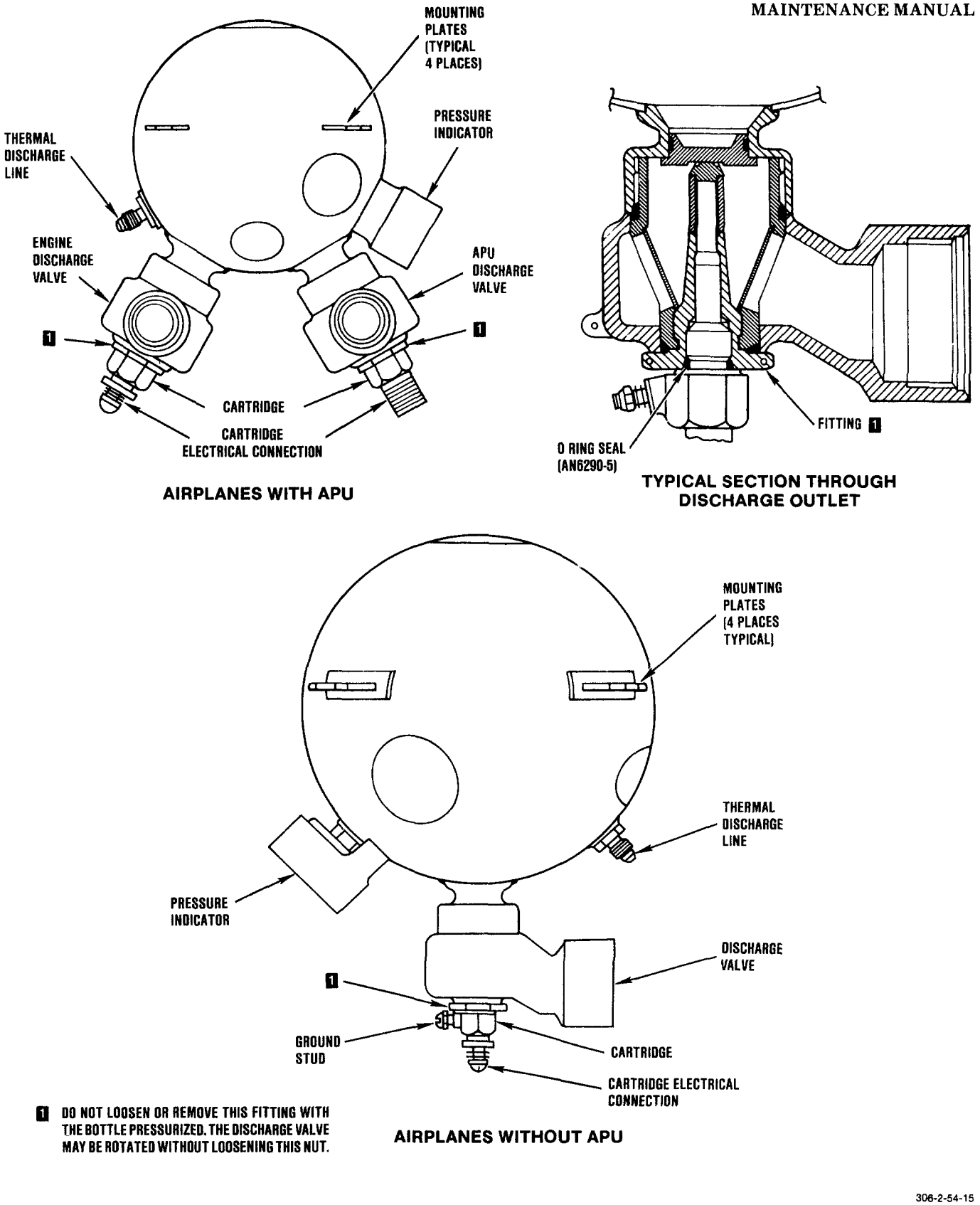

Figure 26-20-6 — Fire-extinguishing Cartridge — HTL Industries Inc. Bottle Installation

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MAINTENANCE PRACTICES (Continued)

CHECK-OUT OF FIRE-EXTINGUISHING SYSTEM

Fire-extinguishing System Electrical Circuit Functional Test

NOTE

Test lamp, GE P/N 1819 or Walter Kidde P/N 206478 (or equivalent) lamp is required for this procedure. The test lamp is required to limit cartridge testing current to 0.035 amperes. This prevents cartridge from firing.

1. Remove the RH and LH engine bleed air valve access doors in lower skins of engine pylons, and the emergency bleed air valve pressure switch, direction valve and extinguisher container access doors on the lower portion of the fuselage. Refer to chapter 6 for access panel location.
2. Disconnect electrical plug from emergency bleed air valve pressure switch.

CAUTION

Ensure that disconnected lead from cartridge terminal does not touch cartridge stud or airframe.

3. Disconnect positive electrical lead at discharge valve of each extinguisher agent container.
4. Pull out the following circuit breakers:
   LH Ovhd Outbd — BOOST PUMP — LH FUEL
   BOOST PUMP — RH FUEL
   LH ENGINE — IGN & BOOST
   RH ENGINE — IGN & BOOST
   Co-Pilot's Console — HYD PUMP CONT

   External 28 VDC electrical power — ON.
5. Push in the following circuit breakers:
   LH Ovhd Outbd — LH ENGINE — MASTER
   RH ENGINE — MASTER
   RH Ovhd Outbd — ALL FIRE CONTROL & WARNING SYSTEMS circuit breakers
6. Set LH — ENGINE MASTER switch at ON. Check for momentary illumination of FUEL SHUT OFF FAIL indicator light.
7. Set RH — ENGINE MASTER switch at ON. Check for momentary illumination of FUEL SHUT OFF FAIL indicator light.
8. Set CABIN AIR rotary switch at BOTH ENG. Check that indicator pin on solenoid of LH and RH engine bleed air valve is out (protruding above top surface of solenoid approximately 1/32 inches).
9. Pull left FIRE PULL handle. Check the following:
   FUEL SHUT OFF FAIL indicator light — momentarily illuminates
   Fire Emergency Control Panel light — illuminated
   Extinguisher System Direction Valve — open to left engine (indicator points to right engine)
   LH engine bleed air valve — indicator pin in (flush with top of solenoid)
   RH engine bleed air valve — indicator pin out

   NOTE

   Extinguisher system direction valve is normally open to left engine. Valve does not move when LH FIRE PULL handle is pulled. Indicator on valve points to blocked engine.
10. Pull right FIRE PULL handle. Check the following:
    Left FIRE PULL — automatically returns to its normal position
    FUEL SHUT OFF FAIL indicator light — momentarily illuminates
    Fire Emergency Control Panel lighting — illuminated
    Extinguisher System Direction Valve — open to right engine (indicator points to left engine)
    Left engine bleed air valve — indicator pin out
    Right engine bleed air valve — indicator pin in
11. Set CABIN AIR rotary switch at EMERG PRESS. Check for 0 (zero) VDC at pin B of emergency bleed air valve overpressure switch electrical connector plug (P808)
12. Return right FIRE PULL handle to its normal position. Check the following:
    FUEL SHUT OFF FAIL indicator light — momentarily illuminates
    Fire Emergency Control Panel lighting — out
    Emergency bleed air valve overpressure switch connector plug — 28 VDC at pin B
13. Set CABIN AIR rotary switch at OFF.
14. Connect test lamp leads between LH cartridge and disconnected positive lead.
15. Pull left FIRE PULL handle. Momentarily set extinguisher selector switch to EXT NO 1. Test lamp should momentarily illuminate.
16. Pull right FIRE PULL handle. Momentarily set extinguisher selector switch to EXT NO 1. Test lamp should illuminate.
17. Set extinguisher selector switch at EXT NO 2. Check that test lamp goes out.
18. Disconnect test lamp leads from cartridge and disconnected positive lead.
19. Connect test lamp leads between RH cartridge and disconnected positive lead.

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MAINTENANCE PRACTICES (Continued)

CHECK-OUT OF FIRE-EXTINGUISHING SYSTEM (Cont'd)

20. Momentarily set extinguisher selector switch at EXT NO 2. Test lamp should illuminate.
21. Pull LH FIRE PULL handle. Momentarily set extinguisher selector switch at EXT NO 2. Test lamp should momentarily illuminate.
22. Return LH FIRE PULL handle to normal position.
23. Set LH and RH ENGINE MASTER switches at OFF.
24. Push in the LH and RH ENGINE IGN & BOOST CONT, LH and RH FUEL-BOOST PUMP and HYD PUMP CONT circuit breakers.
25. Electrical power — OFF.
26. Disconnect test lamp leads.

CAUTION

To prevent accidental firing due to stray voltage, momentarily short cartridge leads to airplane structure before connecting to cartridges.

27. Connect disconnected cartridge leads to LH and RH cartridges.
28. Connect and safety emergency bleed air valve overpressure switch electrical connector plug (P808).
29. On airplanes equipped with APU, refer to chapter 49 for electrical circuit functional test.
30. Install and/or close access panels.

Fire-extinguishing System Obstruction and Purge Check

The following checks must be made to ensure that lines, fittings, and valves are unobstructed, and that components are installed correctly and operate properly.

1. Disconnect electrical leads to system discharge valves at agent containers.
2. Disconnect thermal discharge line at right and left agent container connectors.
3. Remove thermal indicator disk (red) and apply 10 psi air pressure at disk port.
4. Check for airflow at disconnected ends of thermal discharge lines at right and left containers.

NOTE

Airflow indicates that the lines are unobstructed.

5. Shut off air pressure supply, reconnect thermal discharge lines to agent containers, and torque fittings.
6. Apply 100 psi air pressure at thermal indicator disk port and check all connections of thermal discharge line for leakage. No leakage is allowed.
7. Shut off air supply and remove air source from port. Reinstall thermal indicator disk (red).
8. Disconnect manual discharge line at boss on main line just aft of the double check T fitting.
9. Connect an air supply to the boss and increase air pressure to 10 psi. On S/N 370-2 thru -6, -8 and -9, check the eight discharge nozzles in left engine nacelle for airflow.
10. Apply electrical power to airplane.

WARNING

Make sure that electrical leads to both left and right discharge valves on the agent containers are disconnected.

11. Pull out right FIRE PULL handle. Check the discharge nozzles in right engine nacelle for airflow.
12. Shut off airflow. Leave air line connected to boss.
13. On S/N 370-2 thru -6, -8 and -9, disconnect left and right nacelle upper discharge line at union just aft of nacelle canted station 90. This is the union on the line which connects to the Y for the nozzles. Cap or plug both union fittings. On S/N 370-1, -7 and 380-1 and subs, remove left and right engine disconnect flex line at engine and plug flex line.
14. Increase air pressure to 100 psi.
15. Push in right FIRE PULL handle to pressurize lines in both nacelles. Check that there are no air leaks in any ducting or connections.

NOTE

Slight air leakage at the swivel fittings on the HTL fire bottles are acceptable. Air leakage on the Kidde bottles can be corrected by re-torquing the swivel nut.

16. Shut off air pressure. Remove caps or plugs from unions and reconnect discharge lines.
17. Disconnect air line from manual discharge line boss and connect to manual discharge line.
18. Remove yellow indicator disk from discharge indicator and increase air pressure to 100 psi. Check that manual discharge indicator piston extends about 5/32 inch.
19. Shut off air pressure and disconnect air line.
20. Reconnect manual discharge to boss and reinstall yellow disk in indicator.
21. Position all electrical switches to OFF and disconnect external DC electrical power.

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MAINTENANCE PRACTICES (Continued)

INSPECTION OF FIRE-EXTINGUISHING SYSTEM DOUBLE CHECK "T" VALVE

To inspect the double check "T" valve, proceed as follows:

1. Remove access panel 45.
2. Remove double check "T" valve from airplane.
3. Unscrew check valve assembly from valve body. (See Figure 26-20-3.)
4. Remove O-ring packing and discard.
5. Remove groove pin holding valve assembly together. (Pin passes through the O-ring groove in the cap.)
6. Remove spring from swivel pin and remove swivel pin and washer from disc.
7. Wash all parts thoroughly in dry cleaning solvent, P-D-680 or equivalent.
8. Inspect both faces of disc for scratches or corrosion. Check that spring is not corroded, deformed, stretched, or broken. Check all threads for burrs and check internal seating surfaces for wear and corrosion.

NOTE

Any damage, cracks, pits, corrosion, or abnormal wear necessitates replacement of the valve assembly, P/N 966301.

9. Place spring in cap and insert groove pin into cap so that it passes through the spring loop. Drive groove pin in until it is flush with the bottom of the O-ring groove.
10. Place washer over swivel pin; then insert swivel pin up through the disc and hook end of swivel pin on to spring in valve cap. Check that disc assembly seats properly in cap.
11. Lubricate new O-ring, P/N MS28775-219, with MIL-G-4343B grease and carefully install packing in groove on check valve assembly.
12. Apply an anti-seize compound, Fel-Pro C-5 or equivalent, sparingly to cap threads.
13. Screw check valve assembly into valve body and torque to 75 foot-pounds.
14. Install check valve in airplane.
15. Close access panel.

INSPECTION OF FIRE-EXTINGUISHING SYSTEM DIRECTION VALVE

To inspect the direction valve, proceed as follows:

1. Remove access door 3.
2. Remove direction valve.
3. Remove pins from position indicators and remove indicators. (See Figure 26-20-4.)
4. Remove washers and packings from shaft; then remove nut.
5. Remove packing from nut and discard.
6. Remove shaft from valve body.
7. Remove cap and discard packing.
8. Invert valve so that solenoid is down and remove solenoid assembly, washer, and packing.
9. Remove flapper assembly from valve body.
10. Remove spring and spacer from valve body.
11. If necessary, remove cotter pin and clevis pin and disassemble flapper assembly.
12. Clean all parts except solenoid assembly in dry-cleaning solvent and dry thoroughly with a clean lint-free rag or dry compressed air. Wipe off solenoid assembly with a clean cloth moistened with solvent.
13. Inspect all parts for wear, damage and corrosion.
14. Inspect valve body for cracks and burred threads.
15. Inspect flapper faces for scratches and pits.

NOTE

Any damage, cracks, pits, corrosion, or abnormal wear necessitates replacement of the valve assembly, P/N 502000.

16. Check that solenoid plunger moves freely and is spring loaded out.
17. Check that spring in flapper arm has a free length of about 0.257 inch.
18. Check that included angle formed by the ends of the position indicator shaft spring is 32 (±2) degrees.

NOTE

No lubricant is to be used in assembly.

19. Assemble flapper assembly as required, and install spring in flapper arm recess.

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MAINTENANCE PRACTICES (Continued)

INSPECTION OF FIRE-EXTINGUISHING SYSTEM DIRECTION VALVE (Cont'd)

20. Install solenoid and washer, with new packing, into valve body. Use same number of washers (P/N 502620) as were removed in step 8 (none, 1 or 2). Torque solenoid to 100 inch-pounds. Final position of terminal block on solenoid should be not more than 30° clockwise or 90° counter-clockwise from centerline of outlet opposite cap (looking down, solenoid above outlet).
21. Install flapper assembly in valve body so that ear on flapper arm engages notch in solenoid plunger.
22. Insert shaft part way into valve body and through flapper arm.
23. Install spacer and spring inside valve body and slide over end of shaft. Place tang of spring over notched side of flapper arm to hold flapper against normally closed port.
24. Completely insert shaft through valve body so that shaft engages flapper arm and position indicator pin holes in shaft are properly positioned.
25. Install nut with new packing over indicator shaft; then install new packings, washers and indicators.
26. Install cap with new packing and safety to valve body with lockwire.
27. Apply 24 VDC to solenoid terminals and check that valve operates properly.
28. Install valve in airplane.
29. Close access panel.

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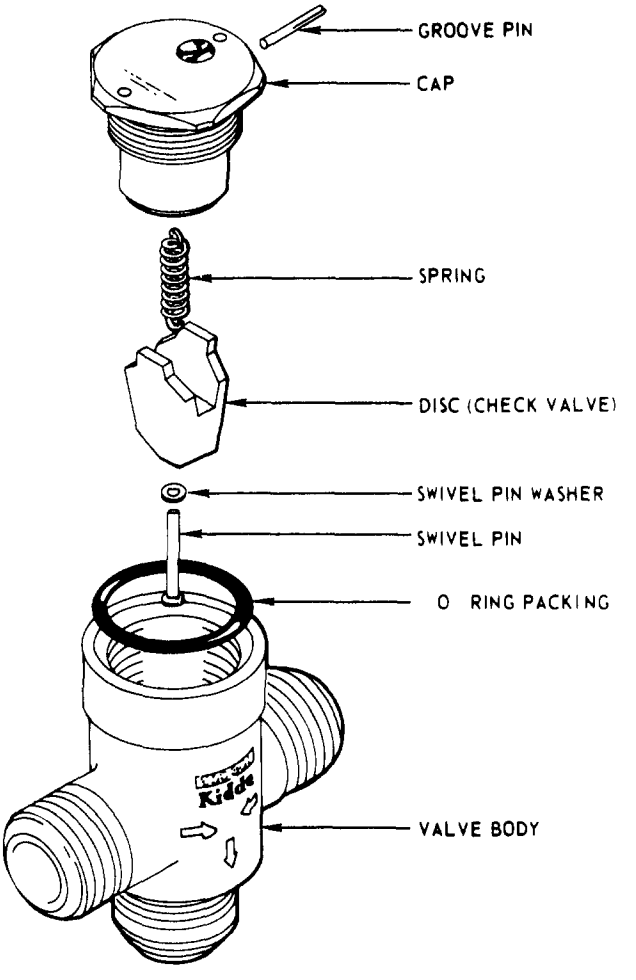

Figure 26-20-3 — Double Check "T" Valve Assembly

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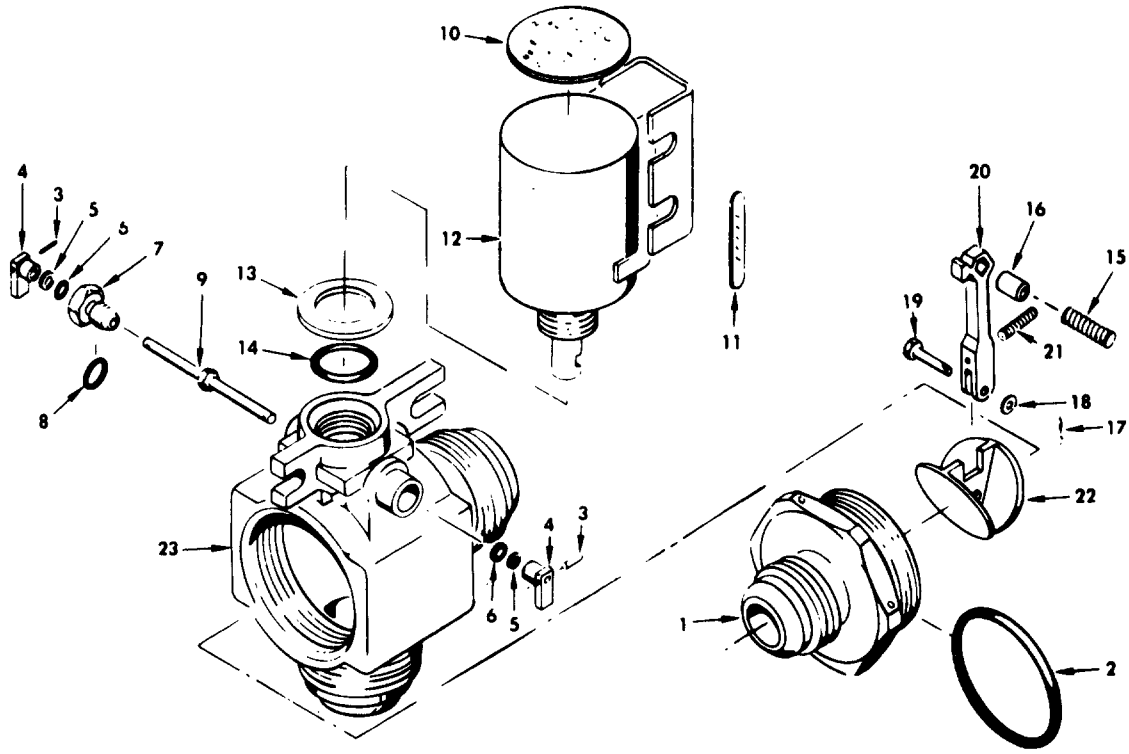
MAINTENANCE PRACTICES (Continued)



1. CAP
2. PACKING (AS113-23-131)*
3. PIN (3/64-DIA X 1/4-INCH MINIATURE SPIRAL PIN) (2 each) (047-0250MCP)*
4. INDICATOR
5. WASHER
6. PACKING (AS113-23-006 2 each)*
7. NUT
8. PACKING (AS113-23-010)*
9. SHAFT
10. DECAL
11. DECAL
12. SOLENOID ASSY (502608-1)
13. WASHER (502620)*
14. PACKING (AS113-23-016)*
15. SPRING (502607)
16. SPACER
17. COTTER PIN
18. WASHER
19. PIN
20. ARM
21. SPRING (502604)
22. FLAPPER
23. BODY

* May be procured from:

Hydraulic Research Textron
10445 Glenoaks Blvd.,
Pacoima, CA 91331
Phone: (818)890-9247

or

Sabreliner Corporation
8849 Seeger Industrial Drive
St. Louis, MO 63134-1088
Phone: (314)537-3660

Figure 26-20-04 — Fire Extinguisher System Direction Valve

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CHAPTER

27

FLIGHT CONTROLS

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FLIGHT CONTROLS - DESCRIPTION

1. General

The flight control systems include an aileron control system, a rudder control system, an elevator control system, and a gust lock system. The control systems are of the conventional mechanical type incorporating electrically operated trim systems. No boost systems are used on this airplane.

The aileron control system consists of two interconnected control wheels in the cockpit, mechanically linked by push rods, bell cranks, cables, and cable sectors to two aerodynamically balanced ailerons. The ailerons are hinged to the outboard trailing edges of both wings, with a trim tab on the left aileron only. Both ailerons are operated, at the same time and in opposite directions, to provide lateral (roll) control of the airplane.

The rudder control system consists of two pairs of interconnected rudder pedals in the cockpit and a centering bungee mechanically linked by push rods, bell cranks, and cables to an aerodynamically balanced rudder. The rudder, with a trim tab at its trailing edge, is hinged to the trailing edge of the vertical stabilizer to provide directional (yaw) control of the airplane.

The elevator control system consists of two interconnected forward and aft moving control columns in the cockpit and a bobweight mechanically linked by push rods, bell cranks, and cables to aerodynamically balanced elevators. The elevators are hinged to the trailing edge of the horizontal stabilizer, to provide longitudinal (pitch) control of the airplane. A spring-type bias bungee is incorporated to establish a no-load center position for the elevator controls by interconnecting the elevators and horizontal stabilizer. This no-load position varies automatically with stabilizer trim position. The complete elevator control system is balanced by a bobweight balance bungee.

The gust lock system provides a means of mechanically locking the aileron control system, rudder control system, and elevator control system in their near-neutral positions while the airplane is on the ground. The gust lock system is mechanically operated by a handle on the left side of the pilot's control pedestal. Both throttle levers must be retarded before the gust lock can be engaged. With the lock engaged, the throttles cannot be advanced beyond the idle range, thereby preventing high power settings with the flight controls locked.

2. Flight Control Trim System (See figure 27-00-1.)

The flight control trim systems are electrically operated and consist of an aileron trim system, a rudder trim system, and a horizontal stabilizer trim system. The trim systems are provided with a normal trim circuit and an alternate trim circuit, both receiving power from the 28-volt dc essential bus. Normal trimming of the airplane for a required flight attitude is controlled by trim switches, accessible to the pilot and copilot. An

alternate trim system, operated by switches on the pilot's side of the cockpit is for use in case of a normal trim system failure. The normal and alternate trim switches control electrical actuators which operate an aileron trim tab, a rudder trim tab, and the horizontal stabilizer. A trim arming switch is provided in each normal trim switch on the control wheels. The trim arming switch must be pressed in to allow normal trim of the ailerons and horizontal stabilizer on airplanes NA-380-1 through -43 not changed by Service Bulletin 75-22 and airplanes NA-370-1 and later. On airplanes NA-380-44 and later and airplanes NA-380-1 through -43 changed by Service Bulletin 75-22, the trim arming switches are moved simultaneously for normal horizontal and aileron trim.

The aileron trim system is operated by a trim switch on the outboard grip of either control wheel on airplanes NA-380-1 through -43 not changed by Service Bulletin 75-22. On airplanes changed by Service Bulletin 75-22 or airplanes NA-380-44 and later, the normal aileron trim system is operated by trim arm switches on the center pedestal. The switches are moved simultaneously for aileron trim. The trim switch controls an electromechanical aileron trim actuator mounted in the left aileron. The actuator mechanically operates the trim tab, hinged to the aileron trailing edge. The rudder trim system is operated by a trim switch on the aft face of the center pedestal. The trim switch controls an electromechanical rudder trim actuator mounted in the rudder. The actuator mechanically operates the trim tab, hinged to the rudder trailing edge. The horizontal stabilizer trim system is operated by a trim switch on the outboard grip of either control wheel. The trim switch controls two electromechanical actuators interconnected by a flexible shaft which ensures synchronized actuator travel. The actuators mechanically position the horizontal stabilizer leading edge. The actuators operate the stabilizer at two speeds: low speed when the landing gear is up, and high speed when the main landing gear is down and locked. On NA-380-38 and later airplanes an audible warning horn will sound whenever the horizontal stabilizer trim actuators are energized. The stabilizer and elevators are interconnected through the bias bungee. Movement of the stabilizer repositions the elevators in a no-load position through the bias bungee.

The trim systems are provided with a trim position indicating system consisting of three trim position transmitters and three trim position indicators. Each system mechanically operates a potentiometer-type transmitter. Each transmitter electrically operates a potentiometer-type indicator. All three indicators are grouped together on the copilot's inboard instrument panel.

A. Trim Selector Switch

A positive-lock trim selector switch, on the alternate trim control panel, is powered by the dc essential bus. The switch has three positions: NORMAL, ALTERNATE, and OFF. When the switch is in the NORMAL position, longitudinal and lateral trim are controlled by the normal trim switches on the control wheels, and directional trim is controlled by the normal rudder trim switch on the center pedestal. With the trim selector switch in the ALTERNATE position, all trim is controlled by the alternate trim switches on the alternate trim control panel. With the switch in OFF position, both trim systems are inoperative.

B. Normal Trim Switches (Airplanes NA-380-1 through -43 not changed by Service Bulletin 75-22 and airplanes NA-370-1 and Later)

Normal trim of the horizontal stabilizer or of the aileron tab is provided through a five-position switch on the aft face of each control wheel outboard grip. Switch positions are L WG DN (left wing down), NOSE DN, R WG DN (right wing down), and NOSE UP. When the switch is released, it automatically returns to the OFF (center) position and trim action stops. Normal trim of the rudder is provided through a three-position switch on the aft face of the center pedestal below the throttles. Holding the rudder trim switch to R (right) trims the nose of the airplane to the right; holding it to L (left) trims the nose left. When the switch is released, it automatically returns to the OFF (center) position and trim action stops. The normal trim switches are powered by the dc essential bus.

B1. Normal Trim Switches (Airplanes NA-380-44 and Later and airplanes NA-380-1 through -43 changed by Service Bulletin 75-22)

Normal trim of the horizontal stabilizer is provided through two adjacent switches (arm and trim) on the aft face of each control wheel outboard grip. When the switches are released, they automatically return to off (center) and the trim action stops. The normal trim of the ailerons is provided through two adjacent switches (arm and trim) located on the center pedestal. Switch positions are LEFT WING DOWN and RIGHT WING DOWN. When the switches are released they automatically return to off (center) and the trim action stops. Normal trim of the rudder is provided through a three-position switch on the aft face of the center pedestal below the throttles. Holding the rudder trim switch to RIGHT trims the nose of the airplane to the right; holding it to LEFT trims the nose left. When the switch is released, it automatically returns to the (center) position and trim action stops. The normal trim switches are powered by the dc essential bus.

C. Alternate Trim Switches

The alternate trim switches are to be used in case of a normal trim system failure. When the trim selector switch is positioned at ALTERNATE, the ailerons and horizontal stabilizer are trimmed through the five-position switch on the alternate trim control panel. Holding the alternate trim switch to either LEFT WING DOWN or RIGHT WING DOWN causes the corresponding wing to be trimmed down. Holding the alternate trim switch forward to NOSE DOWN trims the airplane nose down; holding it aft to NOSE UP trims the nose up. When the switch is released, it automatically returns to the OFF (center) position and trim action stops. Alternate trim of the rudder is provided through a three-position switch on the alternate switch control panel. Holding the trim switch at RIGHT trims the nose of the airplane to the right; holding it at LEFT trims the nose left. When the switch is released, it automatically returns to the OFF (center) position and trim action stops. The alternate trim switches are powered by the dc essential bus.

D. Trim Arming Switches (Airplanes NA-380-1 through -43 not changed by Service Bulletin 75-22 and airplanes NA-370-1 and Later)

A trim arming switch is in the center of the normal trim switch on each control wheel. These switches provide an automatic trim disconnect any time the switch is not depressed, to prevent trim runaway. When the switch is released, no power is available to the normal trim system for the ailerons and horizontal stabilizer. To operate trim, the trim arming switch must first be depressed to connect electrical power through the normal trim switch. Rudder trim operation and alternate trim operation are not affected by the trim arming switches.

D1. Trim Arming Switches (Airplanes NA-380-44 and Later and airplanes NA-380-1 through -43 changed by Service Bulletin 75-22)

A horizontal stabilizer arming switch is the outboard switch on the pilot's control wheel and the inboard switch on the copilot's control wheel, and the aileron arming switch is the forward AILERON TRIM switch on the center pedestal (see figure 27-10-2). These switches provide a trim disconnect any time the switch is not actuated, to prevent trim runaway. To operate trim, the arming and trim switches are actuated simultaneously to connect electrical power through the normal trim switch. Rudder trim operation and alternate trim operation are not affected by the trim arming switches.

E. Trim Emergency Disconnect and Audible Warning Horn Systems

On NA-370-1 and Later and NA-380-1 and Later airplanes changed by Service Bulletin 74-6, a trim disconnect system is installed to simultaneously disconnect the stabilizer, aileron and rudder trim system whenever the trim disconnect switch on the pilot's and copilot's control wheel is depressed. The system consists of a trim disconnect indicator light (installed on the copilot's inboard instrument panel, two trim disconnect switches (one on each control wheel), relays and the necessary electrical wiring to complete the system. The trim disconnect indicator light is also provided with the necessary relays required for the dimming function and master caution light operation (see figure 27-00-1). Incorporation of Service Bulletin 74-7 (Airplanes NA-380-1 through -37 changed by Service Bulletin 74-7 or airplanes NA-380-38 and later) provides the flight crew with an audible warning whenever the horizontal stabilizer actuators are energized. The audible warning horn will sound whenever the horizontal stabilizers are trimmed manually or trimmed by the auto pilot system. The audible warning function is removed whenever the disconnect switch on the pilot's or copilot's control wheel is depressed (see figure 27-00-1). The following paragraphs describe the operation of the trim disconnect system with Service Bulletin 74-6 and 74-7 incorporated. With electrical power applied to the airplane and the trim selector switch in normal or alternate position, when the trim switch on the control wheel is actuated, the selected trim actuator is energized. When the trim actuator is cycling, depressing either trim disconnect switch routes 28 VDC electrical power to the aileron and rudder disconnect relay (K543) coil. Relay K543 is then energized, routing electrical power to the RH and

LH stabilizer trim disconnect relays (K541 and K542) which are then energized, thus opening the ground control circuit to both stabilizer actuators. When relay K543 is energized, it removes the ground circuit from the aileron and rudder trim actuators, applies power to the trim disconnect indicator light and actuates the master caution light system. Contacts 3 and 5 of relay K543 provide an electrical holding circuit to keep relay K543 energized, which in turn disables the stabilizer, rudder and aileron trim actuators and actuates the caution indicator light system. The holding circuit may be removed by positioning the trim selector switch at OFF. During normal trim system operation (trim disconnect system de-energized) whenever the horizontal stabilizer trim is actuated, an audible warning will sound to notify the flight crew that the stabilizer is being trimmed manually or by the autopilot system.

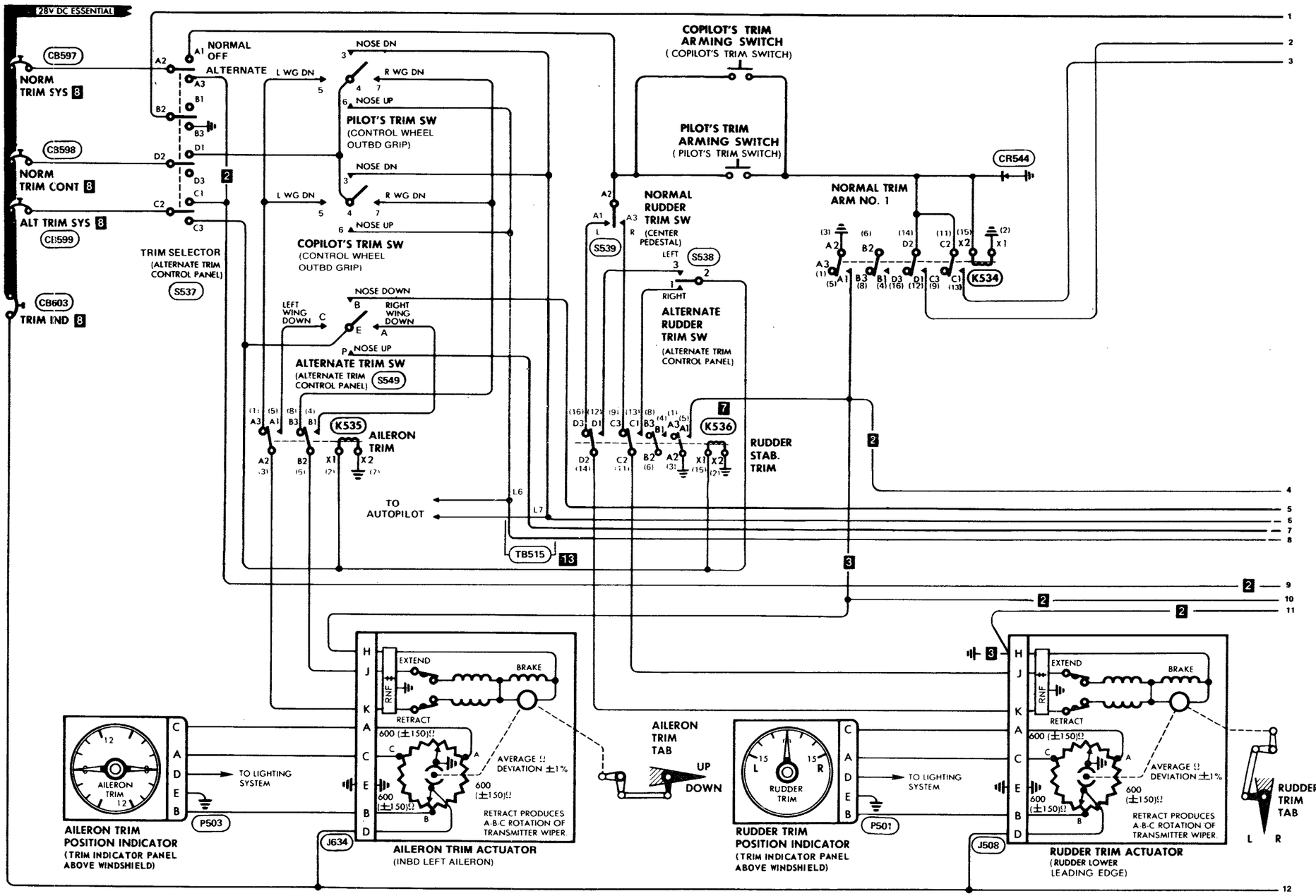
F. Separate Stabilizer and Aileron Dual Actuation Switches and a Positive Normal System Horizontal Stabilizer Power Disconnect. (On airplanes NA-380-1 through -43 changed by Service Bulletin 75-22 or airplanes NA-380-44 and Later)

This change replaces the existing stabilizer and aileron trim switches on both control wheels with a rocker type dual actuation stabilizer split trim switch (see figure 27-00-204), and adding a like switch on the center pedestal for aileron trim (see figure 27-10-2). Each switch assembly has two adjacent switches (arm and trim), so they can be operated simultaneously.

This change (see figure 27-00-2) adds two relays (K544, K545) to disconnect electrical power from the normal system nose up or nose down relay contacts T1 and T3 on K538 and K540 relays, when the alternate system is selected.

NOTES

1. *On S/N 370-1 thru -9, S/N 380-1 thru -37, after S/B 74-7, and S/N 380-38 and subs*
2. *After S/B 74-6*
3. *Before S/B 74-6*
4. *On S/N 370-1 thru -9 and S/N 380-1 thru -37 after S/B 74-7, and SOK 27, and S/N 380-38 thru -72 after SOK 27.*
5. *On S/N 370-1 thru -9, S/N 380-1 thru -37 after S/B 74-7 but after SOK 27, and S/N 380-38 and subs before SOK 27.*
6. *Where two relay contact numbers are given, the numbers in parentheses are for S/N 370-1 and -7, and S/N 380-1 thru -43. The numbers without parentheses are for S/N 370-2 thru -6, -8, and -9.*
7. *S/N 370- thru (FS 401 RH), S/N 380 (relay assy 125, 150 RH)*
8. *Left console circuit-breaker panel*
9. *Radio noise filter*
10. *S/N 370 (FS 400 RH), S/N 380 (PNL 57, FS 428 RH)*
11. *S/N 370 (FS 401 RH), S/N 380 (PNL 104 FS 150 RH)*
12. *Relay assy (FS 427)*
13. *FS 98 RH*
14. *S/N 370 (FS 400 LH)*
15. *S/N 380 (FS 150 RH, PNL 104)*



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Flight Control Trim System - S/N 370-1 Thru -9
and 380-1 Thru -43 Before S/B 75-22
Figure 27-00-1 (Sheet 1 of 2)

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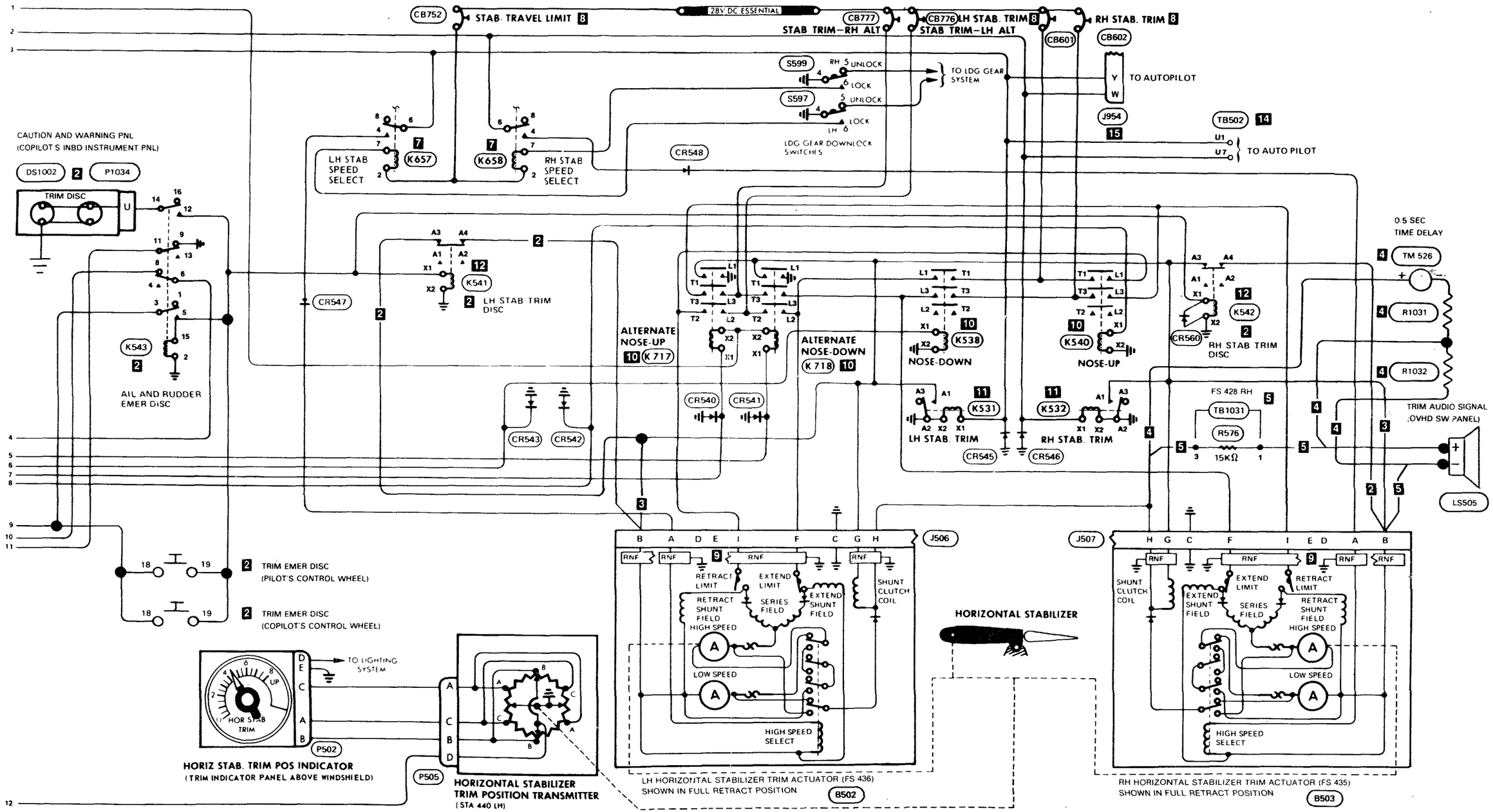

Flight Control Trim System - S/N 370-1 Thru -9
and S/N 380-1 Thru -43 Before S/B 75-22
Figure 27-00-1 (Sheet 2 of 2)

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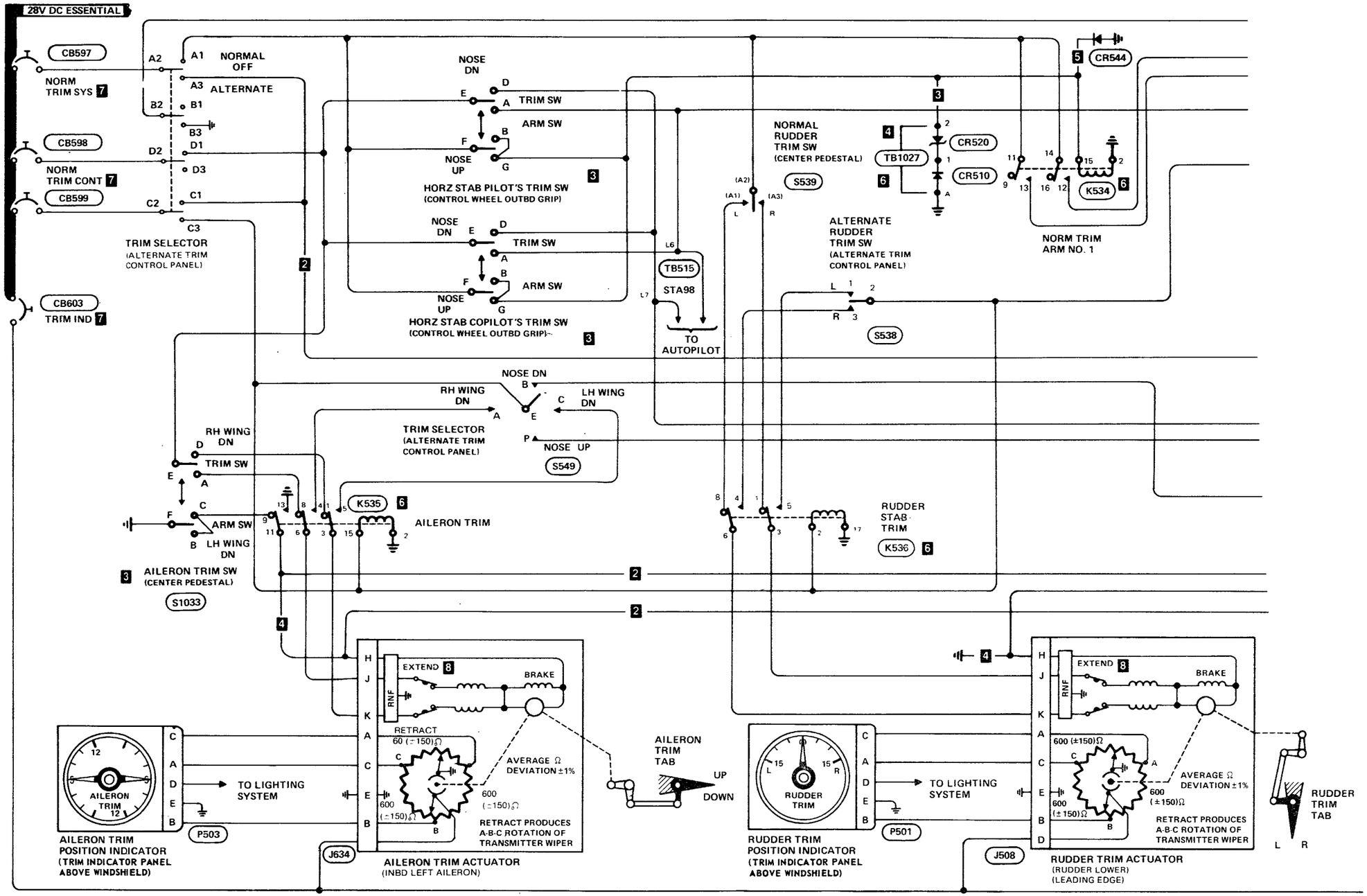
NOTES

1. *S/N 380-38 and subs and S/N 380-1 thru -37, and S/N 370-1 thru -9 after S/B 74-7*
2. *S/N 380-1 and subs after S/N 74-6*
3. *S/N 380-44 and subs*
4. *Before S/B 74-6*
5. *S/N 380-1 thru -43*
6. *Relay assy 125, FS 150*
7. *Left console circuit breaker panel*
8. *Radio noise filter*
9. *FS 428 RH*
10. *FS 150 RH panel 107*



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Flight Control Trim System S/N 380-1 Thru -43
After S/B 75-22 and S/N 380-44 and Subs
Figure 27-00-2 (Sheet 1 of 3)

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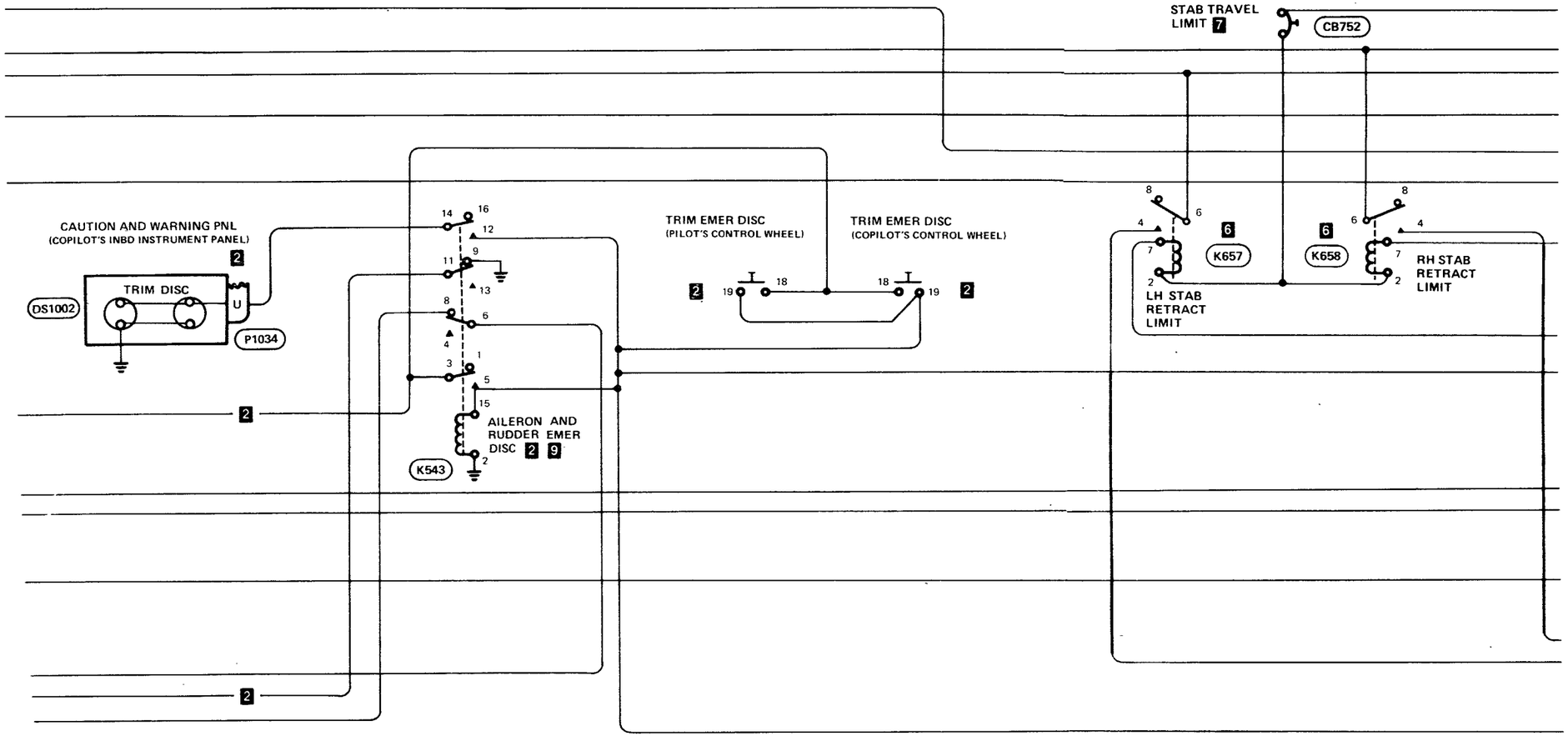

Flight Control Trim System S/N 380-1 Thru -43
After S/B 75-22 and S/N 380-44 and Subs
Figure 27-00-2 (Sheet 2 of 3)

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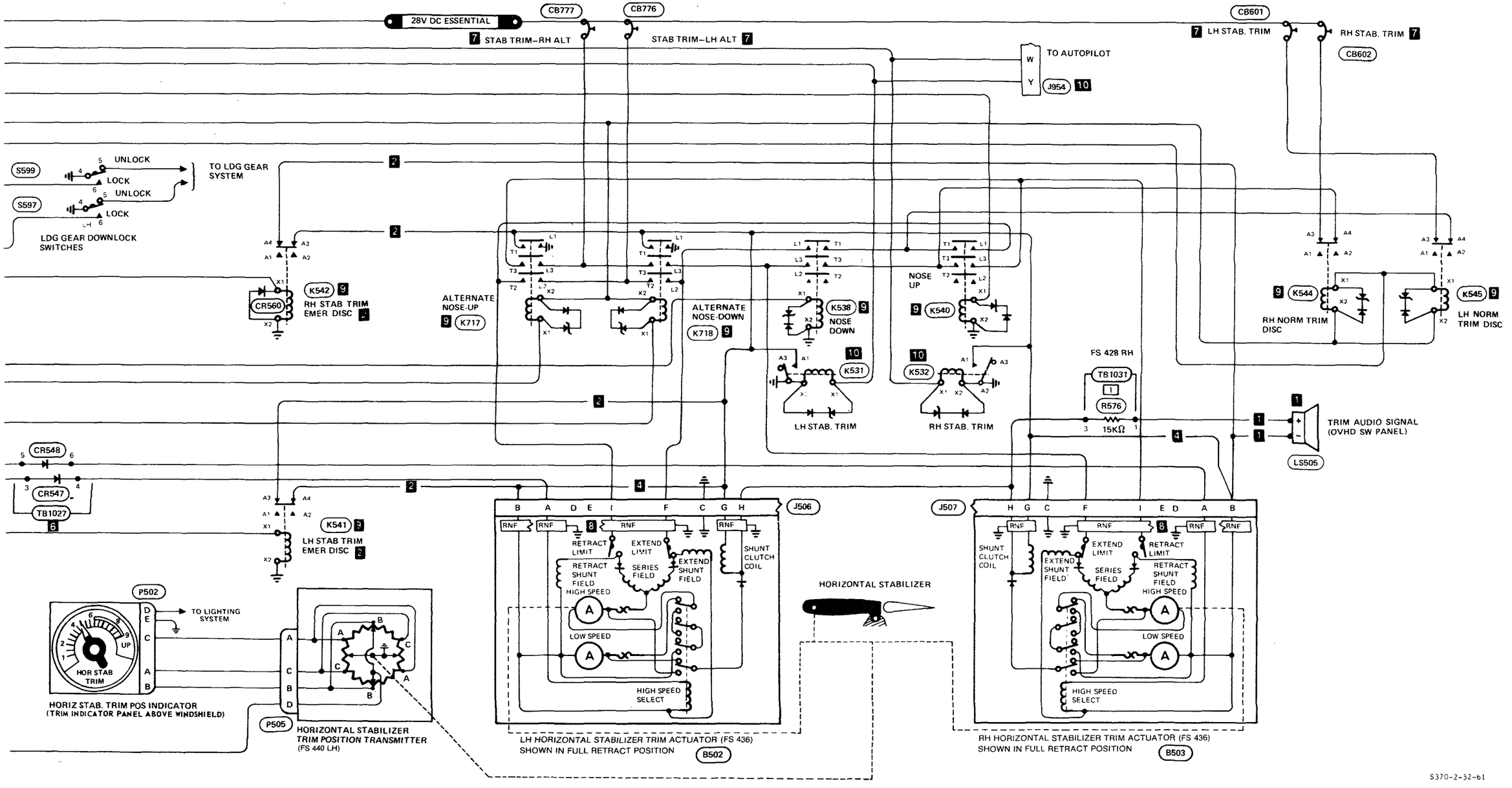

S370-2-52-61

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Flight Control Trim System S/N 380-1 Thru -43
After S/B 75-22 and S/N 380-44 and Subs
Figure 27-00-2 (Sheet 3 of 3)

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FLIGHT CONTROLS - TROUBLE SHOOTING

1. General

For trouble shooting of individual flight control systems, refer to applicable system.

2. Trouble Shooting Flight Control Trim System

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. TRIM THROWS UNSATISFACTORY.		
(1) Trim system improperly adjusted.	Check rigging of trim system.	
(2) Mechanical interference.	Operate trim actuators back and forth while inspecting for mechanical interference.	
(3) Actuators not fully extended or retracted.	With rod ends disconnected, check both actuators for full extension or retraction. If actuators do not make full travel, limit switch is not properly set.	Readjust external limit switches. If this does not correct conditions, replace actuator.
B. TRIM POSITION INDICATOR INOPERATIVE.		
(1) Electrical system faulty.	Perform continuity check.	
(2) Faulty transmitter.	Check transmitter. If transmitter does not meet check-out requirements, a faulty transmitter is indicated.	Replace transmitter.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
(3) Faulty trim position indicator.	If there is continuity in electrical system and transmitter meets check-out requirements, a faulty trim position indicator is indicated.	Replace trim position indicator.
C. TIMING REQUIREMENTS UNSATISFACTORY.		
(1) Improper timing due to faulty switches or improperly functioning actuators.	Check for binding and that all mechanical parts are properly adjusted.	Remove and replace actuator.
(2) Electrical system faulty.	Perform continuity check.	Repair electrical system.

FLIGHT CONTROLS

MAINTENANCE PRACTICES

SERVICING FLIGHT CONTROL SYSTEMS

No servicing of flight control systems is required. All joints in the flight control systems have lubricant-sealed bearings and do not require lubrication.

SERVICING FLIGHT CONTROL TRIM SYSTEMS

The only servicing required in the flight control trim system is lubrication of the horizontal stabilizer trim actuators and idler links.

CABLE TENSION VARIATION WITH TEMPERATURE

See Figures 27-00-201 or 27-00-201A.

LOCATING WING REFERENCE PLANE

Locating the wing reference plane (WRP) is required for current rigging procedures of flaps and ailerons. (See Figure 27-00-202.)

INSTALLING RUDDER, ELEVATOR, AND AILERON HINGE PINS

See Figure 27-00-203.

NOSE WHEEL STEERING, FLIGHT CONTROL TRIM, AND INTERPHONE—MICROPHONE SWITCHES REPLACEMENT

NOTE

Before unsoldering wires from switch terminals, make sure wires are adequately identified for proper reassembly.

See Figure 27-00-204.

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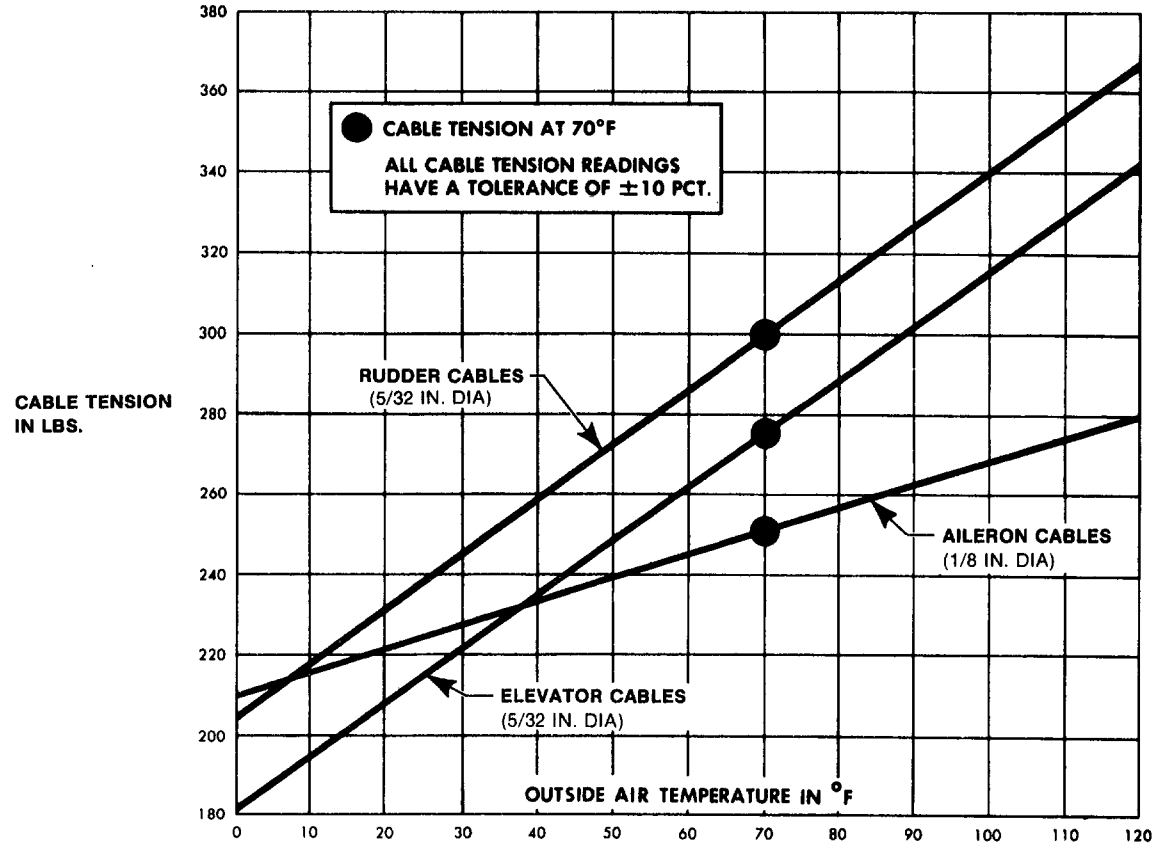

NOTE

If airplane is in the sun, point nose of airplane toward, or away from, sun to obtain an even temperature on exposed surfaces.

1 Before adjusting cable tension, place airplane in an even-temperature area for at least one hour.

2 Take temperature in nose wheel well. If airplane is in sun, add 5 degrees to this temperature.

3 Rig as close to center of tolerance as possible.

AUTOPILOT SERVO(S) CABLE TENSION CHARTS

Adjust aileron autopilot servo cable tension to 50 (±5) pounds. Cable tension variation due to temperature changes is not a factor when adjusting the aileron autopilot servo cable.

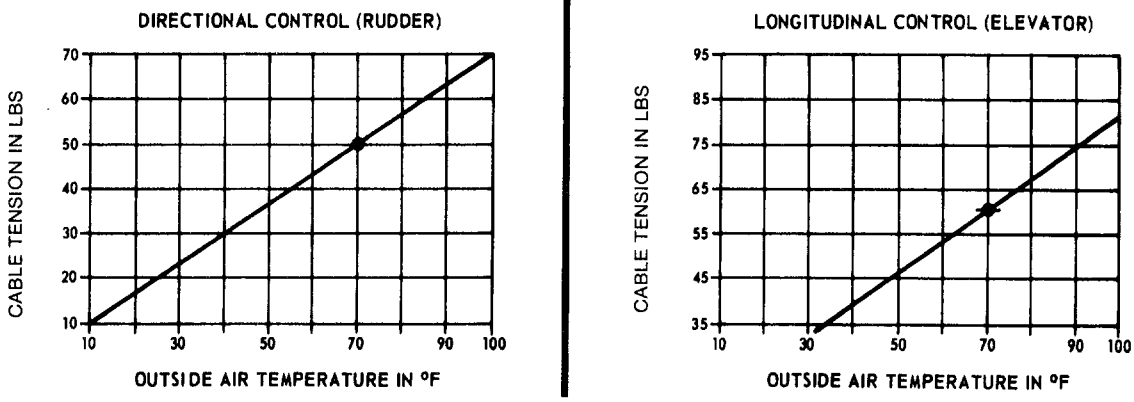

Figure 201 — Cable Tension Variation With Temperature (Before SOK 34)

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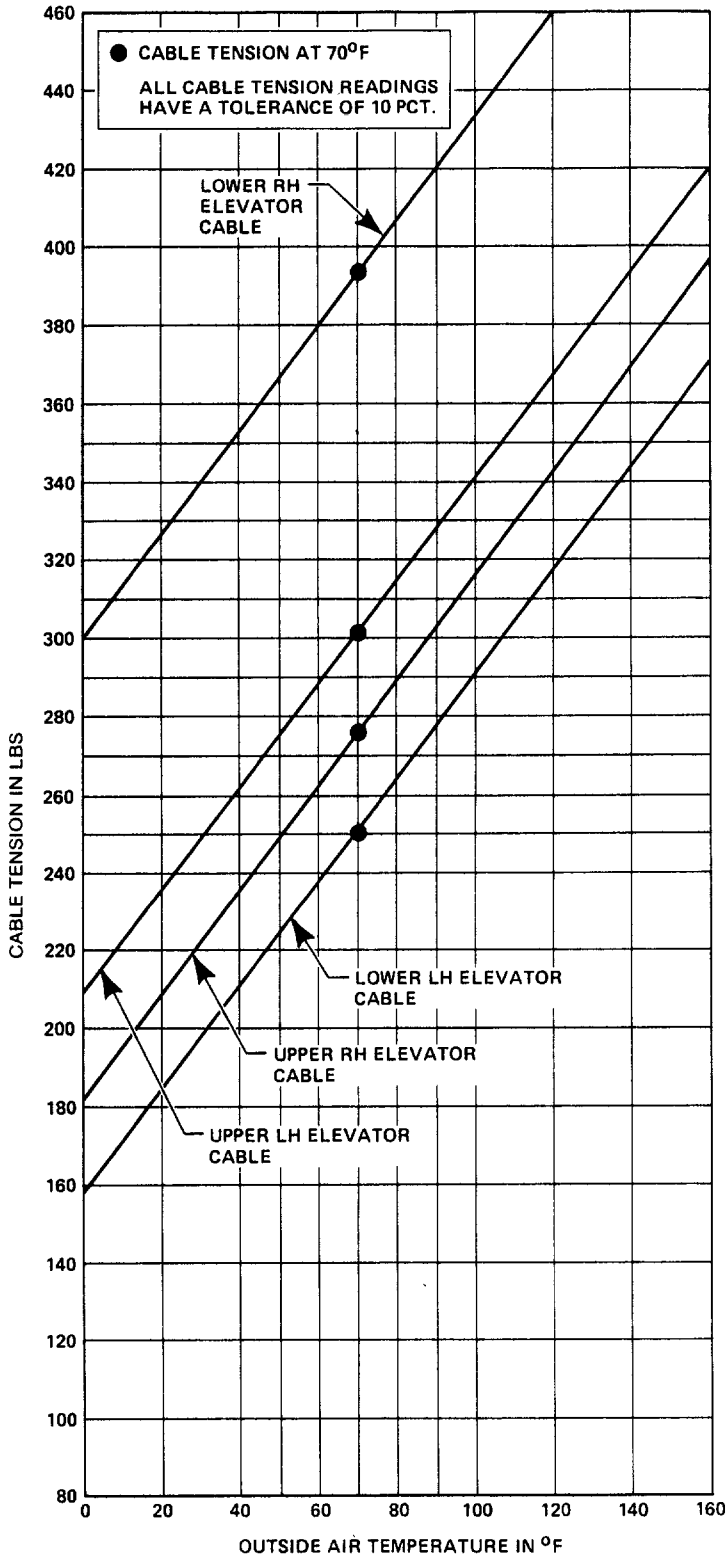

Cable Tension Variation With Temperature — S/N 370-1 and -7, S/N 380-1 and Subs After SOK 34
Figure 201A (Sheet 1 of 2)

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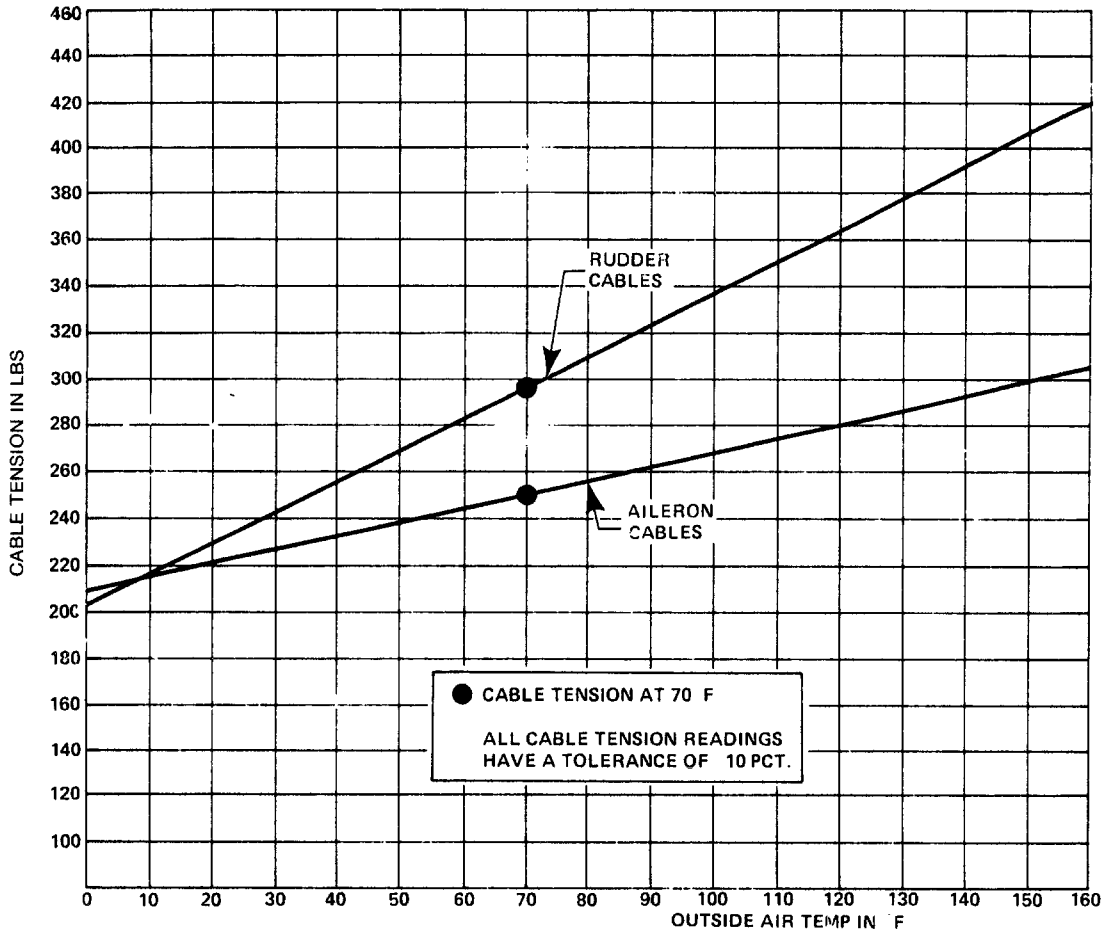


NOTE

If airplane is in the sun, point nose of airplane toward, or away from, sun to obtain an even temperature on exposed surfaces.

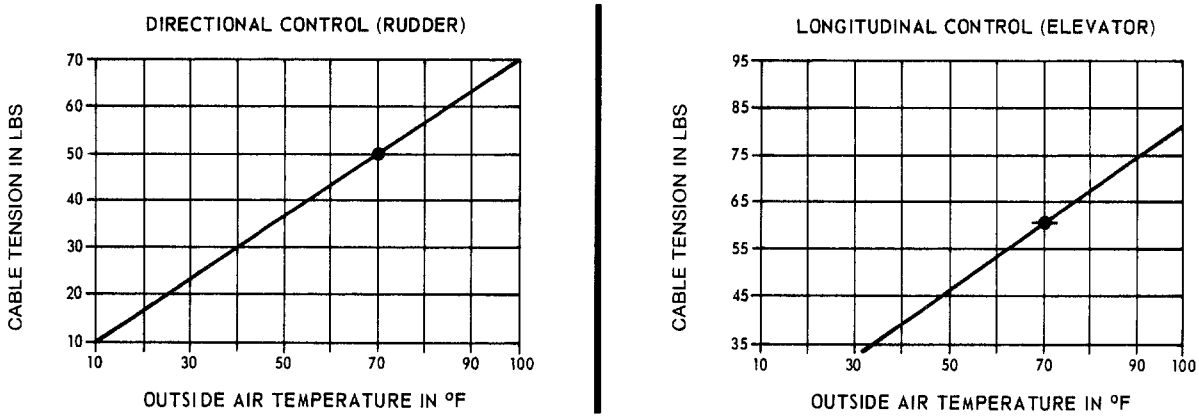
1 Before adjusting cable tension, place airplane in an even-temperature area for at least one hour.

2 Take temperature in nose wheel well. If airplane is in sun, add 5 degrees to this temperature.

3 Rig as close to center of tolerance as possible.

AUTOPILOT SERVO(S) CABLE TENSION CHARTS

Adjust aileron autopilot servo cable tension to 50 (±5) pounds. Cable tension variation due to temperature changes is not a factor when adjusting the aileron autopilot servo cable.



Cable Tension Variation With Temperature — S/N 370-1 and -7, S/N 380-1 and Subs After SOK 34
Figure 201A (Sheet 2 of 2)

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NOTE

The wing inbd and outbd fixed trailing edges cannot be used as a reference, since they are not accurately held to the wing reference plane during the fabrication process.

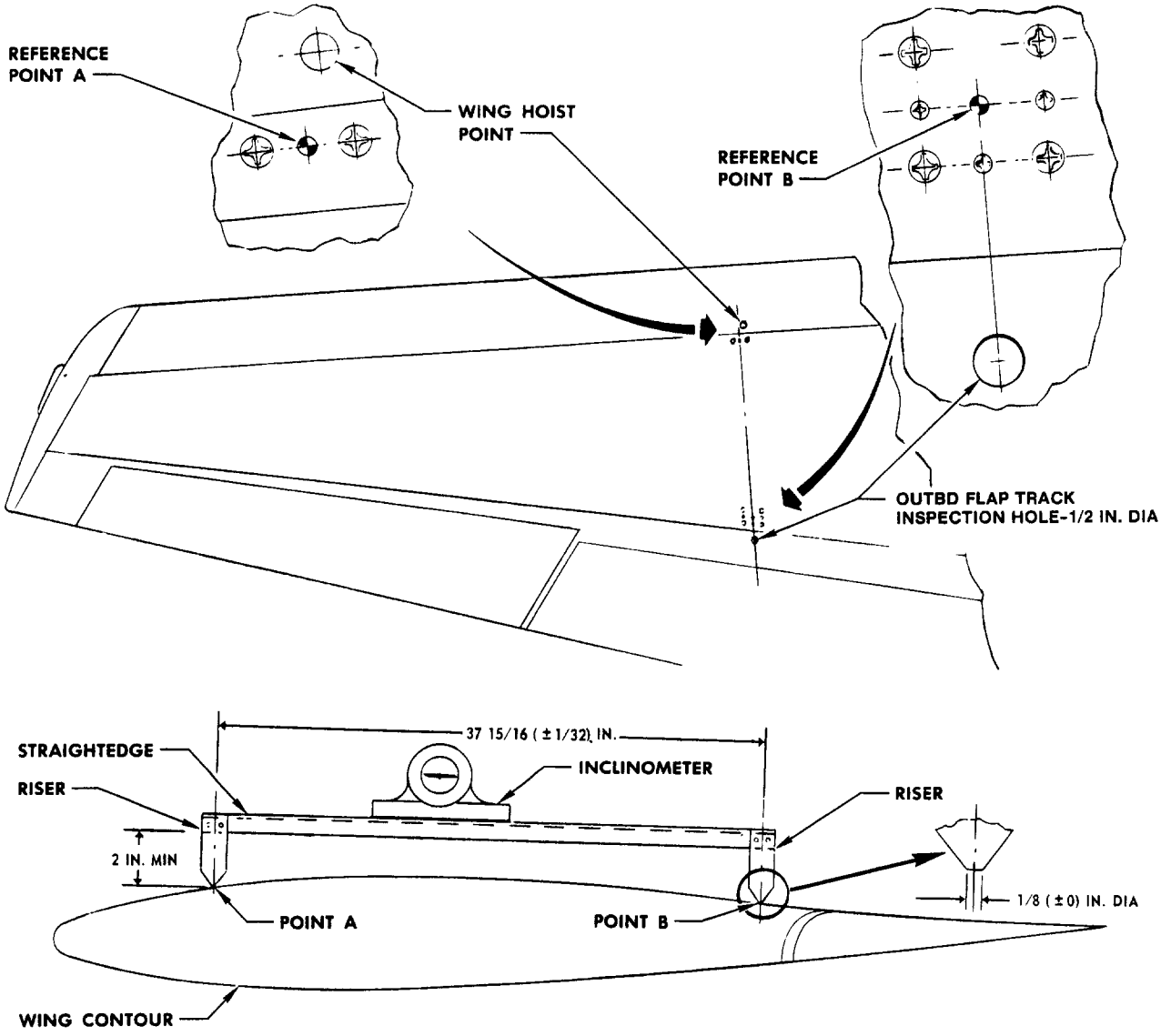
1 Place airplane on jacks and level.

2 Locate reference point A. This point is directly aft of the outbd fwd wing hoist hole and must be located on the CL between the two sealing screws aft of the hoist hole.

3 Locate reference B. This point is fwd of the outbd flap track inspection hole and must be located on the CL of the screws fwd of the inspection hole.

4 Place straightedge (on risers of equal length) parallel to the plane of points A and B. The distance between the points is 37-15/16 (±1/32) in.

5 Measure ground angle of this straightedge (wing reference angle) with an inclinometer or bevel protractor and record this measurement. This angle will be 2°28′ (±30°) with airplane level on jacks.



TYPICAL WING REFERENCE FIXTURE

Locating Wing Reference Plane
Figure 202

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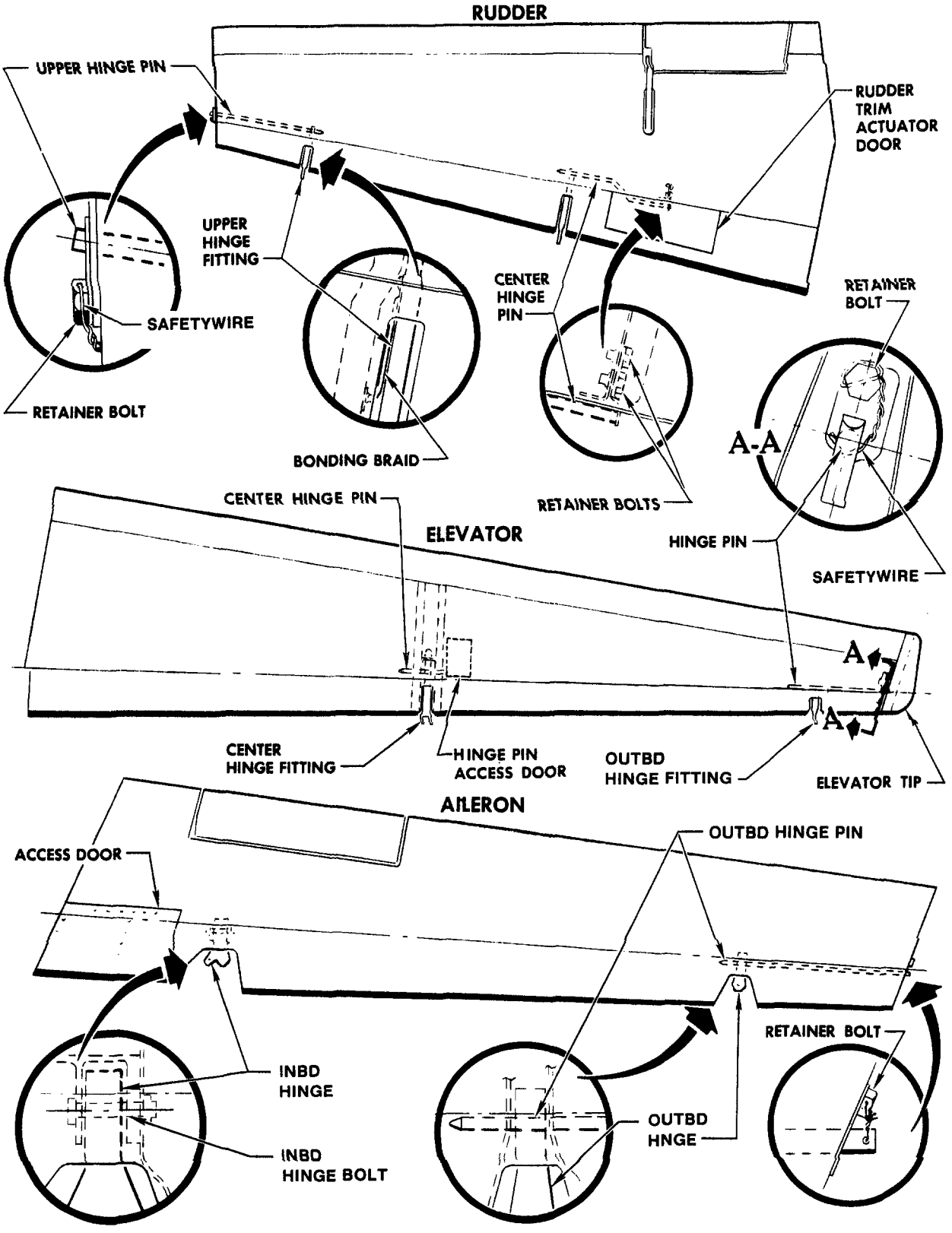

Installing Rudder, Elevator, and Aileron Hinge Pins
Figure 203

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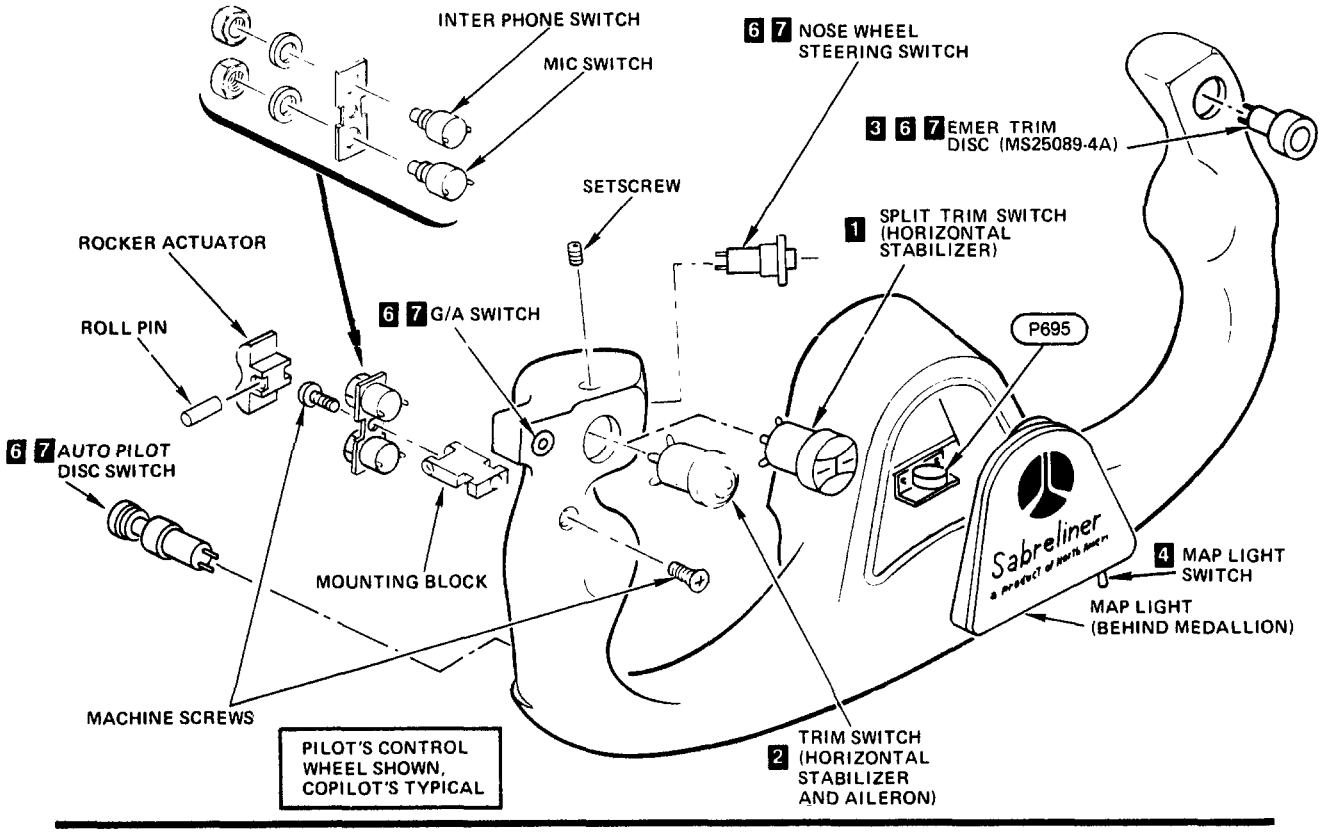

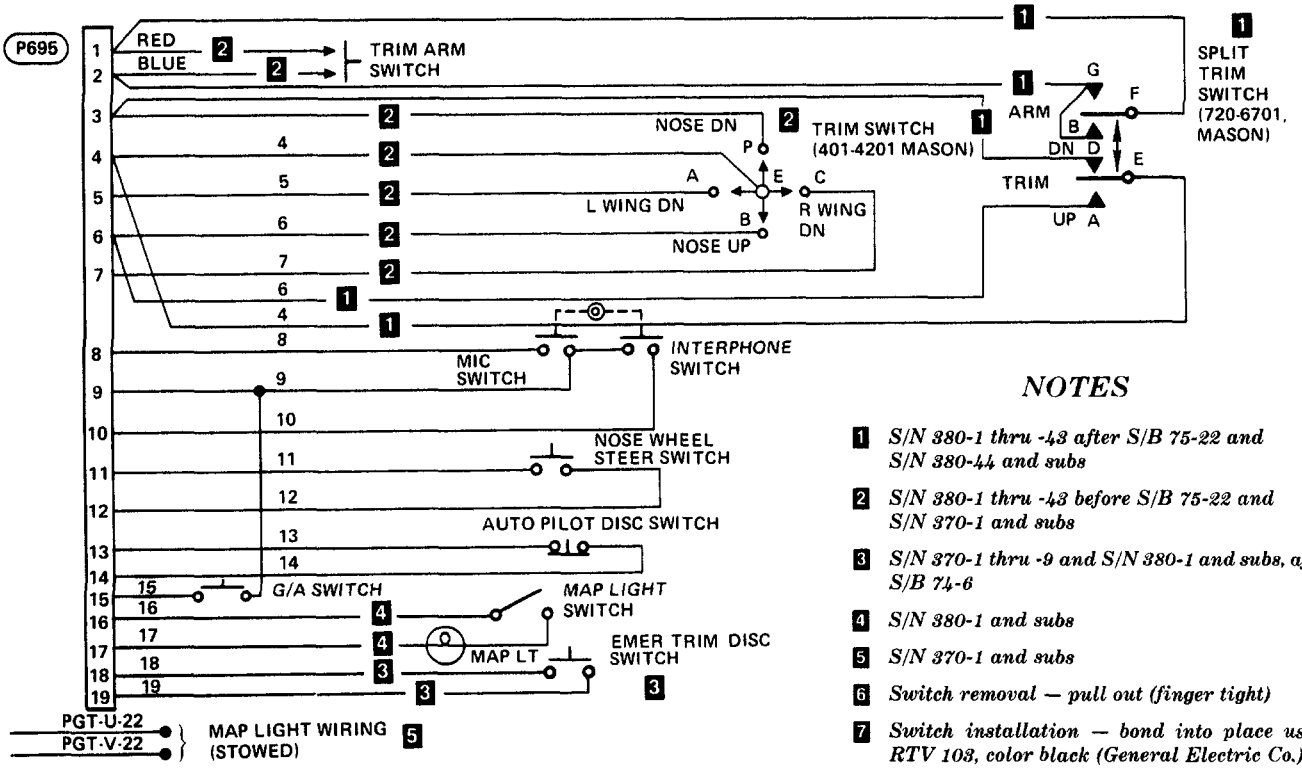

NOTES

1. *S/N 380-1 thru -43 after S/B 75-22 and S/N 380-44 and subs*
2. *S/N 380-1 thru -43 before S/B 75-22 and S/N 370-1 and subs*
3. *S/N 370-1 thru -9 and S/N 380-1 and subs, after S/B 74-6*
4. *S/N 380-1 and subs*
5. *S/N 370-1 and subs*
6. *Switch removal — pull out (finger tight)*
7. *Switch installation — bond into place using RTV 103, color black (General Electric Co.)*

Replacing Control Wheel Switches
Figure 204

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MAINTENANCE PRACTICES (Cont'd)

OPERATIONAL CHECK OF FLIGHT CONTROL TRIM SYSTEMS — S/N 380-1 THRU -43 BEFORE S/B 75-22 AND S/N 370-1 AND SUBS

See table 201 for a general check of the aileron trim system, rudder trim system, and horizontal stabilizer trim system. If trouble is encountered, and the source of trouble is not evident, refer to the more detailed check of the specific system.

Table 201. Operational Check of Flight Control Trim Systems — S/N 380-1 thru -43 Before S/B 75-22 and S/N 370-1 and Subs

PROCEDURE	NORMAL INDICATION
EQUIPMENT/TOOLS • 28 Vdc Electrical Power Supply	
1. Electrical power ..... on	
2. Circuit breakers: NORMAL TRIM (four) .....in ALTERNATE TRIM (three) .....in TRIM IND .....in STAB SPEED CONT.....in	
3. Set trim selector switch .....NORM (Located on ALTERNATE TRIM CONTROLS panel.)	
4. Push and hold trim switch .....L WG DN (Located on left grip of pilot's control wheel.)	A. Aileron trim tab moves down. B. AILERON TRIM indicator.....L WG DN
5. Push and hold trim switch .....R WG DN	A. Aileron trim tab moves up. B. AILERON TRIM indicator.....R WG DN
6. Operate trim switch until AILERON TRIM indicator reads zero.	A. Aileron trim tab aligns with trailing edge of aileron. B. AILERON TRIM indicator.....zero
7. Repeat steps 4 thru 6 for co-pilot's wheel.	
8. Push and hold RUD TRIM switch ..... L (left) (Located on aft face of center pedestal.)	A. Rudder trim tab moves right. B. RUDDER TRIM indicator ..... L (left)
9. Push and hold RUD TRIM switch ..... R (right)	A. Rudder trim tab moves left. B. RUDDER TRIM indicator ..... R (right)
10. Operate RUD TRIM switch until indicator reads zero.	A. Rudder trim tab aligns with trailing edge of rudder. B. RUDDER TRIM indicator .....zero

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MAINTENANCE PRACTICES (Cont'd)

Table 201. Operational Check of Flight Control Trim Systems — S/N 380-1 thru -43 Before S/B 75-22 and S/N 370-1 and Subs (Cont'd)

PROCEDURE	NORMAL INDICATION
CAUTION ***Check that all rig pins are removed from elevator control system before operating horizontal stabilizer. This will prevent damage to elevator push rods.***	
NOTES • ***On airplanes S/N 380-38 and subs an audible warning horn will sound whenever the horizontal stabilizer trim actuators are energized.*** • ***Actuators may be operated for one complete cycle followed by 2 minutes off, or by using momentary pulses to prevent overheating actuators during check procedure. Hold switch until actuator stops running during last in. of travel, to attain full-up or full-down positions when using momentary pulses.***	
11. Check MLG is down and locked.	
12. Push and hold trim switch ..... NOSE UP (Located on left grip of pilot's control wheel.)	A. Horizontal stabilizer moves down. B. HORZ STAB indicator ..... UP
13. Push and hold trim switch ..... NOSE DN	A. Horizontal stabilizer moves up. B. HORZ STAB indicator ..... DN
14. Repeat steps 12 and 13 for co-pilot's control wheel.	
NOTE ***Refer to Checking Horizontal Stabilizer Trim System for low-speed operation of horizontal stabilizer trim system.***	
15. Circuit breakers: RH STABILIZER — NORM TRIM .....out	
16. Push and hold trim switch ..... NOSE DN (This checks left horizontal stabilizer operation.)	A. Horizontal stabilizer moves leading edge up.
17. Circuit breakers: RH STABILIZER — NORMAL TRIM .....in LH STABILIZER — NORMAL TRIM .....out	
18. Push and hold trim switch ..... NOSE UP (This checks right horizontal stabilizer actuator.)	A. Horizontal stabilizer moves leading edge down.

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MAINTENANCE PRACTICES (Cont'd)

Table 201. Operational Check of Flight Control Trim Systems — S/N 380-1 thru -43 Before S/B 75-22 and S/N 370-1 and Subs (Cont'd)

PROCEDURE	NORMAL INDICATION
19. Circuit breaker LH STABILIZER — NORMAL TRIM .....in *NOTE* ***Steps 20 thru 23 check the alternate operation of the aileron, rudder, and horizontal stabilizer trim systems.*** 20. Set trim selector switch ..... ALT (Located on ALTERNATE TRIM CONTROLS panel.) 21. Repeat steps 4 thru 6 with alternate trim switch at LEFT WING DOWN and RIGHT WING DOWN. 22. Repeat steps 8 thru 10 with alternate trim switch at LEFT and RIGHT. 23. Repeat steps 11 thru 13 with alternate trim switch at NOSE UP and NOSE DOWN. 24. Electrical power ..... off 25. Secure airplane as desired.	

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MAINTENANCE PRACTICES (Cont'd)

OPERATIONAL CHECK OF FLIGHT CONTROL TRIM SYSTEM — S/N 380-1 THRU -43 AFTER S/B 75-22 AND S/N 380- 44 AND SUBS

See table 202 for a general check of the aileron trim system, horizontal stabilizer trim system, and rudder trim system.

Table 202. Operational Check of Flight Control Trim System — S/N 380-1 thru -43 After S/B 75-22 and S/N 380-44 and Subs

PROCEDURE	NORMAL INDICATION
EQUIPMENT/TOOLS • 28 Vdc Electrical Power Supply	
1. Electrical power ..... on	
2. Circuit breakers: NORMAL TRIM (four) ..... in ALTERNATE TRIM (three) ..... in TRIM IND ..... in STAB SPEED CONT ..... in	
3. Set trim selector switch ..... ALT (Located on ALTERNATE TRIM CONTROLS panel.) This checks the alternate operation of the aileron trim system.	
4. Push and hold alternate trim switch ..... LEFT WING DOWN	A. Aileron trim tab moves down. B. AILERON TRIM indicator ..... left wing down
5. Push and hold alternate trim switch ..... RIGHT WING DOWN	A. Aileron trim tab moves up. B. AILERON TRIM indicator ..... right wing down
6. Set trim selector switch ..... NORM This checks the normal operation of the aileron trim system.	
7. Push fwd half of AILERON TRIM switch in both directions.	A. Aileron trim tab does not move. B. AILERON TRIM indicator ..... does not move
8. Push aft half of AILERON TRIM switch in both directions.	A. Aileron trim tab does not move. B. AILERON TRIM indicator ..... does not move
9. Push both halves of AILERON TRIM switch simultaneously in both directions.	A. Aileron trim tab moves in the proper direction selected. B. AILERON TRIM indicator .... move in the proper direction selected

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MAINTENANCE PRACTICES (Cont'd)

Table 202. Operational Check of Flight Control Trim System — S/N 380-1 thru -43 After S/B 75-22 and S/N 380-44 and Subs (Cont'd)

PROCEDURE	NORMAL INDICATION
CAUTION *Check that all rig pins are removed from elevator control system before operating horizontal stabilizer. This will prevent damage to elevator doors.*	
NOTE *Actuators may be operated for one complete cycle followed by 2 minutes off, or by using momentary pulses to prevent overheating actuators during check procedure. Hold switch until actuators stops running during last in. of travel. to attain full-up or full-down positions when using momentary pulses.*	
10. Set trim selector switch ..... ALT This checks the alternate operation of the horizontal stabilizer trim system.	
11. Push alternate trim switch ..... NOSE DOWN (Located on ALTERNATE TRIM CONTROLS panel.)	A. Horizontal stabilizer leading edge moves up. B. HORZ STAB indicator ..... UP
12. Push alternate trim switch ..... NOSE UP	A. Horizontal stabilizer leading edge moves down. B. HORZ STAB indicator ..... DOWN
13. Set trim selector switch .....NORM This checks the normal operation of the horizontal stabilizer trim system.	
14. Push pilot's outbd half of the horizontal stabilizer trim switch in both directions. (Located on pilot's control wheel.)	A. Horizontal stabilizer does not moves. B. HORZ STAB indicator ..... does not move
15. Repeat step 14 for inbd half of the pilot's horizontal stabilizer trim switch.	Same as step 14.
16. Push and hold both halves of the pilot's horizontal stabilizer trim switch simultaneously..... NOSE DN	A. Horizontal stabilizer leading edge moves up. B. HORZ STAB indicator ..... UP
17. Push and hold both halves of the pilot's horizontal stabilizer trim switch simultaneously..... NOSE UP	A. Horizontal stabilizer leading edge moves down. B. HORZ STAB indicator ..... DN
18. Repeat steps 16 and 17 using co-pilot's horizontal stabilizer trim switch.	

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MAINTENANCE PRACTICES (Cont'd)

Table 202. Operational Check of Flight Control Trim System — S/N 380-1 thru -43 After S/B 75-22 and S/N 380-44 and Subs (Cont'd)

PROCEDURE	NORMAL INDICATION
NOTE ***Refer to Checking Horizontal Stabilizer Trim System for low-speed operation of horizontal stabilizer trim system.***	
19. Circuit breaker: RH STABILIZER — NORMAL TRIM .....out	
20. Push and hold normal trim switch.... NOSE DN This checks the left horizontal stabilizer operation	A. Horizontal stabilizer leading edge moves up.
21. Circuit breakers: RH STABILIZER — NORMAL TRIM .....in LH STABILIZER — NORMAL TRIM .....out	
22. Push and hold normal trim switches.. NOSE UP This checks the right horizontal stabilizer operation.	A. Horizontal stabilizer leading edge moves down.
23. Circuit breaker: LH STABILIZER — NORMAL TRIM .....in	
24. Set alternate trim selector switch..... ALT (Located on ALTERNATE TRIM CONTROLS panel.)	
25. Push and hold RUD TRIM switch ..... L (left) This checks the alternate operation of the rudder trim system.	A. Rudder trim tab moves right. B. RUDDER TRIM indicator ..... L (left)
26. Push and hold RUD TRIM switch ..... R (right)	A. Rudder trim tab moves left. B. RUDDER TRIM indicator ..... R (right)
27. Set alternate trim selector switch.....NORM (Located on ALTERNATE TRIM CONTROLS panel.)	
28. Push and hold RUD TRIM switch ..... L (left) This checks the normal operation of the rudder trim system.	A. Rudder trim tab moves right. B. *RUDDER TRIM indicator* ..... L (left)
29. Push and hold RUD TRIM switch ..... R (right)	A. Rudder trim tab moves left. B. RUDDER TRIM indicator ..... R (right)
30. Operate RUD TRIM switch until indicator reads zero.	A. Rudder trim tab aligns with trailing edge of rudder. B. RUDDER TRIM indicator .....zero

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MAINTENANCE PRACTICES (Cont'd)

Table 202. Operational Check of Flight Control Trim System — S/N 380-1 thru -43 After S/B 75-22 and S/N 380-44 and Subs (Cont'd)

PROCEDURE	NORMAL INDICATION
31. Set normal and alternate trim selector switches ..... OFF	
32. Actuate all alternate and normal trim switches.	A. No trimming occurs.
33. Place trim selector switch to NORMAL.	
34. Secure airplane as desired.	

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MAINTENANCE PRACTICES (Cont'd)

OPERATIONAL CHECK OF TRIM DISCONNECT SYSTEMS — S/N 370-1 AND SUBS AND S/N 380-1 AND SUBS AFTER S/B 74-6

See table 203 for an operational check of the trim disconnect system.

Table 203. Operational Check of Trim Control Systems — S/N 370-1 and Subs and S/N 380-1 and Subs

PROCEDURE	NORMAL INDICATION
1. Electrical power ..... on	
2. Circuit breakers:	
NORM TRIM SYS .....in	
ALT TRIM SYS .....in	
NORM TRIM CONT.....in	
3. Set trim selector switch to ..... ALT	
4. Press and release TRIM EMERG DISC switch on pilot control wheel.	TRIM DISC indicator light ..... on
5. Set pilot's trim switch.....R WG DN	Aileron trim does not move.
6. Set pilot's trim switch..... NOSE DOWN	Stabilizer trim does not move.
7. Repeat steps 3 thru 6 using co-pilot's trim switch.	
8. Set rudder trim switch..... LEFT or RIGHT	Rudder trim does not move.
9. Position trim selector switch at NORM and repeat steps 4 thru 7.	
10. Set trim selector switch to OFF, then to NORM.	TRIM DISC indicator light ..... off
11. On S/N 380-38 and subs and airplanes after S/B 74-7, position pilot trim to NOSE UP or NOSE DOWN.	An audible warning is present during operation of horizontal stabilizer trim actuators.
12. Secure airplane as desired.	
13. Electrical power ..... off	

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MAINTENANCE PRACTICES (Cont'd)

STABILIZERS

Stabilizers Inspection Procedures

1. Remove access panels on fuselage adjacent to horizontal stabilizer interconnecting beam assembly and torque tube.
2. Inspect horizontal stabilizer skin for cracks, buckles, dents, corrosion, and loose rivets.
3. Inspect horizontal stabilizer attach fittings for security.
4. Inspect elevator skin for cracks, buckles, dents, corrosion and loose rivets.
5. Inspect honeycomb trailing edge for delamination and leading edge skin for cracks around balance weight attach screws.
6. Inspect elevator hinges for cracks, hinge bolts for looseness, and bonding straps for condition and security.
7. Inspect torque tube bearings installed within bushing, for cleanliness and freedom of rotation.
8. Inspect elevator drain holes for obstruction.
9. Inspect elevator inbd stub fairings, attached to fuselage, for cracks and security of attachment (airplanes S/N 370-1 thru -6, -8 and -9).
10. Inspect vertical stabilizer skin for cracks, buckles, dents, abrasions, wrinkles, and loose rivets or screws.
11. Inspect fiberglass antenna fairing for condition.
12. Inspect antennas for condition and security.
13. Inspect rudder and trim tab for security, tears, and missing rivets, and bonding straps for condition and security.
14. Inspect rudder hinges for cracks, hinge bolts for looseness, and bonding straps for condition and security.
15. Inspect rudder torque tube jo-bolts for looseness. Insert rig pin in aft cable bellcrank and push on the rudder side to side. Check the jo-bolts and movement in the torque tube. (Do not check with wrench or attempt to tighten.)
16. Inspect rudder torque tube bearing for cleanliness and freedom of rotation. (Make sure bearing race does not turn on tube.)
17. Install access panels.

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AILERON AND TAB - DESCRIPTION AND OPERATION

1. Aileron Control System (See figure 27-10-1.)

The aileron control system consists of two interconnected control wheels, in the cockpit, mechanically linked by push rods, bell cranks, cables, and cable sectors to two aerodynamically balanced ailerons. The ailerons are hinged to the outboard trailing edges of both wings. A trim tab is hinged to the left aileron only. Both ailerons are operated at the same time and in opposite directions, to provide lateral (roll) control of the airplane.

Overtravel stop bolts on the lower forward fuselage bell crank limit the travel of both aileron control wheels. Travel stop bolts on the sector at the outboard ends of the aileron control cables limit the travel of the ailerons.

The aileron push rods extend along the right side of the fuselage, near the floor of the pressurized cabin, to a pressure-seal bell crank. The pressure-seal bell crank permits the aileron controls to extend through the pressurized area without releasing cabin pressure. A push rod connects the pressure-seal bell crank to two inboard cable sectors. Aileron control cables enclosed in aluminum tubing extend along the aft side of the rear spar of both wings to outboard cable sectors. A push rod connects the outboard cable sector to a bell crank attached to the aileron.

The outboard cable sectors are utilized as aileron position indicator protractors to facilitate checking and rigging. Index marks are engraved on the outboard cable sectors. An aileron position indicator pointer is attached directly over the outboard cable sector to the wing structure on each wing. Movement of the outboard sectors moves the protractors under the pointers, indicating the positions of the ailerons.

2. Aileron Trim System (See figure 27-10-2.)

The aileron trim system consists of a trim tab, an electromechanical actuator, control switches, a trim position indicator, and a trim position transmitter.

The trim tab is hinged to the trailing edge of the left aileron. The trim tab is positioned by a push rod and a bell crank connected to the electrical trim actuator. The trim actuator is at the inboard leading edge of the left aileron. The trim actuator is operated by a pilot's trim switch and a copilot's trim switch on the aft face of the outboard grip of either control wheel, and an alternate trim switch on the alternate trim control panel. The normal and alternate trim circuits are selected by a trim selector switch. A pilot's trim arming switch and a copilot's trim arming switch are provided on the normal trim switch of either control wheel. These switches, when released, interrupt the normal trim circuit in case of runaway trim or a sticking trim switch.

The aileron trim position indicator, on the copilot's inboard instrument panel, indicates the trimmed position. The indicator is electrically positioned by a transmitter.

The transmitter (part of the actuator assembly) is internally geared to the aileron trim actuator. The aileron trim tab limits are 12 (±1/2) degrees or 57/64 (±1/32) inch in either direction from the trim tab trailing edge to aileron trailing edge.

Lateral trimming of the airplane is controlled by operating switches in the normal or alternate trim circuits. The switches direct electrical power from the 28-volt dc essential bus to the actuator that mechanically operates the aileron trim tab, on the left aileron, to the up or down position.

NOTE: On airplanes NA-380-1 through -43 not changed by Service Bulletin 75-22 and airplanes NA-370-1 and later, have the normal system aileron trim switches on outboard grip of either control wheel.

- On airplanes NA-380-1 through -43 changed by Service Bulletin 75-22, and NA-380-44 and later, has one normal system aileron trim switch located on center pedestal.

The aileron trim tab is normally operated by positioning the trim selector switch at NORMAL and pushing the five-position trim switch, on the outboard grip of the pilot's or copilot's control wheel, to L WG DN (left wing down) or R WG DN (right wing down). The trim tab can also be positioned with the trim selector switch at ALTERNATE. The alternate trim switch, on the alternate trim control panel, is moved to LEFT WING DOWN or RIGHT WING DOWN to operate the aileron trim actuator. The aileron trim actuator then mechanically operates the trim tab up or down.

The actuator also mechanically operates the integrally mounted position transmitter. The transmitter electrically operates the aileron trim position indicator. Placing either of the trim switches at RIGHT WING DOWN extends the actuator. Extending the actuator moves the trim tab up and indicates right-wing-down on the aileron trim position indicator. Releasing the trim arming switch, on either trim switch, breaks the normal trim circuit. The trim emergency disconnect switch will stop the actuator in any position. The trim arming switch will not affect the trim when the trim selector switch is at ALTERNATE.

A. Ailerons

The ailerons are of all-metal construction. The basic structure consists of stringers, conventional ribs, and a honeycomb trailing edge covered by aluminum alloy skin. The ailerons are hinged to the outboard trailing edges of the wings and are mechanically operated. Balance is obtained by weights mounted in the leading edge. The left aileron contains the aileron trim actuator and the aileron trim tab. The right aileron does not have a trim actuator and tab.

B. Aileron Trim Tab

The trim tab is hinged at two points on the left aileron and is operated by the aileron trim actuator. The trim tab is of aluminum-covered honeycomb construction. A balance weight is attached to the leading edge.

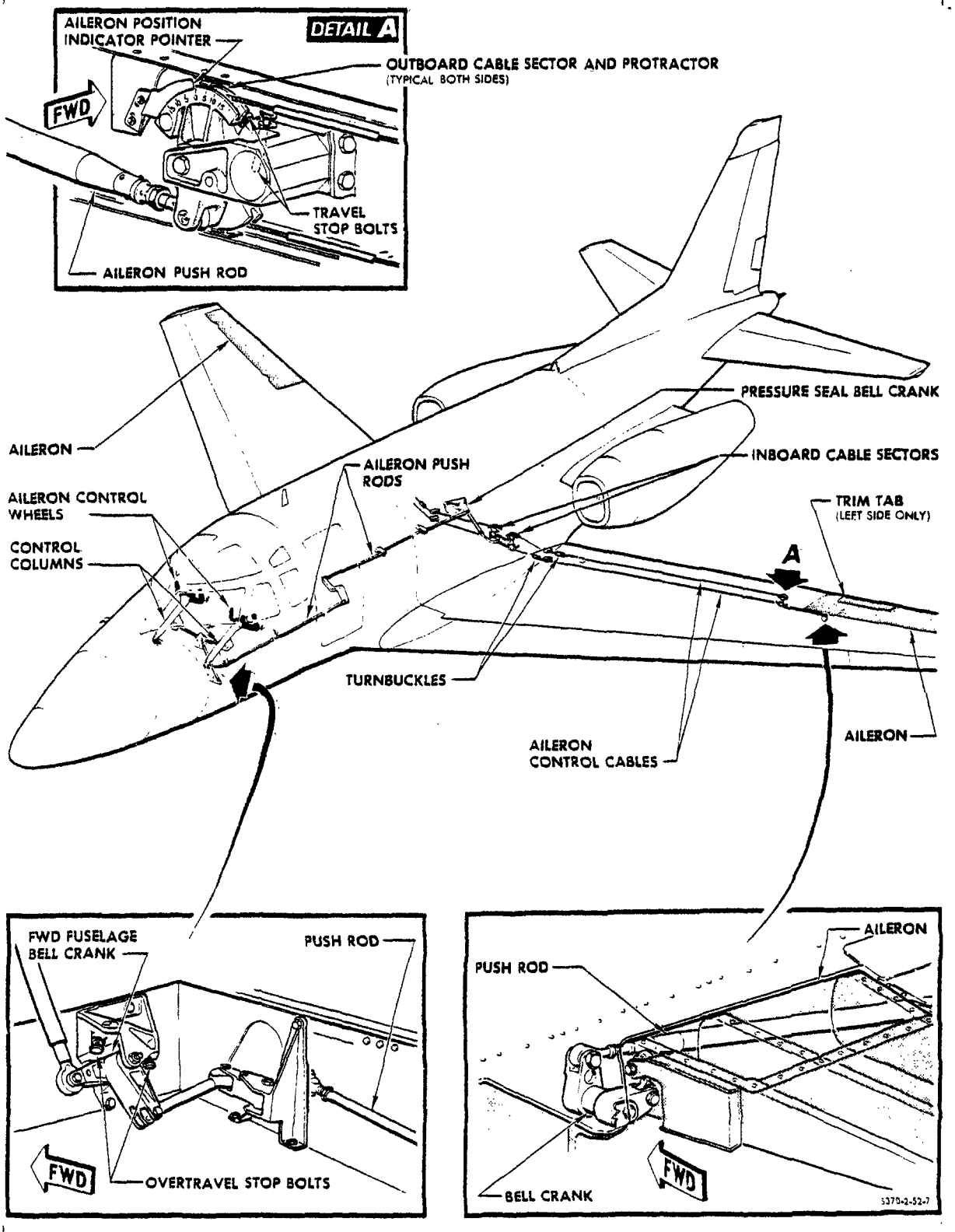
C. Aileron Trim Actuator

A linear-type, 28-volt dc electromechanical actuator operates the aileron trim tab. The aileron trim actuator is mechanically linked to the aileron trim tab by a bell crank and a push rod. The actuator is mounted near the inboard leading edge of the left aileron. The actuator has two internal travel limit switches to limit the travel of the jackscrew. Nonjamming mechanical stops are employed to restrict jackscrew travel in case of travel limit switch failure. An antirotational device is used to prevent the jackscrew end fitting from rotating. An aileron trim position transmitter (part of the actuator assembly) is internally connected to the jackscrew. Movement of the jackscrew operates the transmitter. The transmitter electrically operates the aileron trim position indicator in the cockpit. The transmitter is a potentiometer type designed to operate through 360 degrees or more of shaft rotation.

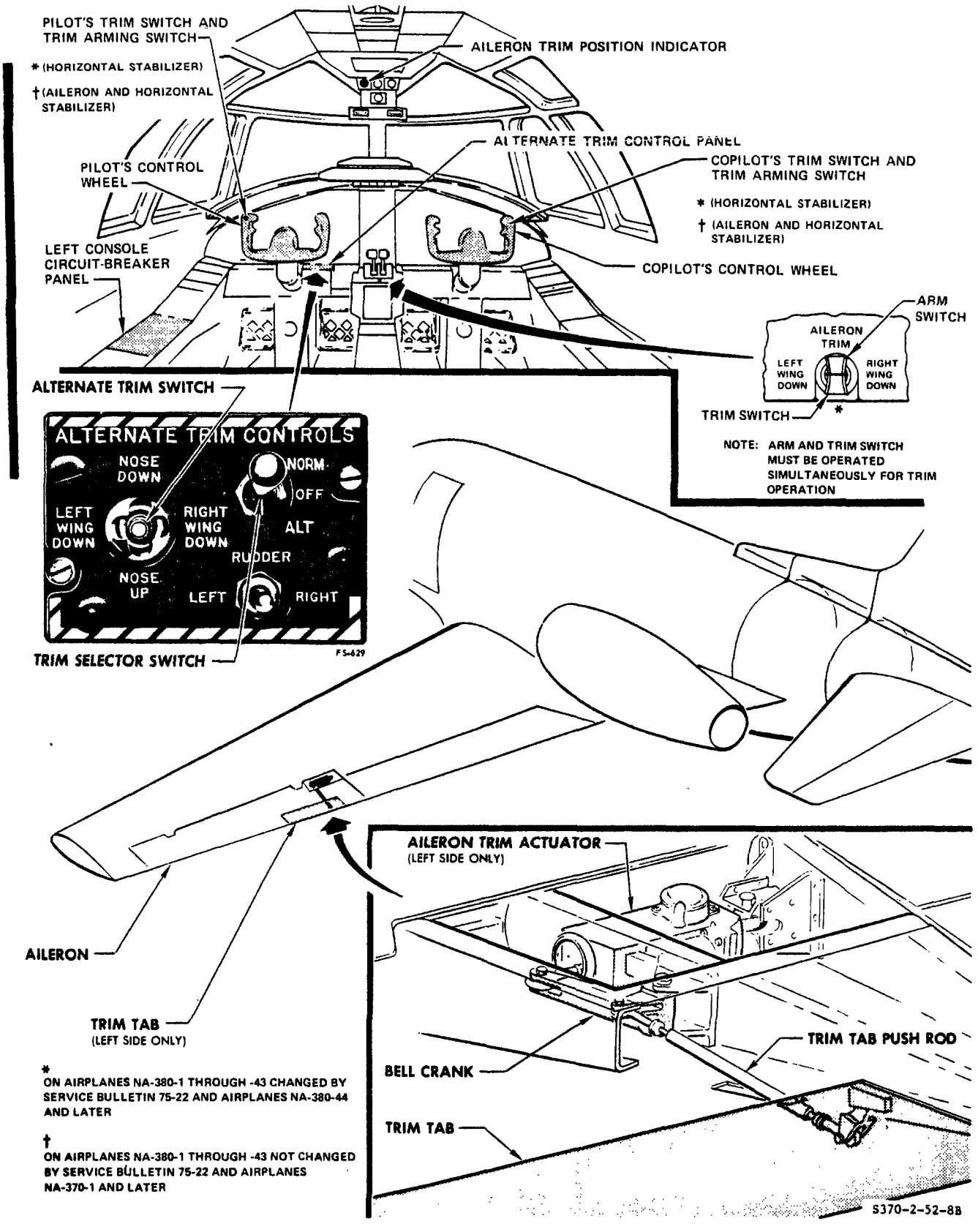
The transmitter wiper is centered on tap "B" within 1-1/2 degrees with the actuator in neutral position. Retracting the actuator produces A, B, C direction of rotation of the wiper. Movement of the wiper electrically reproduces a similar movement in the indicator.

D. Aileron Trim Position Indicator

The aileron trim position indicator, on the copilot's inboard instrument panel, is powered by the 28-volt dc essential bus. Two rotating pointers, fixed opposite each other, indicate left-wing-down and right-wing-down trim positions of the trim tab in degrees. The indicator is electrically positioned by a trim position transmitter on the aileron trim actuator. The indicator is calibrated 12 degrees wing-up or wing-down in increments of 4 degrees.



Aileron Control System
Figure 27-10-1



* ON AIRPLANES NA-380-1 THROUGH -43 CHANGED BY SERVICE BULLETIN 75-22 AND AIRPLANES NA-380-44 AND LATER

† ON AIRPLANES NA-380-1 THROUGH -43 NOT CHANGED BY SERVICE BULLETIN 75-22 AND AIRPLANES NA-370-1 AND LATER

AILERON AND TAB

MAINTENANCE PRACTICES

AILERON REPLACEMENT

Aileron Removal

NOTE

Removal procedures for LH and RH ailerons are the same except for disconnecting the trim actuator electrical plug on the left wing.

1. Disconnect aileron operating link from outbd bellcrank at aileron inbd hinge fitting.
2. Disconnect bonding wires at inbd and outbd hinges.
3. Remove two bolts from inbd hinge fitting and one bolt from outbd hinge fitting. Remove right aileron.

NOTE

The LH aileron cannot be removed until step 4 is accomplished.

4. Carefully lower inbd end of left aileron to gain access to trim actuator electrical plug.

NOTE

The trim actuator electrical plug is in the LH wing at the aileron inbd hinge station.

5. Disconnect electrical plug and remove aileron.

Aileron Installation

NOTE

Use procedures in figure 201 if the same unaltered aileron is to be installed. If a new aileron is to be installed, use steps 1 thru 4.

1. Temporarily install aileron and mark correct gap as shown in figure 201.

NOTES

- ***Spare aileron assemblies are furnished with a small amount of material on the skin ends for finish trimming. The excessive skin material will not interfere with installation.***
- ***The aileron has a honeycomb trailing edge and the ends are filled with Corefil 615. The trailing edges are pretrimmed to the correct dimension at the time of manufacture and should not require additional trimming. If additional trimming is required, the Corefil 615 may be filed. To prevent delamination, file with the skin and not across it.***

2. Remove aileron. Trim and file to mark.
3. Perform a static balance check if a new, repaired, or a newly painted aileron is being installed. (Refer to Sabreliner Structural Repair Manual, NA-66-1032.)
4. Perform procedures in figure 201.

AILERON HINGE PIN REPLACEMENT

Aileron Hinge Pin Removal

1. Remove aileron from wing.
2. Remove safetywire from outbd hinge pin and remove from outbd hinge. (See figure 203.)
3. Remove access door at inbd hinge bolt.
4. Remove safetywire from inbd hinge bolt and remove bolt from inbd hinge.

Aileron Hinge Pin Installation

1. Position inbd hinge in place and install inbd hinge bolt. (See figure 203.) Safetywire hinge bolt.
2. Position outbd hinge in place and install outbd hinge pin. Safetywire hinge pin to retainer bolt.
3. Install access door on aileron.

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MAINTENANCE PRACTICES (Cont'd)

AILERON TRIM TAB REPLACEMENT

Aileron Trim Tab Removal

NOTE

The aileron trim is on the LH aileron.

1. Disconnect trim actuator rod end from trim tab inbd hinge fitting.
2. Disconnect bonding wire at inbd hinge fitting.
3. Rotate trim tab upward and remove three screws from inbd hinge fitting and remove trim tab.
4. Attach hinge fitting to aileron to prevent loss of fitting.

Aileron Trim Tab Installation

NOTE

Perform a static balance check if a new, repaired, or a newly painted trim tab is installed. (Refer to Sabreliner Structural Repair Manual, NA-66-1032.)

See figure 202.

AILERON TRIM ACTUATOR REPLACEMENT

Aileron Trim Actuator Removal

NOTE

The aileron trim actuator is the LH aileron.

1. Remove access cover.
2. Disconnect trim actuator electrical plug.
3. Remove attaching bolts from both ends of actuator and remove actuator.

Aileron Trim Actuator Installation

See figure 203.

AILERON CONTROL SYSTEM RIGGING

See figure 204.

AILERON TRIM SYSTEM RIGGING

See figure 205.

ICEX APPLICATION

Consumables

- Dry Cleaning Solvent, P-D-680 or Equivalent
- ICEX (B.F. Goodrich Transportation Products)

Procedure

1. Install maintenance ground safety locks.
2. Remove wheel well cover in RH wheel well.
3. Remove all dirt and grease from sectors and supports from solvent.
4. Blow dry all srufaces.

WARNINGS

- ***Application of ICEX requires the use of skin and eye protection (Neoprene gloves and chemical goggles). If material accidentally contacts skin or eyes, flush immediately with plenty of clean water for 15 minutes and consult a physician if eye or skin irritation persists.***
- ***If accidentally swallowed, do not induce vomiting, see a physician immediately.***
- ***Use only in well ventilated area, avoid prolonged breathing of vapos.***
- ***Keep container tightly closed when not is use. Keep container away from open flame or electric heater.***
- ***Properly dispose of empty container. Do not burn as toxic vapors are released.***

CAUTIONS

- ***Too heavy an application will result in a sticky surface that will collect dust and dirt reducing ICEX efficiency.***
- ***Ensure ICEX does not contact sector bearings as deterioration may occur.***

5. Apply ICEX with a swab or small brush to obtain complete coverage in a single continuous back and forth motion.
6. Install wheel well cover.
7. Remove maintenance ground safety locks.
8. Secure airplane as desired.

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MAINTENANCE PRACTICES (Continued)

INSPECT AILERON CABLES — P/N 276-523006-11, 276-523008-11 and 276-523005-11 (CABLES-WITHOUT NYLON JACKETS)

Inspect the aileron cables for fraying in the area of pulley contact, just outboard of the main wheel wells. This cable inspection is applicable to the carbon steel cables and the stainless steel cables. Examine cables for broken wires by passing cloth over the cable. This will clean the cable for a visual inspection and detect broken wires if the cloth snags on the cable. The absence of snags is not positive evidence that broken wires do not exist. Bend the cable and inspect with a four power, or greater, magnifying glass in the area of pulley contact. Should 1 to 3 wires be found broken, the next scheduled inspection of the cables would be at 600 flight hours or 12 months. However, should 4 to 6 wires be found broken, the cables should be removed from the pulleys for inspection at each subsequent 100 flight hour inspection. If a cable has an excess of 6 broken wires, it must be replaced.

The four aileron cables may be inspected without removing the cables from the airplane using the following procedure.

CAUTION

If lower wing trailing edge skin has to be removed (P/N 265-130104-21 or -22) to replace a pulley or bracket, hydraulic pressure must not be applied to the airplane and normal and auxiliary accumulator pressures depleted when either skin is off the airplane. This will prevent damage to the tee-section trailing edge member (main landing gear actuating cylinder attach member).

1. Lower wing flaps.
2. Remove wheel well (dish pan) covers in LH and RH wheel wells.
3. Dump both Normal and Auxiliary hydraulic accumulator pressure.
4. Disconnect lower LH cable turnbuckle in LH wheel well.
5. Remove cable guard pin and disconnect upper LH cable from the wheel well sector.
6. Remove safetywire from LH outbd sector pivot bolt. Remove pivot bolt.
7. Disconnect the upper and lower cables from LH outbd sector.
8. Pull the upper LH cable into landing gear strut well and the lower LH cable into the wheel well.
9. Inspect pulleys for wear and/or damage.
10. Inspect cables for wear, broken strands and corrosion. Special attention should be given to the cables where they pass over the pulleys on the wing rear spar at the root ribs.
11. If cable replacement is required, refer to Aileron Cable Replacement.
12. Connect cables to LH outbd sector.
13. Install LH outbd sector pulley and pivot bolt. Tighten and safetywire pivot bolt.
14. Connect upper LH cable to inbd (wheel well) sector. Install cable guard pin.
15. Connect lower LH cable turnbuckle and adjust cable tension. Check for a "slip fit" of LH inbd and outbd sector rig pins. Safetywire turnbuckle.
16. Disconnect lower RH cable turnbuckle in RH wheel well.
17. Remove cable guard pin and disconnect upper RH cable at aileron sector in RH wheel well.
18. Pull the upper cable into RH landing gear strut well and the lower RH cable into the RH wheel well.
19. Inspect pulleys for wear and/or damage.
20. Inspect cables for wear, broken strands and corrosion. Special attention should be given to the cables where they pass over the pulleys on the wing rear spar at the root ribs. If cable replacement is required, refer to Aileron Cable Replacement.
21. Connect upper RH cable to aileron sector in RH wheel well.
22. Connect lower RH cable turnbuckle and adjust cable tension. Check for "slip fit" of inbd and outbd sector rig pins. Safetywire turnbuckle.
23. Check entire aileron cable installation for security and safeties.
24. Install aileron sector and cable guards in LH and RH wheel wells.
25. Rotate pilot's control wheel clockwise to its full-travel position: right aileron trailing edge should move up; left aileron trailing edge should move down. Check that aileron position indicator protractor on outbd cable sector of each wing reads 15-1/2 (± 1/2)°.
26. Rotate pilot's control wheel counterclockwise to its full-travel position: right aileron trailing edge should move down; left aileron trailing edge should move up. Protractor reading should be the same as in step 25.
27. Install wheel well (dish pan) covers in LH and RH wheel wells.
28. Airplane ..... secure as desired.

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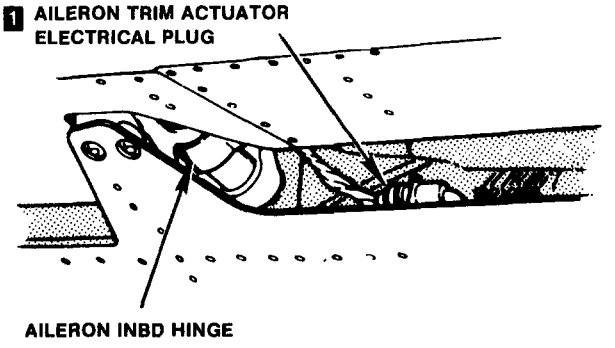
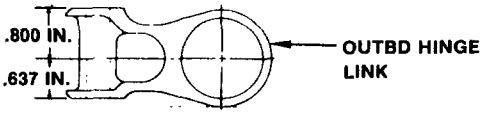
NOTES

The LH and RH aileron installation procedures are the same, except the connection of the aileron trim actuator electrical plug for the LH aileron.

- *Ensure outbd hinge link is properly installed if it was removed or misalignment of the aileron with the wing mold line will result.*
- *Ensure that aileron well is free of loose tools and hardware.*

1. Position aileron. (LH aileron only — connect aileron electrical plug to receptacle in LH wing at aileron inbd hinge station.)
2. Install one bolt in outbd and two bolts in inbd hinge fittings.





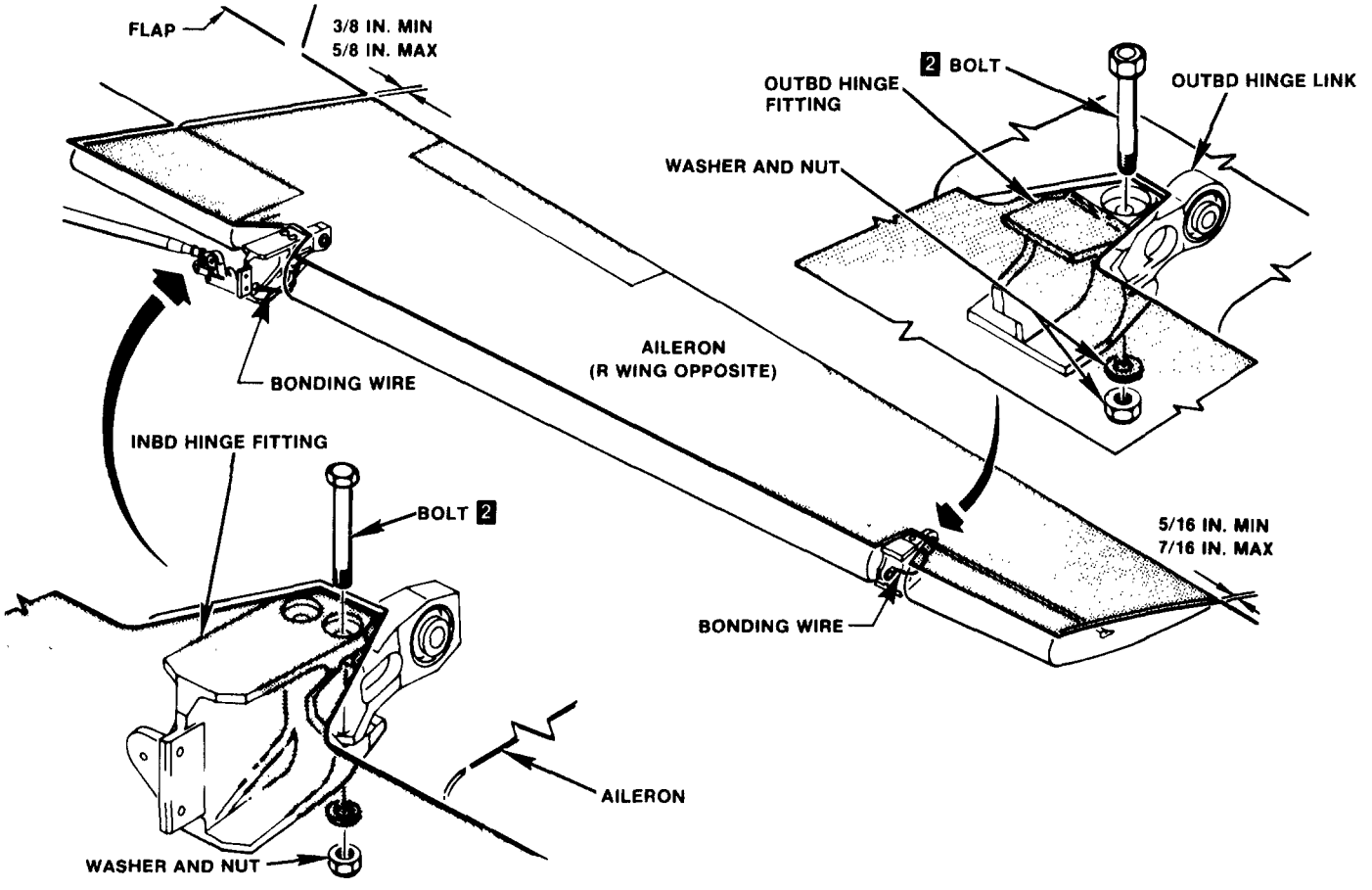


Figure 201 — Aileron Installation (Sheet 1 of 2)

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3 Connect outbd bellcrank to aileron operating link at aileron inbd hinge fitting.

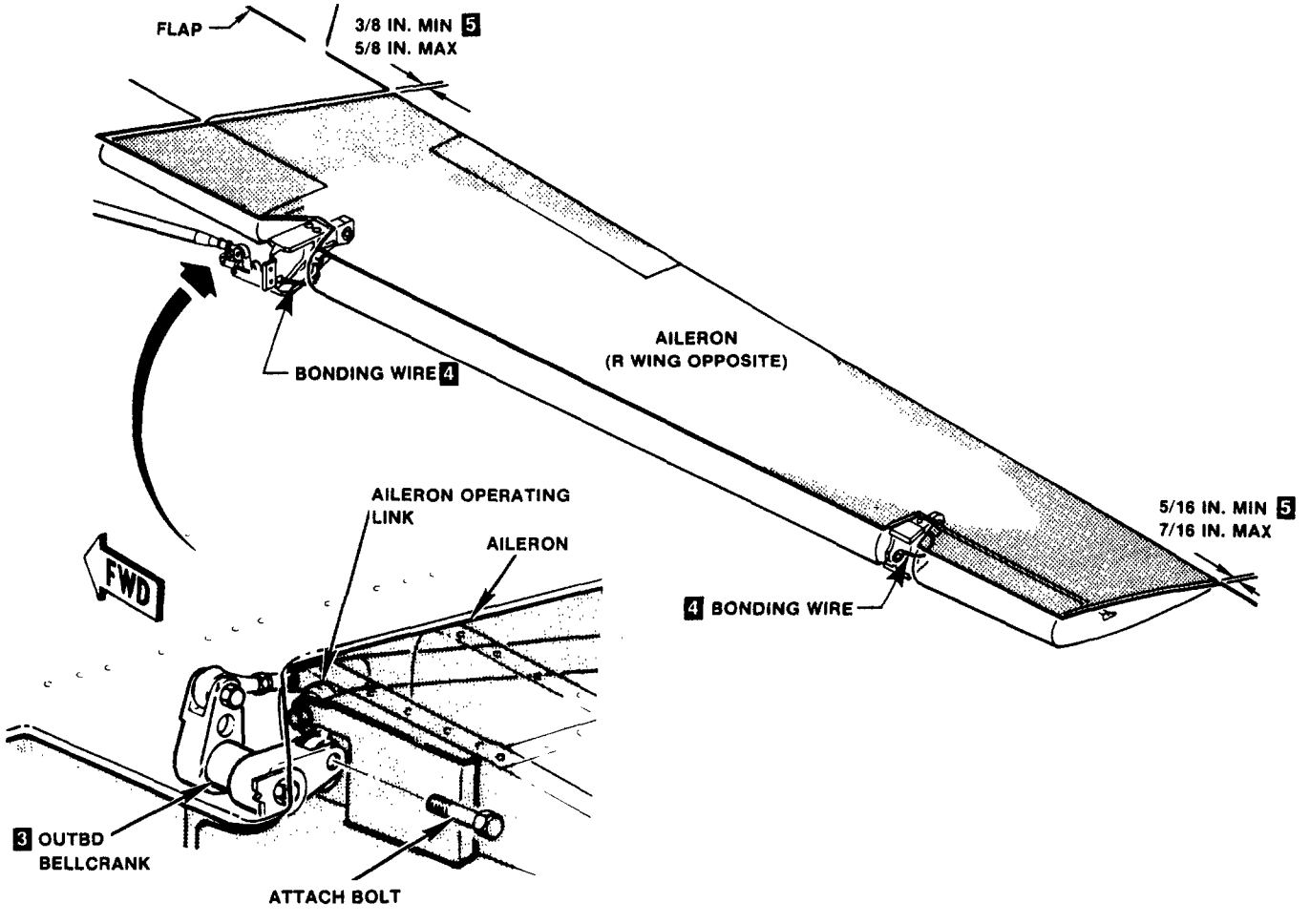
4 *Connect bonding wires at inbd and outbd hinges.*

5 Ensure ailerons are at rig position and check clearance between aileron and flap and between outbd edge of aileron and wing. Obtain clearances by trimming aileron.

6 Perform procedures in Checking Aileron Control System.

7 Perform procedures in Checking Aileron Trim System.

8 Inspect all connections. Check that area is free of tools, rags, and hardware.



Aileron Installation
Figure 201 (Sheet 2 of 2)

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1 Remove inbd hinge fitting from trim tab. Insert inbd hinge fitting into its pivot on aileron inbd side.

2 Insert trim tab outbd end into its pivot on aileron. Slip trim tab inbd end into alignment with inbd hinge fitting and install the three screws attaching hinge fitting to trim tab. Rotate trim tab for access to install screws.

NOTE

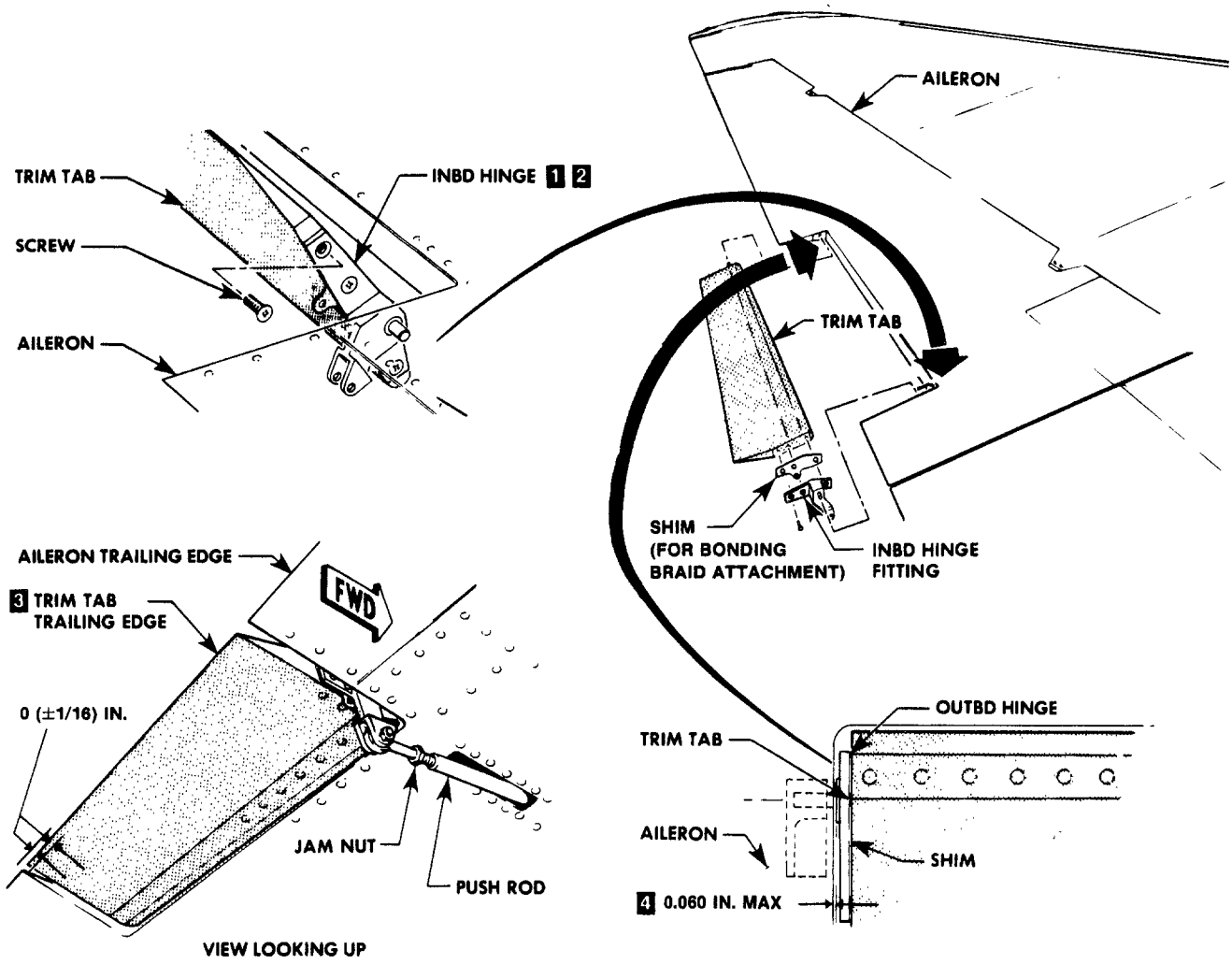
Replace trim tab if the outbd trailing edge is more than ±1/16 in. out of alignment with the aileron trailing edge.

3 Align trim tab inbd trailing edge with aileron trailing edge. Check that trim tab trailing edge is aligned ±1/16 in. with aileron trailing edge.

NOTE

Make shims from 2024-T42 aluminum alloy and the same shape as the outbd hinge.

4 Check inbd-outbd movement does not exceed 0.060 in. max. Add shims between outbd hinge and trim tab to bring movement within tolerance.



Aileron Trim Tab Installation
Figure 202 (Sheet 1 of 2)

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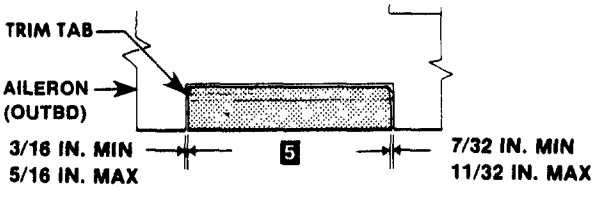
5 Align trim tab with aileron and check clearances between trim tab and aileron. Trim as necessary.

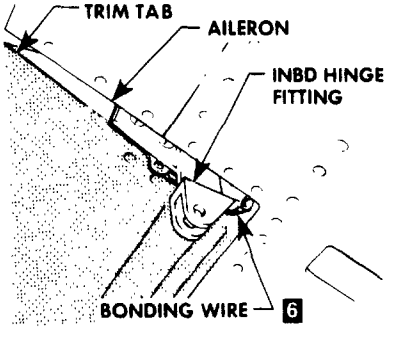
6 Connect bonding wire at inbd hinge fitting to aileron.

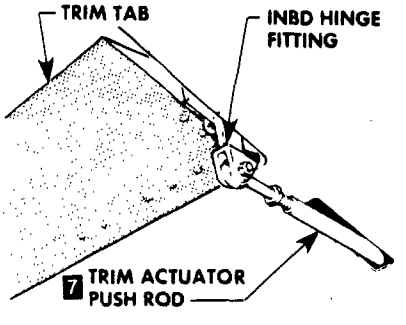
7 Connect trim actuator push rod end to inbd hinge fitting. Install bolt and washer, and tighten nut. Install cotter pin.

8 Operate trim tab through its full range of travel. There should be no evidence of binding.

9 Check trim tab. (*Refer to Checking Aileron Trim System.*)







Aileron Trim Tab Installation
Figure 202 (Sheet 2 of 2)

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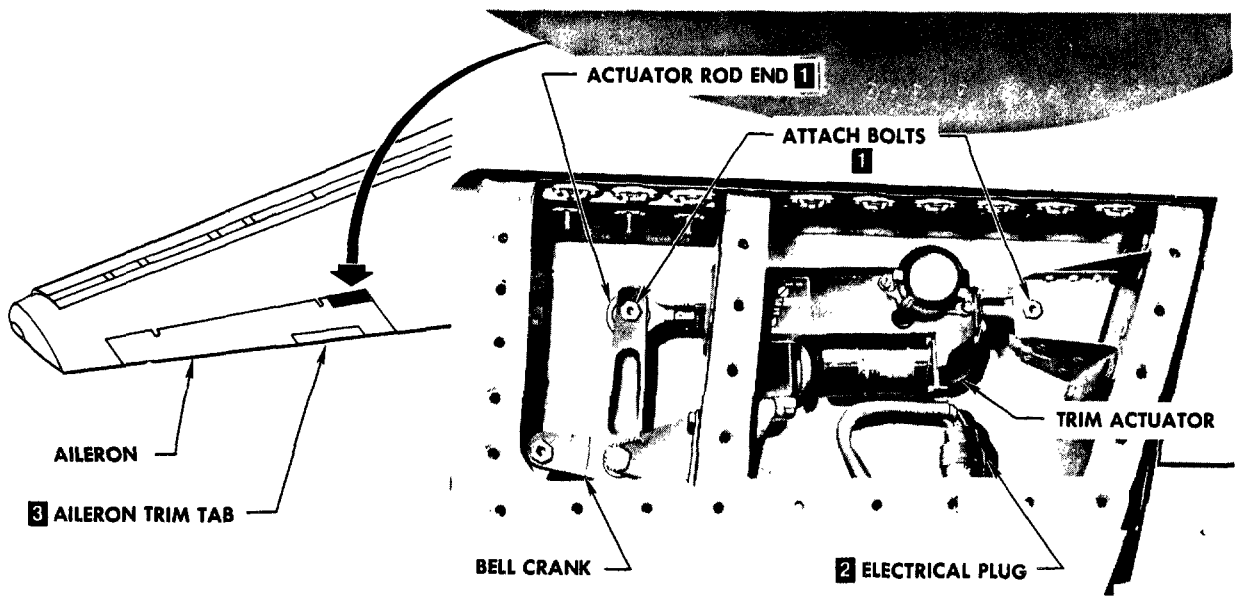
1 Position trim actuator with rod end on outbd side toward bellcrank and install attach bolts.

2 Connect actuator electrical plug, and safetywire.

3 Check trim tab.

4 Ensure area is free of rags, tools, and loose hardware.

5 Secure airplane as desired.



MAJOR CHANGE

Aileron Trim Actuator Installation
Figure 203

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NOTE

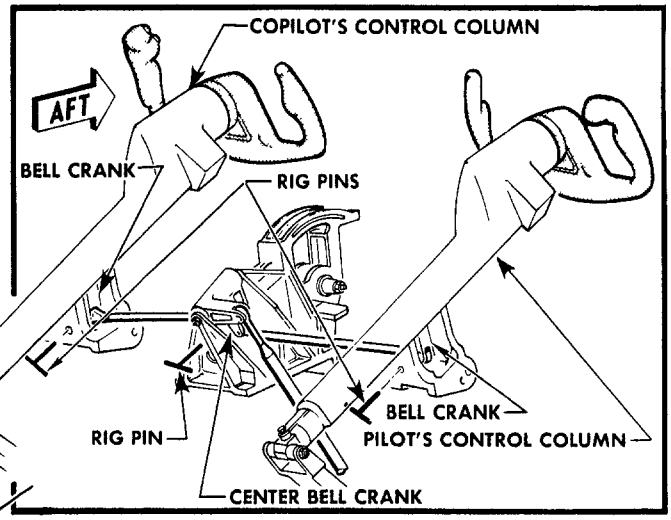
Use 5/16-in. diameter rods or bolts as rig pins. Prevent forced alignment of rods or bellcranks by maintaining a "drop fit" on all rig pins.

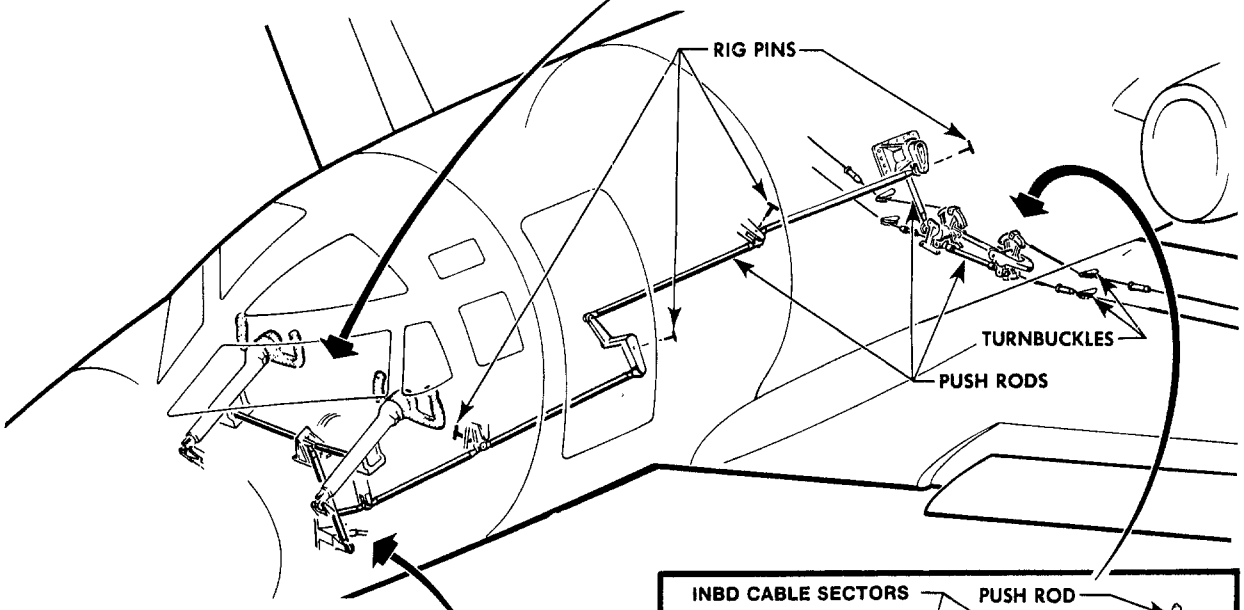
1 Move gust lock handle to the disengaged position; then insert rig pins in pilot's and co-pilot's control column bellcranks.

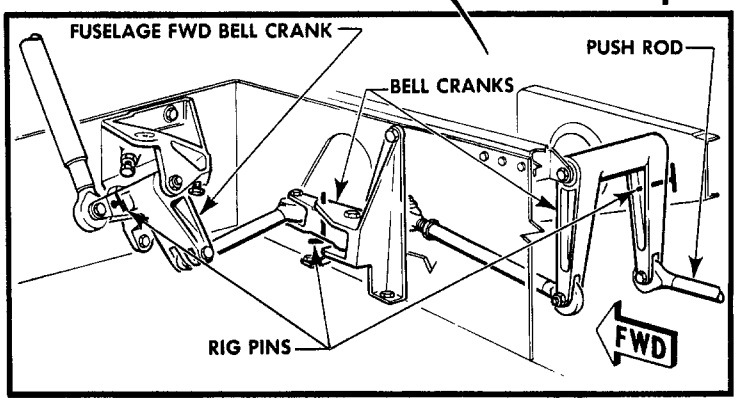
2 Insert a rig pin in center bellcrank, between pilot's and co-pilot's control columns. Adjust both LH and RH push rods until rig pin has a "drop fit." Lock jam nuts on rod-end fittings.

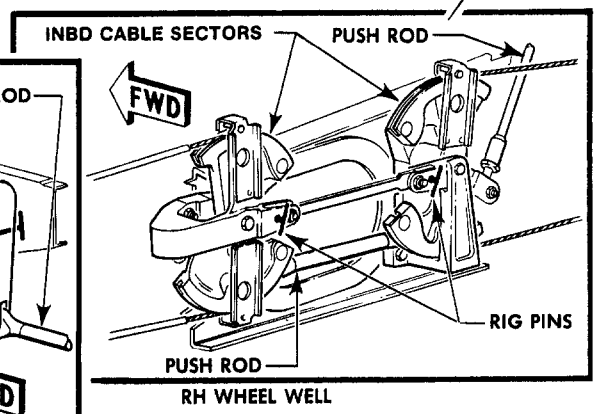
3 If push rods are not properly adjusted or if they are being replaced, insert rig pins in remaining aileron control system bellcranks. Adjust push rods until rig pins have a "drop fit." Lock jam nuts on rod-end fittings.

4 Insert rig pins in inbd cable sectors. Adjust push rods until rig pins have a "drop fit." Lock jam nuts on rod-end fittings.









Rigging Aileron Control System
Figure 204 (Sheet 1 of 3)

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5 To gain access to aileron control cables and outbd cable sectors, operate flaps to down position. To prevent raising flaps, pull LH FLAP and RH FLAP circuit breakers on left console circuit-breaker panel.

6 Insert rig pin in outbd cable sector in LH wing. Rig pin should have a "drop fit," and cable tension should be 250 (±25) lbs at 70° F. (Refer to Cable Tension Variation With Temperature chart in this chapter.) Tension can be checked near the outbd side of the flap actuator. Alternately adjust turnbuckles until previously stated cable tension is obtained. Turnbuckle adjusting nut should be not less than 1/8 in. from edge of turnbuckle. Safetywire turnbuckles.

7 Check that aileron position indicator on outbd cable sector is set at zero degrees. Adjust position indicator pointer, to obtain this setting. Secure screws, and safetywire.

8 Ensure that cable sector rig pins are installed. In order to properly rig ailerons, they must be adjusted with respect to the wing reference plane. The outbd wing fixed trailing edges cannot be used as reference, since they are not accurately held to the wing reference plane during the fabrication process.

9 Determine wing reference plane and record angle.

10 Disconnect outbd aileron push rod (outbd end) attach bolt.

11 Place an inclinometer on a straightedge, as shown in detail A, on aileron between aileron trailing edge and heel along first row of rivets outbd of inbd edge.

12 Measure angle of straightedge. This angle must be 3° (±30') greater than wing reference angle for required 30' aileron up trailing edge rig position.

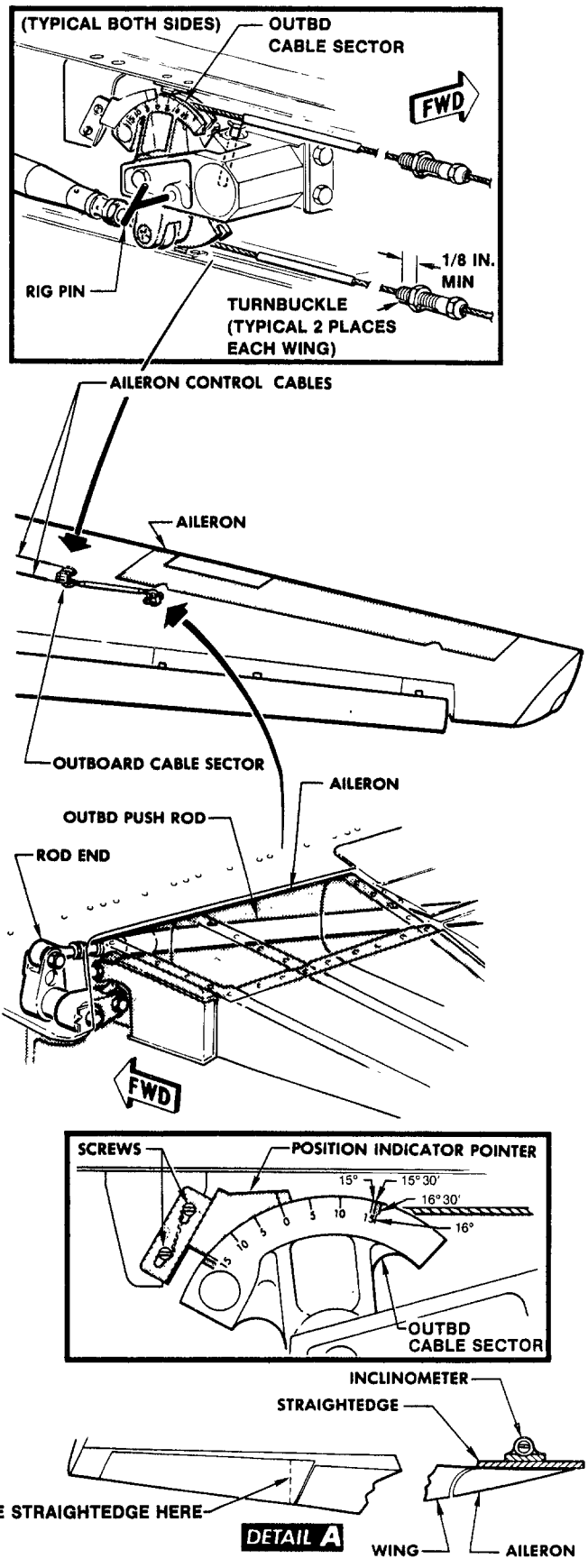
13 Hold aileron to rig position and adjust outbd push rod.

14 Adjust outbd push rod in the following manner:

- **A** Loosen outbd push rod inner jam nut.
- **B** Adjust outbd push rod to obtain a "drop fit" of outbd attach bolt. (This is accomplished by rotating entire rod assembly.)
- **C** Replace outbd push rod attach bolt and tighten.
- **D** Tighten inner jam nut and lock.

15 Repeat steps 6 thru 14 on right wing.

16 Remove all rig pins.



Rigging Aileron Control System
Figure 204 (Sheet 2 of 3)

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17 Rotate pilot's control wheel cw to is full-travel position; RH aileron trailing edge should move up; LH aileron trailing edge should move down. Trailing edge of each aileron should be within the dimensional limits shown in detail B. Check that aileron position indicator protractor on outbd cable sector of each wing reads 15°30' (±30'). Obtain this setting of adjusting travel stop bolts on support assembly of each outbd cable sector.

NOTE

At full travel, simultaneous striking of LH and RH aileron travel stops need not occur.

18 Rotate pilot's control wheel ccw to its full-travel position: RH aileron trailing edge should move down; LH aileron trailing edge should move up. Results should be as in step 11. Secure jam nuts, and safetywire stop bolts.

19 Apply a force of 20 lbs to pilot's control wheel in a cw direction. Maintain this force while checking clearance of overtravel stop bolt. The overtravel stop bolt is on the support fitting of the aileron control system fwd bellcrank. Clearance between overtravel stop bolt and bellcrank should be 0.005 to 0.032 in. Adjust overtravel stop bolt, to obtain this clearance. Secure jam nut.

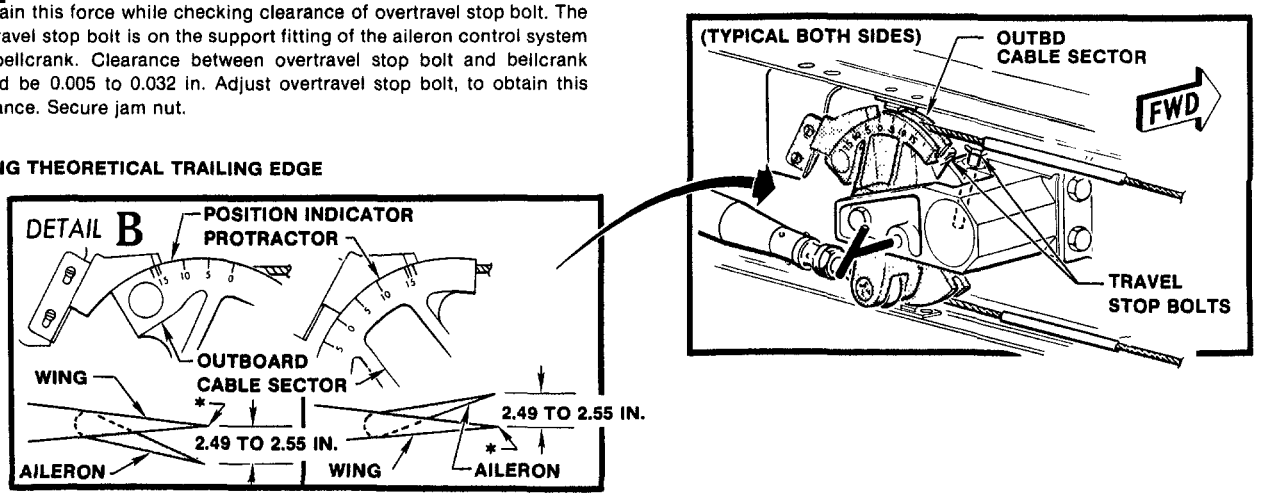
20 Repeat step 19; however, apply a force of 20 lbs to pilot's control wheel in a **ccw** direction. Adjust remaining overtravel stop bolt, to obtain 0.005 to 0.032-in. clearance. Secure jam nut.

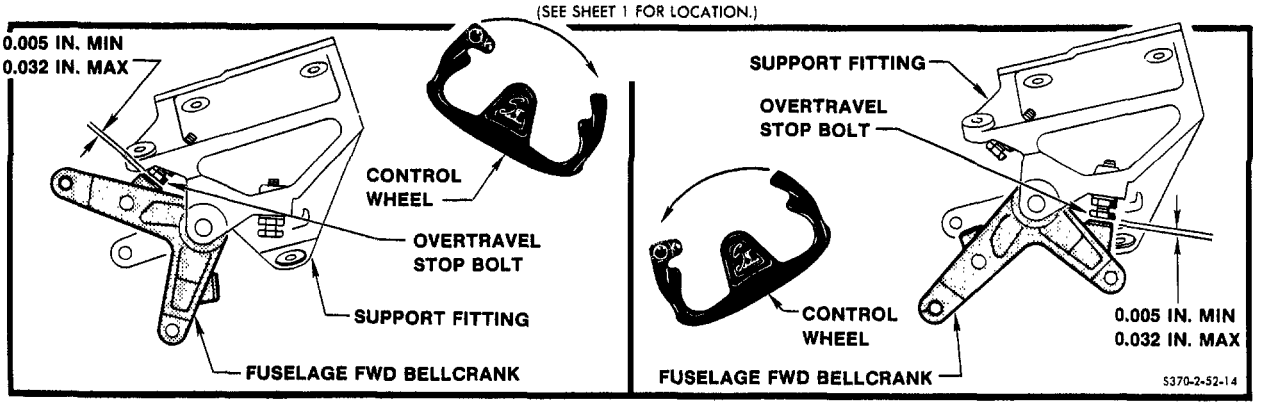
NOTE

Steps 17 thru 20 may be performed on the co-pilot's control wheel as well as the pilot's control wheel.

21 Do procedures in Checking Aileron Control System.

22 Thoroughly inspect aileron control system. Check that all moving parts do not bind or touch structure while system is operated through its full range of travel. Check that all jam nuts are secured. Check that all bolts and turnbuckles are properly safetied. Be sure that areas are free of tools, rags, and loose hardware. Install access doors.





Rigging Aileron Control System
Figure 204 (Sheet 3 of 3)

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NOTE

The object of this procedure is to synchronize, within tolerances and through the full range of travel, the aileron trim tab trailing edge with the travel of the aileron trim position indicator needle. The total travel of the aileron trim tab trailing edge should be equally divided between the full up and full down positions from the aileron trailing edge.

1 Connect 28 vdc utility external power source to airplane.

2 Set trim selector switch, on ALTERNATE TRIM CONTROL panel, to NORM.

3 Operate trim switch, on outbd grip of pilot's or co-pilot's control wheel (**A**) on center pedestal (**B**), until aileron trim position indicator reads zero degrees.

4 Check that trim tab trailing edge is aligned, within ±1/32 in., with aileron trailing edge. If it is not, adjust push rod until tab trailing edge and aileron trailing edge align within this tolerance.

5 Move trim switch to L WG DN. Hold switch at this position until aileron trim tab stops moving.

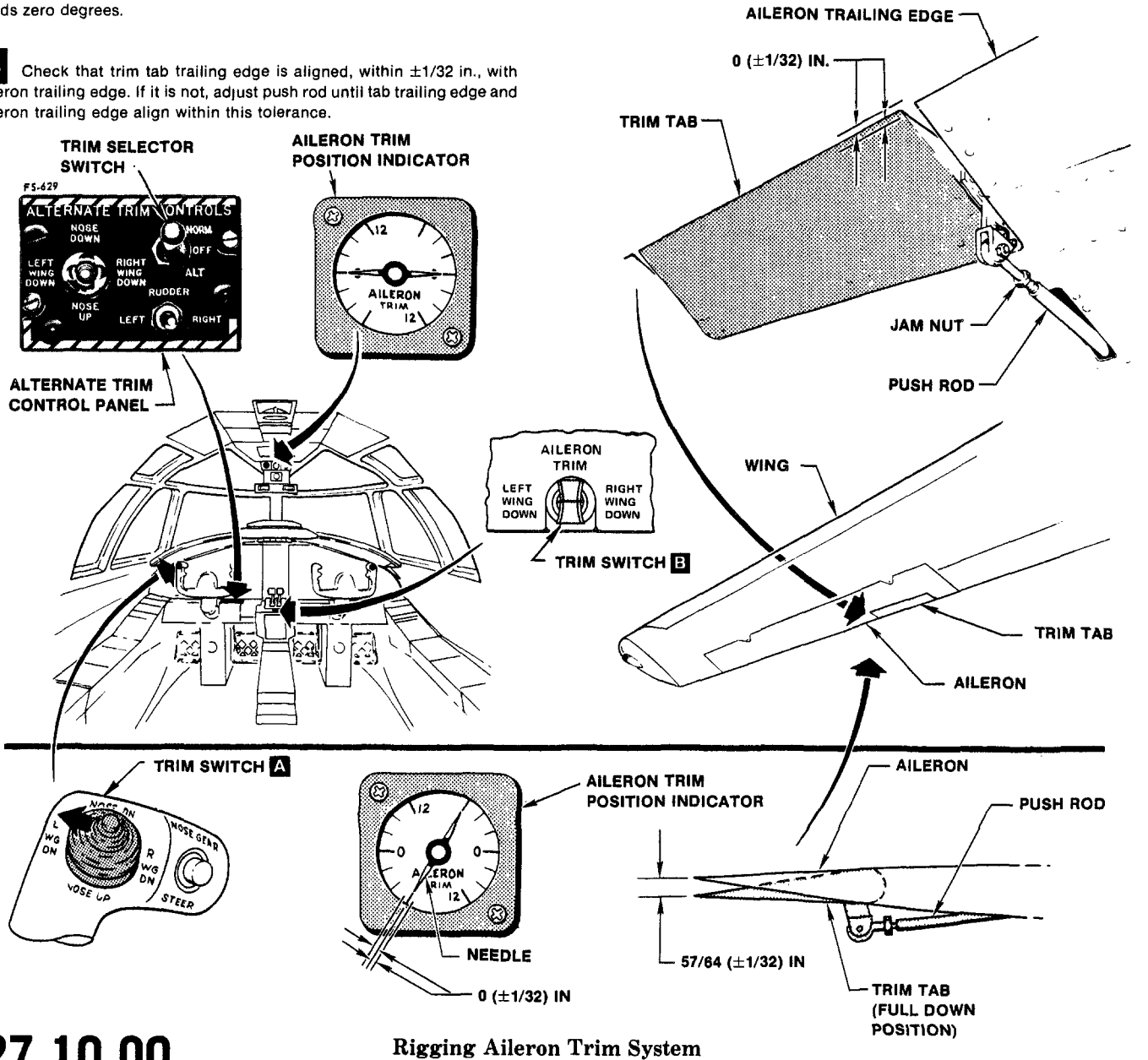
6 Check the full travel position of the aileron trim position indicator needle. The needle should align, within ±1/32 in., with the 12° left-wing-down index line. If indicator needle does not align within this tolerance, refer to Aileron Trim System in Trouble Shooting Flight Control Trim Systems.

7 Check that trim tab trailing edge is 57/64 (±1/32) in. down from aileron trailing edge. If it is not, adjust push rod until it is.

EFFECTIVITIES

A *S/N 380-1 thru -43 before S/B 75-22 and S/N 370-1 and subs.*

B *S/N 380-1 thru -43 after S/B 75-22 and S/N 380-44 and subs.*



Rigging Aileron Trim System
Figure 205 (Sheet 1 of 2)

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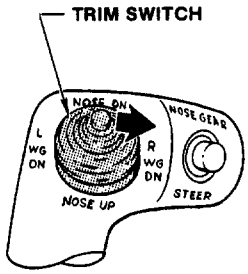
8 Move trim switch to a R WG DN. Hold switch at this position until aileron trim tab stops moving.

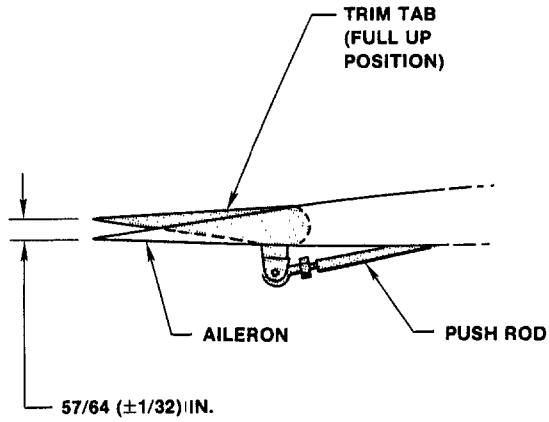
9 Check that aileron trim position indicator needle aligns, within ±1/32 in., with the 12° right-wing-down index line. If indicator needle does not align within this tolerance, refer to Aileron Trim System in Trouble Shooting Flight Control Trim Systems.

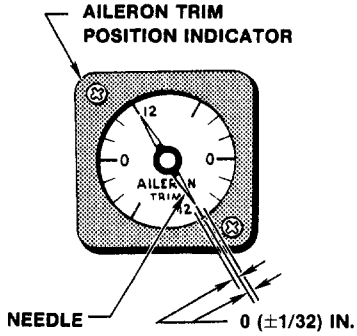
10 Check that trim trailing edge is 57/64 (±1/32) in. up from aileron trailing edge. Readjust push rod as necessary, to equally divide total travel of trim tab.

11 Repeat steps 3 and 4. Secure and safety jam nut on push rod.

12 Do procedures in Checking Aileron Trim System.







Rigging Aileron Trim System
Figure 205 (Sheet 2 of 2)

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MAINTENANCE PRACTICES (Cont'd)

Aileron Cable Inspection — P/N 276-523005-21, 276-523006-31, and 276-523008-31 (Cables With Nylon Jackets)

CAUTION

If lower wing trailing edge skin (P/N 265-130104-21 or -22) is removed to replace a pulley or bracket, hydraulic pressure must not be applied to the airplane and normal and auxiliary accumulators must be depleted. This will prevent damage to tee-section trailing edge member (main landing gear actuating cylinder attach member).

NOTE

It is recommended to replace all pulleys that have groove wear before cable installation.

1. Lower wing flaps.
2. Remove wheel well covers (dishpans).
3. Inspect LH (upper and lower) aileron cables per steps A thru E.
   A. Disonnect lower LH cable turnbuckle in LH wheel well. (See figure 27-10-406.)
   B. Remove cable guard pin and disconnect upper LH cable from the wheel well sector.
   C. Remove safetywire from LH outbd sector pivot bolt. Remove pivot bolt.
   D. Disconnect the upper and lower cables from LH outbd sector.
   E. Pull the upper LH cable into landing gear strut well and the lower LH cable into the wheel well.
4. Inspect cables with special attention given to the cable-to-pulley contact area per steps A thru H.
   A. Inspect pulleys for wear and/or damage.
   B. Clean cables specifically in the area of cable-to-pulley contact.
   C. Using a 4-power (or greater) magnifying glass, inspect the nylon jacket and/or cables for cracks, broken sections, worn areas, peeling, bubbles and a max of five broken wires. If any one of these conditions is present, the cables shall be re-inspected at end of each six months of operation.
   D. If any one of the following conditions exists, the cable shall be replaced:
      (1) Complete circumferential crack or separation of the nylon jacket.
      (2) Fifty percent (50%) or more bare cable is showing due to a worn or split nylon jacket.
      (3) Six or more broken wires.
   E. If cable replacement is required, refer to Aileron Cable Replacement.
   F. Connect cables to LH outbd sector and install outbd sector pulley and pivot bolt. Tighten and safetywire pivot bolt.
   G. Connect upper LH cable to wheel well sector. Install cable guard pin.
   H. Connect lower LH cable turnbuckle and adjust cable tension. Check for "slip fit" of LH inbd and outbd sector rig pins. Safetywire turnbuckle.
5. Inspect RH (upper and lower) aileron cables per steps A thru H.
   A. Disconnect lower RH cable turnbuckle in RH wheel well.
   B. Remove cable guard pin and disconnect upper RH cable at aileron sector in RH wheel well.
   C. Pull the upper cable into RH main landing gear strut well and the lower RH cable in the RH wheel well. Inspect cables. (Refer to step 4.)
   D. If cable replacement is required, refer to Aileron Cable Removal and Replacement.
   E. Connect upper RH cable to aileron sector.
   F. Connect turnbuckle and adjust cable tension.
   G. Check for "slip fit" to inbd and outbd sector rig pins.
   H. Safetywire turnbuckle.
6. Check entire installation for security and safeties.
7. Rotate pilot's control wheel cw to its full-travel position: RH aileron trailing edge should move up; LH aileron trailing edge should move down. Check that aileron position indicator protractor on outbd cable sector of each wing reads 15°30′ (±30′).

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MAINTENANCE PRACTICES (Continued)

Aileron Cable Inspection — P/N 276-523005-21, 276-523006-31, and 276-523008-31 (Cables With Nylon Jackets) (Cont'd)

8. Rotate pilot's control wheel ccw to its full-travel position: RH aileron trailing edge should move down; LH aileron trailing edge should move up. Protractor reading should be the same as in step 7.
9. Install wheel well covers.
10. Raise wing flaps and secure airplane.

AILERON CABLE REMOVAL AND REPLACEMENT

Aileron Cable Removal

NOTE

It is recommended that all pulleys having cable groove wear be replaced.

1. Lower wing flaps.
2. Remove wheel well covers.
3. Remove LH or RH (upper and lower) aileron cables per steps A thru E.
   A. Disconnect upper and lower turnbuckles. (See Figure 206.)
   B. Remove cable guard pins (one only on RH installation) and disconnect cables from the wheel well sector.
   C. Remove safetywire from LH or RH outbd sector pivot bolt. Remove pivot bolt.
   D. Disconnect the upper and lower cables from LH or RH outbd sector.
   E. Remove cable pulleys from wing rear spar and remove cables from airplane.

Aileron Cable Replacement

1. Position cables in airplane.
2. Connect cables to applicable outbd sector and install outbd sector and pivot bolt. Tighten and safetywire pivot bolt.
3. Connect LH or RH cable to wheel well sector. Install cable guard pin(s).
4. Install cable pulleys on wing rear spar.
5. Connect upper and lower cable turnbuckles and adjust cable tension. (See Figure 201.)
6. Rig aileron control system. (See Figure 204.)
7. Perform operational check of aileron control system.
8. Install wheel well covers.

CONTROL WHEEL REMOVAL AND INSTALLATION

Control Wheel Removal

1. Remove medallion retaining screw. The screw is reached through a hole in the underside of the control wheel.
2. Using a sharp instrument, lift out lower edge of medallion; then slide medallion out of place.
3. Remove retaining screws in wiring disconnect plug.
4. Cut safetywire and remove wheel retaining nut.
5. Pull wheel from shaft and carefully work wheel over electrical disconnect plug.

Control Wheel Installation

1. Position control in neutral position, then place wheel on shaft in a level attitude.
2. Install thin washer and nut. Torque nut to 100-150 in.-lb.
3. Remove laminations from the Control Column Adjustment Shim (See IPC NA-69-420, Figure 10-15 for location) as required to obtain .0005 to .0020 inch backlash between the two Control Column Bevel Roll Control Gears.
4. Connect wiring disconnect plug and install lock screws.
5. Install medallion and retaining screw.
6. Perform operational check of aileron and horizontal stabilizer trim systems.

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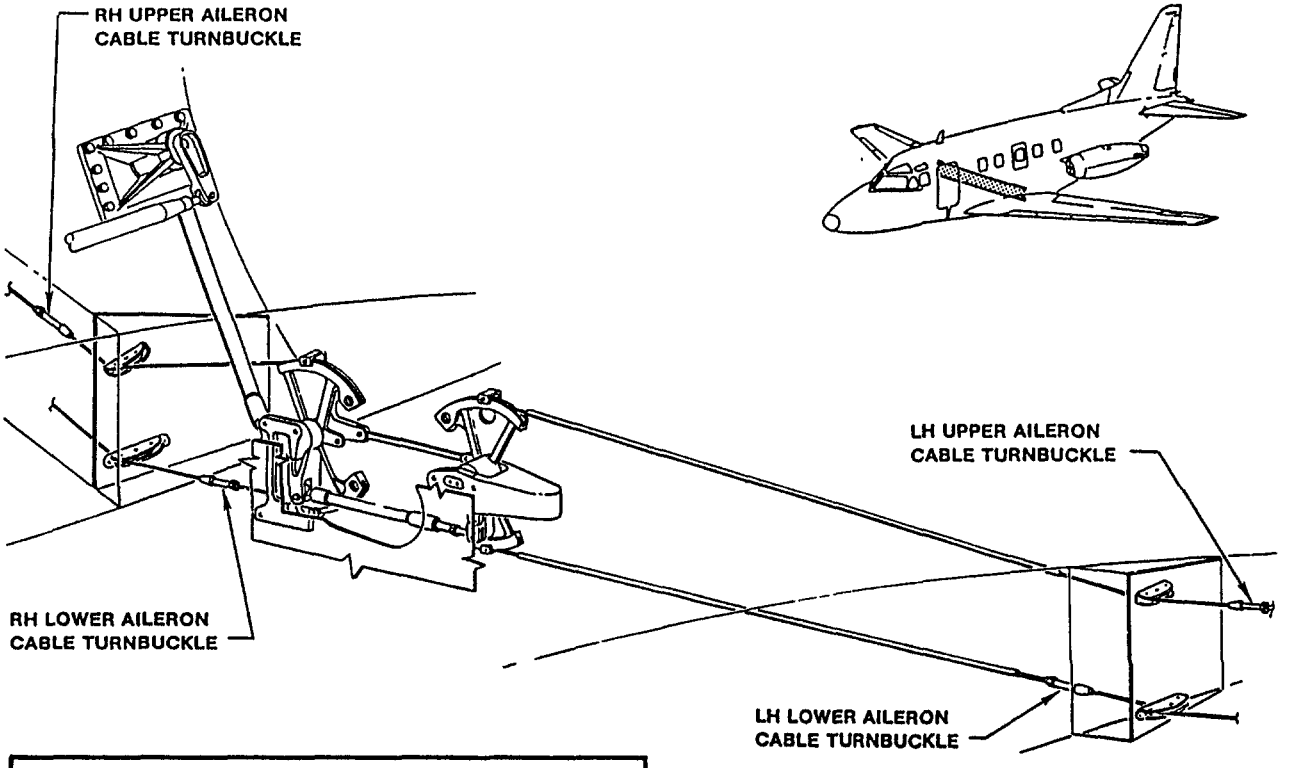

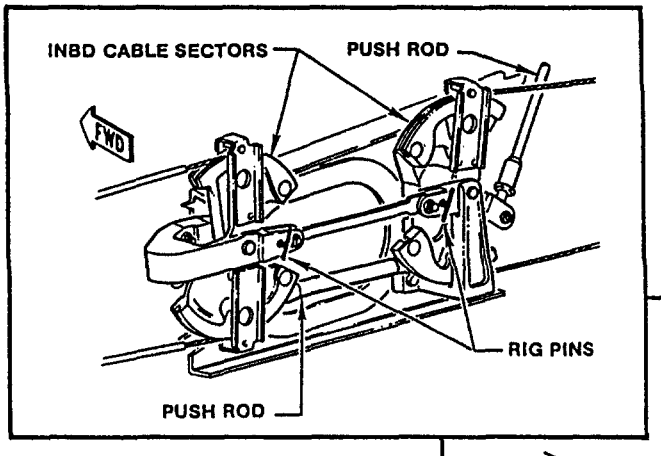

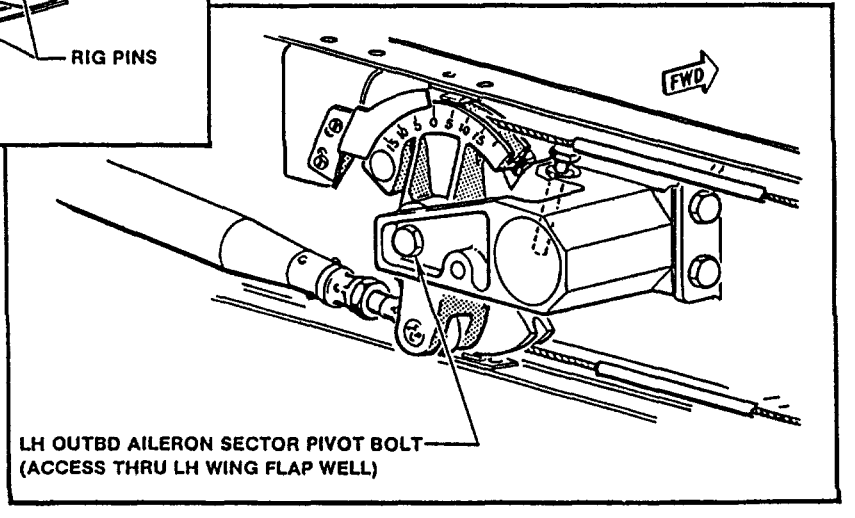

Aileron Cable Inspection
Figure 206

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AILERON AND TAB SYSTEM - TROUBLE SHOOTING

1. Trouble Shooting Aileron Control and Aileron Trim Systems

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
	AILERON CONTROL SYSTEM	
A. CONTROLS DO NOT MOVE.		
(1) Gust lock mechanism in engaged position.	Check elevator controls and rudder controls. If they cannot be moved, gust lock handle is engaged.	Disengage gust lock handle by moving handle down and forward from engaged position.
(2) Mechanical obstruction in control system linkage.	Check control system linkage to determine cause of mechanical interference. Check especially at bell cranks for rig pins.	
B. CONTROLS MOVE, BUT SURFACE DOES NOT FOLLOW.		
(1) Push rods, cables, or bell cranks disconnected.	Check that control system linkage is not disconnected at any point.	Reconnect linkage and secure jam nuts.
(2) Cable disconnected or broken.	Lower flaps and check cables.	Reconnect or replace cable.
C. FULL SURFACE TRAVEL CANNOT BE OBTAINED.		
(1) Improper adjustment of aileron stop bolts or overtravel stop bolts.	Refer to adjustment procedure in figure 27-10-404.	
(2) Mechanical interference or binding in control system.	Thoroughly inspect control system to determine cause of interference or binding.	Make certain that control system has adequate clearance from all structure, plumbing, wire bundles, etc.
(3) Improperly rigged aileron control system.	Lower flaps, and check aileron position indicators. If travel is short in one direction and if there is overtravel in opposite direction, linkage is probably misrigged.	Rerig aileron actuating linkage. (See figure 27-10-404.)

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
D. CONTROLS BIND MOMENTARILY DURING OPERATION.		
(1) Mechanical interference in control system.	While moving controls slowly back and forth over point where binding occurs, check entire system to determine cause of interference.	Make certain that control system has adequate clearance from all structure, plumbing, wire bundles, etc.
E. FRICTION FORCES TOO HIGH.		
(1) Cable tension too high.	Check cable tension.	Adjust cable tension. (See figure 27-00-201.)
(2) Mechanical interference in system.	While slowly moving controls back and forth, inspect entire system, including control wheels, to ensure that it is properly installed and that it has adequate clearance.	Make certain that control system has adequate clearance from all structure, plumbing, wire bundles, etc.
AILERON TRIM SYSTEM		
F. TRIM SYSTEM INOPERATIVE.		
(1) Circuit breakers not pushed in.	Check "NORMAL TRIM" and "ALTERNATE TRIM" circuit breakers on left console circuit-breaker panel.	
(2) Trim selector switch positioned incorrectly.	Check trim selector switch position on alternate trim control panel.	
(3) Electrical system faulty.	Perform continuity check.	Repair electrical system.
G. TRIM THROWS UNSATISFACTORY.		
(1) Trim system improperly adjusted.	Check rigging of trim system. (See figure 27-10-405.)	
(2) Mechanical interference in linkage.	Remove access 6. Operate trim actuator back and forth while inspecting linkage, especially at bell crank, for mechanical interference.	

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
(3) Faulty trim actuator.	Check trim tab travel from neutral to right and left extreme positions. (Measure from tab trailing edge to aileron trailing edge.) If travel is less than 7/8 inch or more than 59/64 inch either side of neutral, actuator is faulty.	Replace faulty actuator.
(4) Faulty normal trim switch or electrical wiring.	Position TRIM SELECTOR switch at NORMAL and hold trim and arm switches at L WG DN. Check for 28-volt dc at terminal (A3) or 1 and ground indication at terminal (C3) or 9 on K535 relay at right hand station 150. (Relay assembly 125). Hold trim and arm switches at R WG DN. Check for 28-volt dc at terminal (B3) or 8 and ground indication at terminal (C3) or 9 on D535 relay at right-hand station 150. If voltage is not available at either terminal, the switch is faulty or faulty electrical wiring. NOTE: Terminal numbers not in parenthesis are for airplanes NA-370-1 and -7, and NA-380-1 and later. Terminal numbers with parenthesis are for airplanes NA-370-2 through -6, -8, and -9.	Replace trim switch or repair electrical wiring.
(5) Faulty alternate trim switch or electrical wiring.	Position TRIM SELECTOR switch at ALT and hold trim switch at LEFT WING DOWN. Check for 28-vdc at terminal (A1) or 5 on K535 relay at right hand station 150 (relay assembly 125). Hold trim switch at RIGHT WING DOWN. Check for 28-vdc at terminal (B1) or 4 on K535 relay. If voltage is not available at either terminal, the switch is faulty or faulty electrical wiring.	Replace alternate trim switch or repair electrical wiring.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
H. TRIM POSITION INDICATOR INOPERATIVE.		
(1) Electrical system faulty.	Perform continuity check.	Repair electrical system.
(2) Faulty transmitter on trim actuator.	Check transmitter. If transmitter does not meet check-out requirements, a faulty actuator is indicated.	Replace actuator.
(3) Faulty trim position indicator.	If there is continuity in electrical system and transmitter meets check-out requirements, a faulty trim position indicator is indicated.	Replace trim position indicator.

1. Operational Check-out of Aileron Control System

A. Before check-out, the aileron control system must be properly rigged. (See figure 27-10-404.) If check-out does not meet requirements, refer to trouble shooting procedures.

(1) With gust lock disengaged, move aileron control wheels through full travel and check whether they operate freely. There should be no sign of binding.

(2) Rotate aileron control wheels clockwise. The right aileron trailing edge should go up. The left aileron trailing edge should go down.

(3) Repeat procedure with aileron control wheels rotated in opposite direction. The right aileron trailing edge should go down. The left aileron trailing edge should go up.

(4) Check aileron travel positions at aileron position indicators on outboard cable sectors of both wings as follows:

(a) To gain access to aileron outboard cable sectors, operate flaps to down position.

(b) To prevent raising flaps, pull "LH - FLAP" and "RH - FLAP" circuit breakers on left console circuit-breaker panel. Hold aileron trailing edges level with wing theoretical trailing edge. The aileron position indicator on each outboard cable sector should read 0 (±1/2) degrees.

(c) Rotate aileron control wheel clockwise to mechanical stop and check full travel of ailerons; then rotate aileron control wheel counterclockwise to mechanical stop and check full travel of ailerons. The position indicators should read 15-1/2 (±1/2) degrees in the full travel position.

2. Checking Aileron Control System Friction Forces

A. Using an accurate-reading push-pull type scale and a flexible tape rule, check friction forces as follows:

(1) Rotate pilot's control wheel to full counterclockwise position.

(2) Attach scale to one handle of control wheel.

(3) By slowly, steadily, and continuously pulling on scale, move control wheel clockwise.

(4) Record scale readings at one-inch increments from 1/2 inch of start position to 1/2 inch of full-travel position. Record all pull forces of wheel as "A" and all push forces of wheel as "B." Record forces in a vertical row.

(5) Rotate pilot's control wheel to full clockwise position and repeat check in opposite direction.

(6) By slowly, steadily, and continuously pushing on scale, move control wheel counterclockwise.

(7) Record all pull forces of wheel as "A" and all push forces of wheel as "B". Record forces at same points in a vertical row from bottom to top, next to first row of forces.

(8) Compute friction forces as follows: For positions where force recorded in one direction is "B" and in opposite direction is "A", add these two forces. Then divide by two. Where both forces are either "B" or "A", subtract smaller force from larger force. Then divide by two. The results will be friction forces. Friction forces at any position must not exceed 3 pounds.

3. Checking Aileron Trim System - Airplanes NA-380-1 Through -43 Not Changed By Service Bulletin 75-22 and Airplanes NA-370-1 and Later

A. Before check-out, the aileron trim system must be properly rigged. (See figure 27-10-405). If trim system does not meet check-out requirements, refer to trouble shooting procedures.

(1) Connect 28-volt dc utility external power source to airplane.

(2) Push in the following circuit breakers on left console circuit-breaker panel:

- "NORMAL TRIM - SYSTEM"
- "ALTERNATE TRIM - SYSTEM"
- "NORMAL TRIM - CONT"
- "TRIM IND"

(3) Place trim selector switch, on alternate trim control panel at NORMAL.

(4) Engage trim arm switch and move trim switch, on left grip of pilot's control wheel, to L WG DN. Hold until aileron trim tab stops moving. The aileron trim tab should go down. The needle of the aileron trim position indicator should align with ±1/32 inch of the 12-degree left-wing-down index line. Check that trailing edge of trim tab is 57/64 (±1/32) inch down from trailing edge of aileron.

(5) With trim arm switch engaged, hold trim switch at R WG DN until aileron trim tab stops moving. The aileron trim tab should go up. The needle of the aileron trim position indicator should align within ±1/32 inch of the 12-degree right-wing-down index line. Check that trailing edge of trim tab is 57/64 (±1/32) inch up from trailing edge of aileron.

(6) Operate trim switch until aileron trim position indicator reads zero degrees. Check that trailing edge of trim tab is aligned with trailing edge of aileron within ±1/32 inch.

(7) Repeat procedure on copilot's control wheel. Engage trim arm switch and hold trim switch at L WG DN. The aileron trim tab should go down. The aileron trim position indicator should read left-wing-down 12 degrees.

(8) With trim arm switch engaged, hold trim switch at R WG DN. The aileron trim tab should go up. The aileron trim position indicator should read right-wing-down 12 degrees.

(9) Check pilot's trim arming switch, on trim switch of control wheel, as follows:

(a) Push in trim arming switch and push trim switch in either direction.

(b) When trim tab starts to move, release trim arming switch while maintaining trim switch position. The trim tab should stop moving.

(10) Repeat step (9) on copilot's control wheel.

(11) Position trim selector switch, on alternate trim control panel, to ALTERNATE.

(12) Repeat steps (7) and (8) with alternate trim switch, on alternate trim control panel.

(13) Check timing of aileron trim actuator. Use a stop watch to check time it takes for aileron trim actuator to operate from one extreme trim position to other extreme trim position (24 degrees). The trim rate should not be less than 1.27 degrees per second (about 18.80 seconds total travel time) and no more than 2.8 degrees per second (about 8.6 seconds total travel time).

(14) Disconnect 28-volt dc external electrical power from airplane.

4. Checking Aileron Trim System - Airplanes NA-380-1 Through -43 Changed By Service Bulletin 75-22 and NA-380-44 and Later

A. Before check-out, the aileron trim system must be properly rigged. (See figure 27-10-405). If trim system does not meet check-out requirements, refer to trouble shooting procedures.

(1) Connect 28-volt dc utility external power source to airplane.

(2) Push in the following circuit breakers on left console circuit breaker panel:

- "NORM TRIM CONT"
- "ALTERNATE TRIM-SYSTEM"
- "NORM TRIM - SYSTEM"
- "TRIM IND"

(3) Place trim selector switch, on alternate trim control panel at NORMAL.

(4) Actuate trim arm switch and trim switch simultaneously on center pedestal to LEFT WING DOWN. Hold until aileron trim tab stops moving. The aileron trim tab should go down. The needle of the aileron trim position indicator should align with ±1/32 inch of the 12-degree left-wing-down index line. Check that trailing edge of trim tab is 57/64 (±1/32) inch down from trailing edge of aileron.

(5) With trim arm switch engaged, hold trim switch at RIGHT WING DOWN until aileron trim tab stops moving. The aileron trim tab should go up. The needle of the aileron trim position indicator should align within ±1/32 inch of the 12-degree right-wing-down index line. Check that trailing edge of trim tab is 57/64 (±1/32) inch up from trailing edge of aileron.

(6) Operate trim switch until aileron trim position indicator reads zero degrees. Check that trailing edge of trim tab is aligned with trailing edge of aileron within ±1/32 inch.

(7) Check trim arming switch on center pedestal, as follows:

(a) Actuate the trim and arming switches simultaneously in either direction.

(b) When trim tab starts to move, release trim arming switch (forward switch) while maintaining trim switch position. The trim tab should stop moving.

(8) Position trim selector switch, on alternate trim control panel, to ALTERNATE.

(9) Check timing of aileron trim actuator. Use a stop watch to check time it takes for aileron trim actuator to operate from one extreme trim position to other extreme trim position (24 degrees). The trim rate should not be less than 1.27 degrees per second (about 18.80 seconds total travel time) and no more than 2.80 degrees per second (about 8.60 seconds total travel time).

(10) Disconnect 28-volt dc external electrical power from airplane.

INSPECTION/CHECK (Continued)

INSPECT — AILERON SONIC DOUBLERS

Equipment/Tools

- None

Consumables

- None

Inspection

See Figure 27-10-601 for visual inspection procedure.

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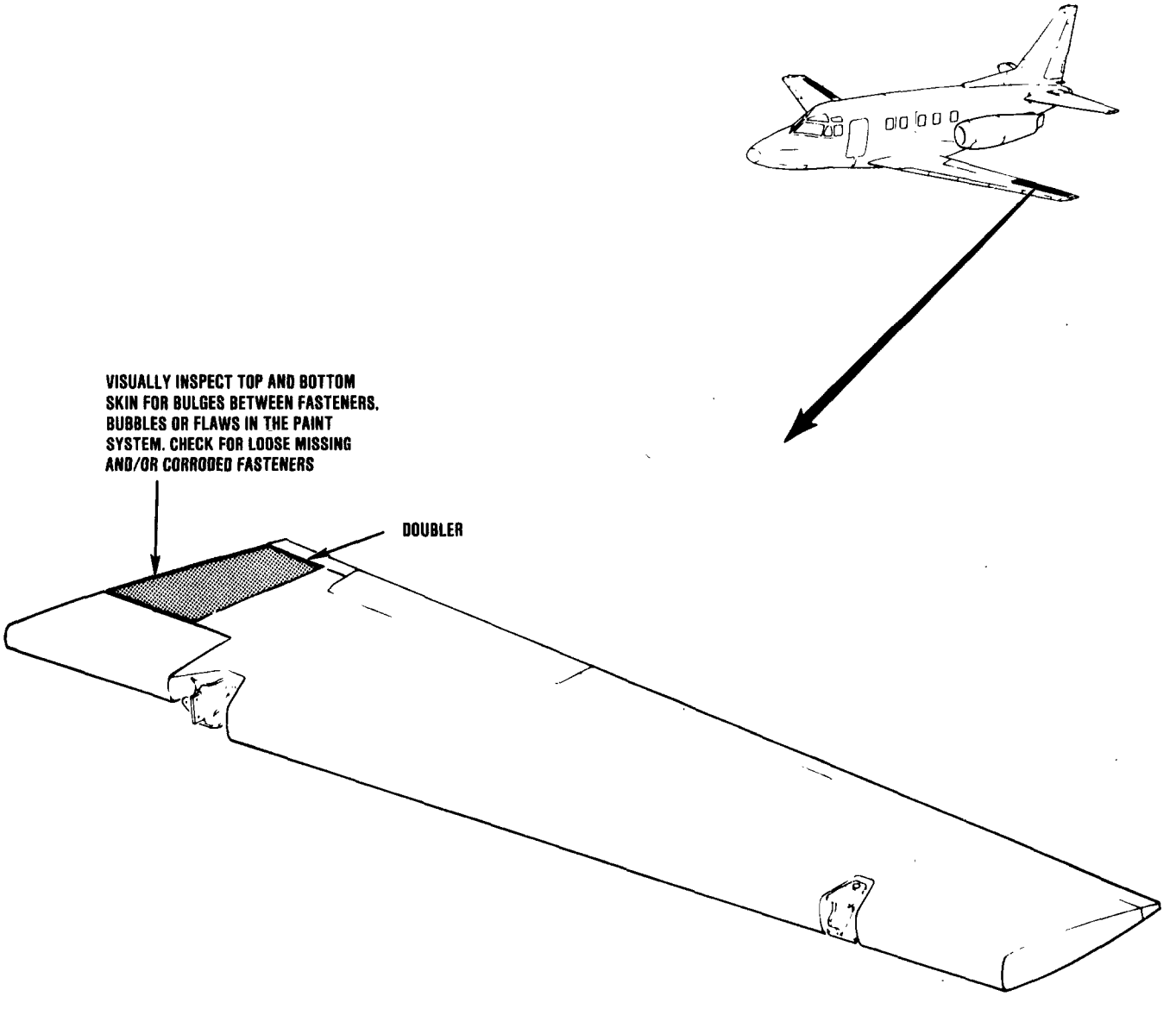
VISUALLY INSPECT TOP AND BOTTOM SKIN FOR BULGES BETWEEN FASTENERS, BUBBLES OR FLAWS IN THE PAINT SYSTEM. CHECK FOR LOOSE MISSING AND/OR CORRODED FASTENERS
DOUBLER

27-10-601 — Inspect — Sonic Doublers — Ailerons

NEW

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RUDDER AND TAB - DESCRIPTION AND OPERATION

1. Rudder Control System (See figure 27-20-1.)

The rudder control system consists of two pairs of interconnected rudder pedals and a centering bungee mechanically linked through push rods, bell cranks, and cables to the rudder. Movement of the rudder is limited by travel stop bolts in the rudder torque tube support.

The rudder push rods extend along the right side of the fuselage, near the floor of the pressurized cabin, to a pressure-seal bell crank. The pressure-seal bell crank permits the rudder controls to extend through the pressurized area without releasing cabin pressure. A push rod connects the pressure-seal bell crank to a forward cable bell crank. Rudder control cables extend along the right side of the fuselage rear section from the forward cable bell crank to the aft cable bell crank. A push rod connects the aft cable bell crank to the rudder horn attached to the rudder surface torque tube. A pilot force bungee is connected to the aft cable bell crank to center the rudder pedals. The pilot's and copilot's rudder pedal position may be adjusted to comfortable leg-length by rotating the respective pedal adjustment knob on either control pedestal. Clockwise rotation of the adjustment knobs moves pedals forward; counterclockwise rotation of the adjustment knobs moves pedals aft.

2. Rudder Trim System (See figure 27-20-2.)

The rudder trim system consists of a trim tab, an electromechanical actuator, control switches, a trim position indicator, and a trim position transmitter. The trim tab is hinged to the trailing edge of the rudder. The trim tab is positioned by a push rod and a bell crank connected to the electrical trim actuator. The trim actuator is in the lower leading edge of the rudder. The trim actuator is operated by a rudder trim switch on the aft face of the center pedestal, and an alternate trim switch on the alternate trim control panel. The normal or alternate trim circuits are selected by a trim selector switch.

Directional trimming of the airplane is controlled by manually operating switches in the normal or alternate trim circuits. The switches direct electrical power from the 28-volt dc essential bus to the actuator that mechanically operates the rudder trim tab to the left or right position. (See figure 27-00-1.) The rudder trim tab is normally operated by positioning the trim selector switch at NORMAL and pushing the three-position trim switch, on the aft face of the center pedestal, to L (left) or R (right). The trim tab can also be positioned with the trim selector switch at ALTERNATE. The alternate trim switch, on the alternate trim control panel, is moved to LEFT or RIGHT to operate the rudder trim actuator. The rudder trim actuator then mechanically operates the trim tab to the left or right.

The rudder trim position indicator, on the copilot's inboard instrument panel, indicates the trimmed position. The indicator is electrically positioned by a transmitter. The transmitter (part of the actuator assembly) is

internally geared to the rudder trim actuator. The rudder trim tab limits are 15 (±1/2) degrees or 1-1/8 (±1/32) inches in either direction from trim tab trailing edge to rudder trailing edge.

The actuator also mechanically operates the intergrally mounted position transmitter. The transmitter electrically operates the rudder trim position indicator. Placing the rudder trim switch at R or the alternate trim switch at RIGHT extends the actuator. Extending the actuator moves the trim tab left and indicates rudder right on the rudder trim position indicator.

A rudder position indicator (a built-in rigging fixture), near the end of the rudder surface torque tube, is provided as an aid during checking and rigging procedures. A rudder position indicator protractor is attached to the rudder surface torque tube support. A rudder position indicator pointer is attached to the rudder torque tube directly over the protractor. Movement of the rudder surface torque tube moves the pointer along the protractor, indicating the position of the rudder.

A. Rudder

The rudder is of all-metal construction. The basic structure consists of stringers, conventional ribs, and a honeycomb trailing edge covered by stressed aluminum alloy skin. The rudder is hinged to the trailing edge of the vertical stabilizer and is mechanically operated. Balance is obtained by weights mounted in the leading edge. The rudder contains the rudder trim actuator and the rudder trim tab.

B. Rudder Pedals

Two identical sets of rudder pedals (pilot's and copilot's) are provided to operate the rudder and brakes. The rudder pedals control the rudder action through direct mechanical linkage to the rudder. The wheel brakes are actuated by pressure on the top of the rudder pedals. The pedals are attached to the hangers by a short tubular arm. The hangers are suspended from pivot points on the pilot's and copilot's control column. This arrangement allows the rudder pedals to move forward and aft, enabling rudder control, and to pivot around the tubular arm, enabling brake control. The rudder pedals may be adjusted to the correct leg length.

C. Rudder Pedal Adjustment Knob

Comfortable leg-length adjustment of the rudder pedals may be accomplished by a rudder pedal adjustment knob on the control pedestal directly below the control wheels. Counterclockwise turning of the knob adjusts the pedals aft (toward the pilot); clockwise turning of the knob adjusts the pedals forward.

D. Rudder Control Pilot Force Bungee

The rudder control pilot force bungee is in the lower part of the fuselage rear section at canted station 457. The bungee is a single-action type. The bungee has a movable piston rod and a preloaded spring. It

DESCRIPTION AND OPERATION (Continued)

Rudder Control Pilot Force Bungee (Cont'd)

is spring-loaded to its retracted position. The bungee is connected to the rudder control system to center the rudder pedals. When the rudder pedals are moved, the bungee spring is compressed, creating a force against the rudder pedals. When the rudder pedals are released, the bungee spring returns the bungee to its retracted position, centering the rudder pedals.

Rudder Trim Actuator

A linear-type, 28 VDC, electromechanical actuator operates the rudder trim tab. The rudder trim actuator is mechanically linked to the rudder trim tab by a bellcrank and push rod. The actuator is mounted near the leading edge of the rudder. The actuator has two internal travel limit switches to limit the travel of the jackscrew. Non-jamming mechanical stops are employed to restrict jackscrew travel in case of travel limit switch failure. An antirotational device is used to prevent the jackscrew end fitting from rotating. A rudder trim position transmitter (part of the actuator assembly) is internally geared to the jackscrew. Movement of the jackscrew operates the transmitter. The transmitter electrically operates the rudder trim position indicator in the cockpit. The transmitter is a potentiometer type designed to operate through 360 degrees or more of shaft rotation.

The transmitter wiper is centered on top B within 1-1/2 degrees with the actuator in neutral position. Retracting the actuator produces A, B, C direction of rotation of the wiper. Movement of the wiper electrically reproduces a similar movement in the indicator.

Rudder Trim Position Indicator

The rudder trim position indicator, on the co-pilot's inboard instrument panel, is powered by the 28 VDC essential bus. A rotating pointer indicates rudder left and rudder right trim positions of the trim tab in degrees. The indicator is electrically positioned by a trim position transmitter on the rudder trim actuator. The indicator is calibrated 15 degrees right or left in increments of 5 degrees.

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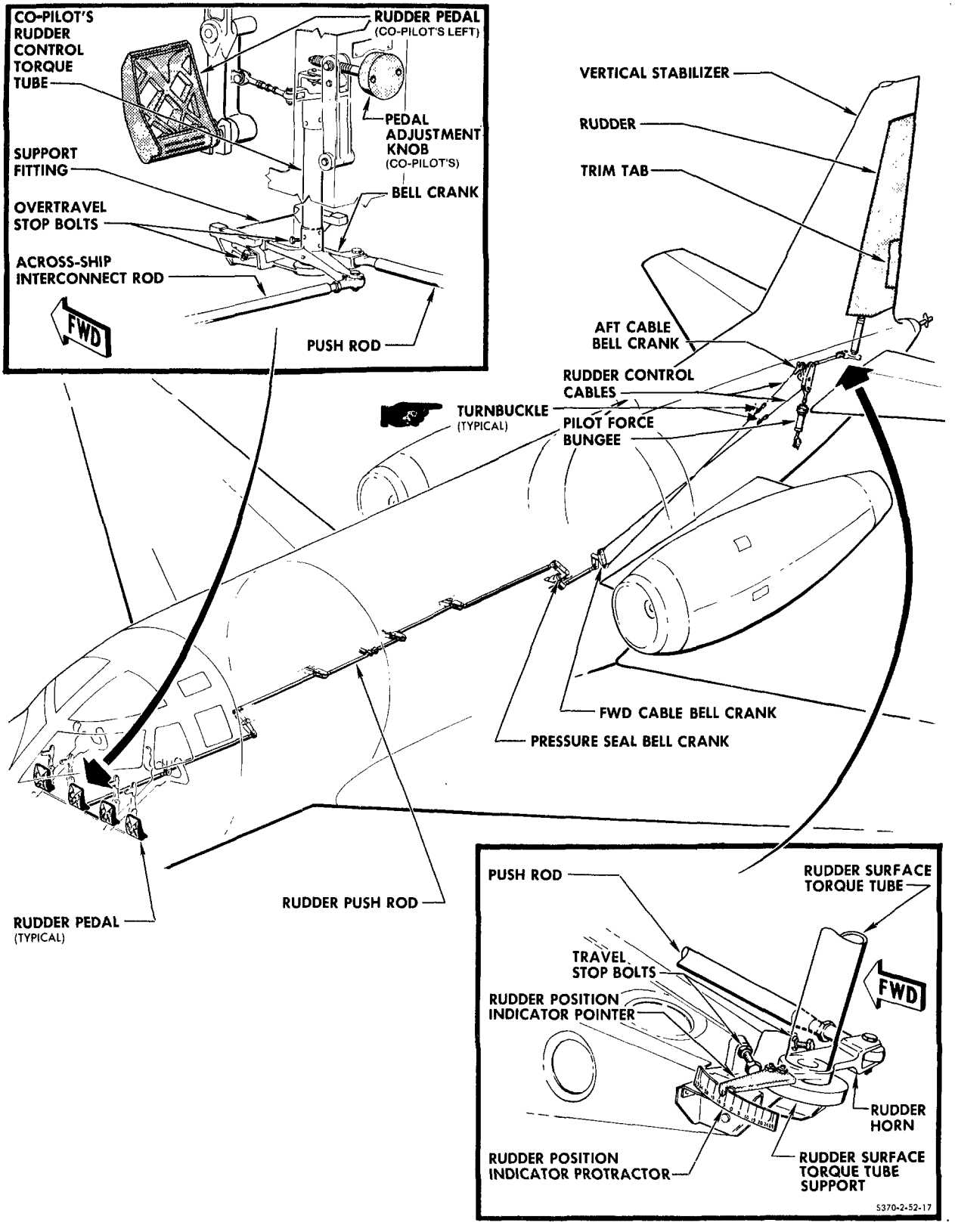

Figure 27-20-1 — Rudder Control System

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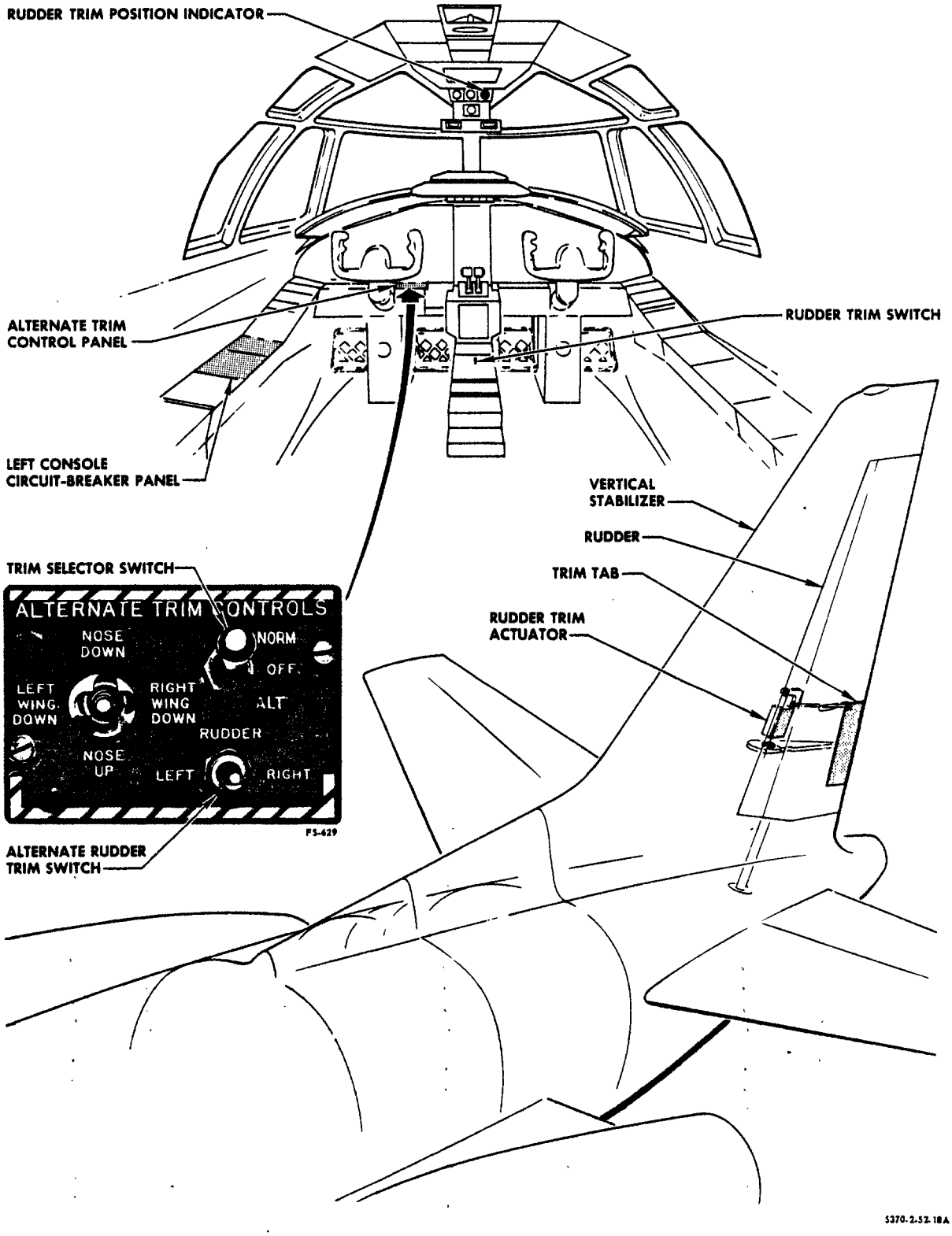

Rudder Trim System
Figure 27-20-2

RUDDER AND TAB - TROUBLE SHOOTING

1. Trouble Shooting Rudder Control and Rudder Trim Systems

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
	RUDDER CONTROL SYSTEM	
A. CONTROLS DO NOT MOVE.		
(1) Gust lock mechanism in engaged position.	Check elevator controls and aileron controls. If they cannot be moved, gust lock handle is engaged.	Disengage gust lock handle by moving handle down and forward from engaged position.
(2) Mechanical obstruction in control system linkage.	Check control system linkage to determine cause of mechanical interference. Check especially at bell cranks for rig pins.	
B. CONTROLS MOVE, BUT SURFACE DOES NOT FOLLOW.		
(1) Push rods, cables, or bell cranks disconnected.	Check that control system linkage is not disconnected at any point.	Reconnect linkage and secure jam nuts.
(2) Cable disconnected or broken.	Check cables in fuselage rear section.	Reconnect or replace cable.
C. FULL SURFACE TRAVEL CANNOT BE OBTAINED.		
(1) Improper adjustment of rudder stop bolts or overtravel stop bolts.	Refer to adjustment procedure in figure 27-20-403.	
(2) Mechanical interference or binding in control system.	Thoroughly inspect control system to determine cause of interference or binding.	Make certain that control system has adequate clearance from all structure, plumbing, wire bundles, etc.
(3) Improperly rigged rudder control system.	Remove access, and check position indicators. If travel is short in one direction	Rerig rudder control system. (See figure 27-20-403.)

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
	and if there is over-travel in opposite direction, linkage is probably misrigged.	
D. CONTROLS BIND MOMENTARILY DURING OPERATION.		
(1) Mechanical interference in control system.	While moving controls slowly back and forth over point where binding occurs, check entire system to determine cause of interference.	Make certain that control system has adequate clearance from all structure, plumbing, wire bundles, etc.
(2) Pilot force bungee faulty.	Remove bungee from airplane. Check bungee.	Replace bungee is faulty.
E. FRICTION FORCES TOO HIGH.		
(1) Cable tension too high.	Check cable tension.	Adjust cable tension to 300 (±25) pounds at 70°F.
(2) Mechanical interference in system.	While slowly moving controls back and forth, inspect entire system to determine whether it is properly installed and whether it has adequate clearance.	Make certain that control system has adequate clearance from all structure, plumbing, wire bundles, etc.
F. RUDDER PEDAL FORCES TOO HIGH OR TOO LOW.		
(1) Improperly rigged bungee.	Check bungee installation.	Adjust bungee. Bungee may be adjusted without being disconnected.
(2) Faulty bungee.	Remove bungee from airplane. Check bungee.	Replace bungee is faulty.
G. RUDDER PEDALS WILL NOT RETURN TO NEUTRAL WITHIN ±5 DEGREES.		
(1) Mechanical interference in control system.	While slowly moving controls back and forth, inspect entire system to determine whether it is properly installed and whether it has adequate clearance.	Make certain that control system has adequate clearance from all structure, plumbing, wire bundles, and other objects.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
(2) Improperly rigged bungee.	Check bungee installation.	Adjust bungee. Bungee may be adjusted without being disconnected.
(3) Faulty bungee.	Remove bungee from airplane. Check bungee.	Replace bungee is faulty.

NOTE: The following trouble shooting information is based on the assumption that electrical power is available to the system. It is suggested that frequent reference be made to the trim system schematic when trouble shooting the systems. This will aid in isolating a specific trouble.

RUDDER TRIM SYSTEM

H. TRIM SYSTEM INOPERATIVE.

(1) Circuit breakers not pushed in.	Check "NORMAL TRIM" and "ALTERNATE TRIM" circuit breakers on left console circuit-breaker panel.	
(2) Trim selector switch positioned incorrectly.	Check trim selector switch position on alternate trim control panel.	
(3) Electrical system faulty.	Perform continuity check.	Repair electrical system.

I. TRIM THROWS UNSATISFACTORY.

(1) Trim system improperly adjusted.	Check rigging of trim system.	
(2) Mechanical interference in linkage.	Remove trim actuator access door. Operate trim actuator back and forth while inspecting linkage, especially at bell crank, for mechanical interference.	
(3) Faulty trim actuator.	Check total trim tab travel from neutral position. If travel is less than 1-3/32 inches or more than 1-5/32 inches, the trim actuator is faulty.	Replace faulty actuator.

J. TRIM POSITION INDICATOR INOPERATIVE.

(1) Electrical system faulty.	Perform continuity check.	Repair electrical system.
(2) Faulty transmitter on trim actuator.	Check transmitter. (Refer to procedure for checking rudder trim system.) If transmitter does not meet check-out requirements, a faulty actuator is indicated.	
(3) Faulty trim position indicator.	If there is continuity in electrical system and transmitter meets check-out requirements, a faulty trim position indicator is indicated.	Replace trim position indicator.

RUDDER AND TAB - MAINTENANCE PRACTICES

1. Servicing Rudder Control and Rudder Trim Systems

Servicing of the rudder control and rudder trim systems is not required. All joints in both systems have lubricant-sealed bearings and do not require additional lubrication.

1. Removing Rudder

A. The rudder may be removed without disturbing the rudder system rigging. To remove rudder, proceed as follows:

(1) Remove access panel.

(2) Disconnect rudder trim tab actuator wiring harness from terminal strip TB532. The wiring harness extends from the bottom of the rudder torque tube.

(3) Disconnect rudder from rudder torque tube. This is done by rotating rudder to one side and removing two bolts from mounting brackets under rudder.

(4) Disconnect bonding wire under rudder near torque tube.

(5) Disconnect bonding wire and remove the four bolts securing the upper and center hinge supports from the vertical stabilizer.

(6) Disengage the hinge support brackets from the vertical stabilizer, move the rudder to one side to clear the trailing section of the vertical stabilizer. Lift rudder and pull rudder trim tab actuator wiring harness from rudder torque tube.

(7) If a new rudder is to be installed, remove the hinge support from the old rudder as follows:

(a) Disconnect bonding wires from the upper and center hinge supports. Remove rudder hinge pins and the hinge supports.

2. Installing Rudder

A. If the same unaltered rudder is to be installed, see figure 27-20-401. If a new rudder is to be installed, proceed as follows:

(1) Temporarily install rudder and mark correct gap as shown in figure 27-20-401.

NOTE: Spare rudder assemblies are furnished with a small amount of material on the skin ends for finish trimming. The excessive skin material will not interfere with installation.

(2) Remove rudder. Trim and file to mark.

NOTE: The rudder has a honeycomb trailing edge and the ends are filled with Corefil 615. The trailing edges are pretrimmed to the correct dimension at the time of manufacture and should not require additional trimming. If additional trimming is required, the Corefil 615 may be filed. To prevent delamination, file with the skin and not across it.

(3) If a new, repaired or a newly painted rudder is being installed, perform a static balance check as per the Structural Repair Manual NA-66-1032 or Sabreliner Service Bulletin No. 17.

(4) Do procedures in figure 27-20-401.

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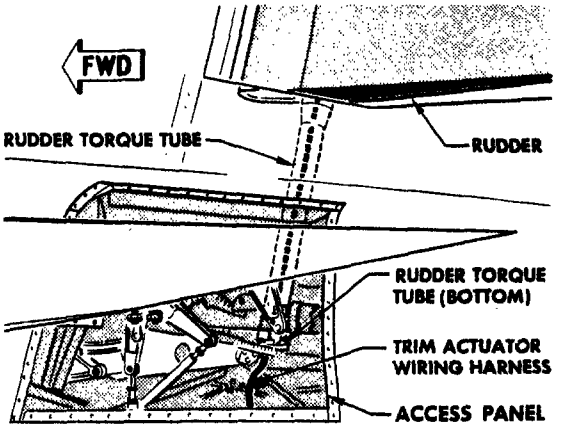
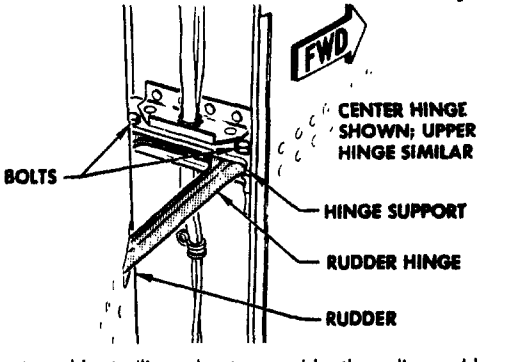
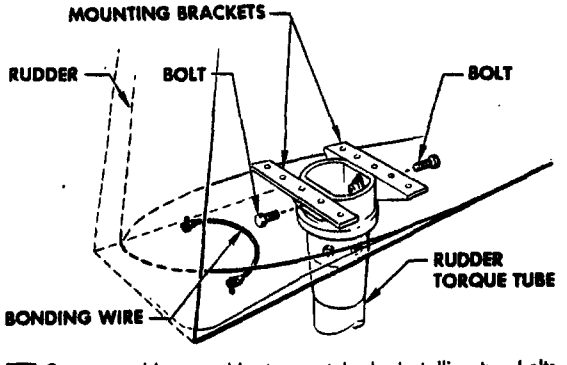


1 Insert rudder trim tab actuator wiring harness at bottom of rudder into rudder torque tube on fuselage.

2 While pulling rudder trim actuator wiring harness through bottom of rudder torque tube in access panel, carefully position rudder on rudder torque tube.

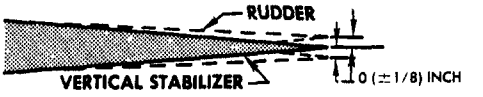


3 Rotate rudder trailing edge to one side; then align rudder hinges with hinge supports. Install bolts at center and upper hinges.



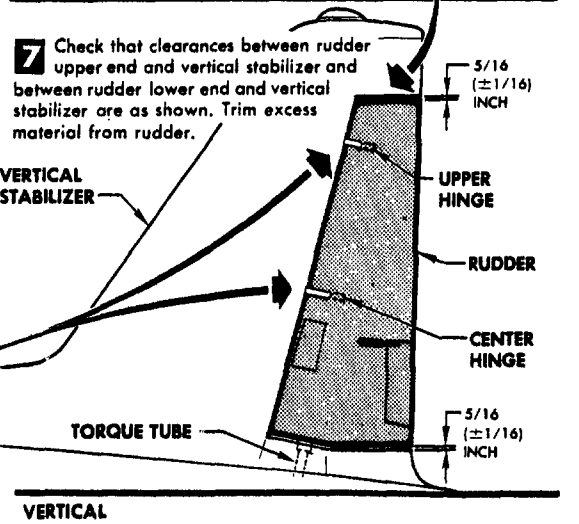
4 Connect rudder to rudder torque tube by installing two bolts into mounting brackets under rudder. Safety bolts to brackets with lockwire.

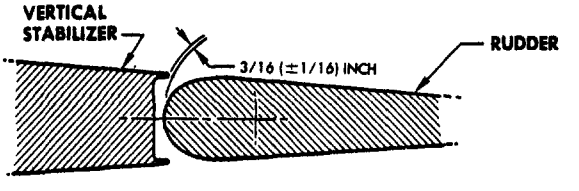
5 Connect bonding wire under rudder near torque tube.



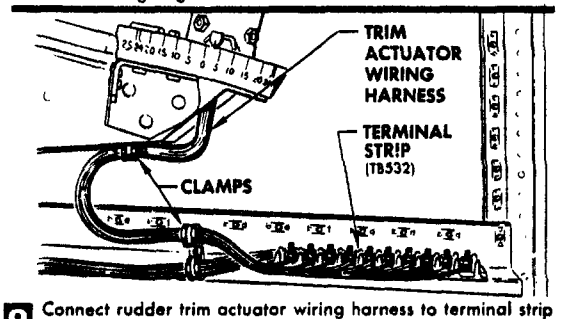
6 Align rudder lower trailing edge with vertical stabilizer trailing edge; then check that rudder upper trailing edge is aligned with vertical stabilizer trailing edge within ±1/8 inch. If rudder upper trailing edge is out of alignment with vertical stabilizer trailing edge more than ±1/8 inch, replace rudder.

7 Check that clearances between rudder upper end and vertical stabilizer and between rudder lower end and vertical stabilizer are as shown. Trim excess material from rudder.





8 Move rudder through its full range of travel. Check that clearance between rudder leading edge and vertical stabilizer trailing edges is as shown. Trim excess material from vertical stabilizer trailing edges.



9 Connect rudder trim actuator wiring harness to terminal strip TB532. Install clamps.

10 Do procedures in "Checking Rudder Control System."

11 Be sure that areas are free of rags, tools, and loose hardware; then install access panel.

S370-2-52-19

Installing Rudder
Figure 27-20-401

3. Removing Rudder Control Pilot Force Bungee

A. The pilot force bungee is in the lower fuselage rear section at canted station 457. To remove bungee, proceed as follows:

(1) Loosen bungee rod end, and rotate rod to allow rod stop to ride on housing of bungee to relieve preload.

(2) Disconnect bungee rod end from aft cable bell crank.

(3) Disconnect bungee end from bracket on bulkhead.

4. Installing Rudder Control Pilot Force Bungee

A. To install pilot force bungee, proceed as follows:

(1) Connect fixed end of bungee to bracket on bulkhead.

(2) Extend opposite end of bungee as necessary to connect to aft cable bell crank.

(3) To obtain correct setting, loosen rod-end jam nut and back off lock washer; rotate rod inside rod end until gap between rod stop and bungee end cap is 1/4 ($\pm$1/32) inch. Engage lock washer to rod end, tighten jam nut, and safety with lockwire.

5. Removing Rudder Trim Tab

A. To remove rudder trim tab, proceed as follows:

(1) Disconnect trim actuator rod end from upper hinge fitting.

(2) Disconnect bonding wire from upper hinge fitting.

(3) Rotate trim tab to one side, remove the three screws from lower hinge fitting, and remove trim tab.

(4) Attach hinge fitting to rudder to prevent loss of fitting.

6. Installing Rudder Trim Tab

For rudder trim tab installation, see figure 27-20-402.

NOTE: If a new, repaired or a newly painted trim tab is installed, perform a static balance check as per the Structural Repair Manual NA-66-1032 or Sabreliner Service Bulletin No. 17.

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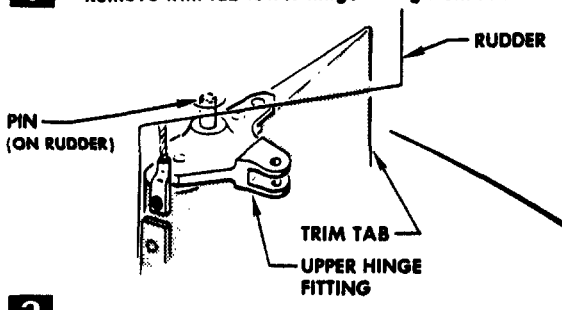
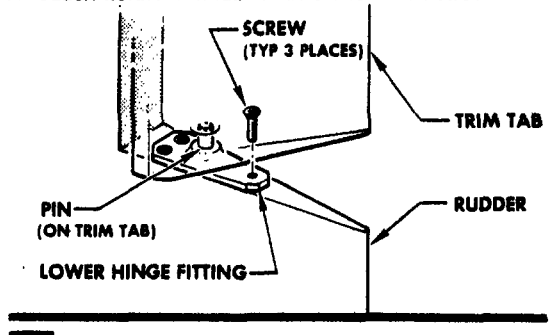
1 Remove trim tab lower hinge fitting from rudder.

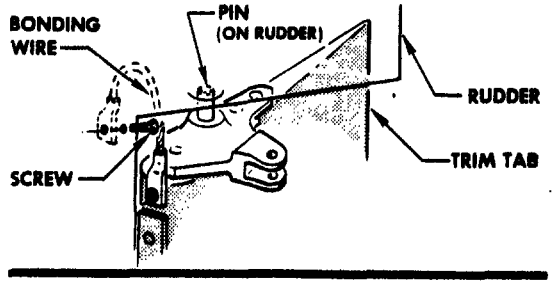


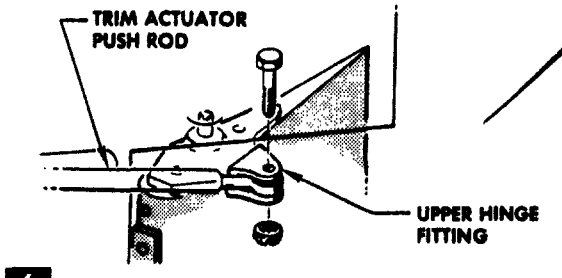
2 Position trim tab, in tab opening in rudder, so that trim tab upper hinge fitting is engaged with hinge pin on rudder. Hold trim tab lower end slightly out of alignment with rudder. Engage lower hinge fitting with trim tab; then slide fitting and tab into alignment with rudder.

3 Install the three screws securing lower hinge fitting to rudder. Rotate trim tab for access to install screws.



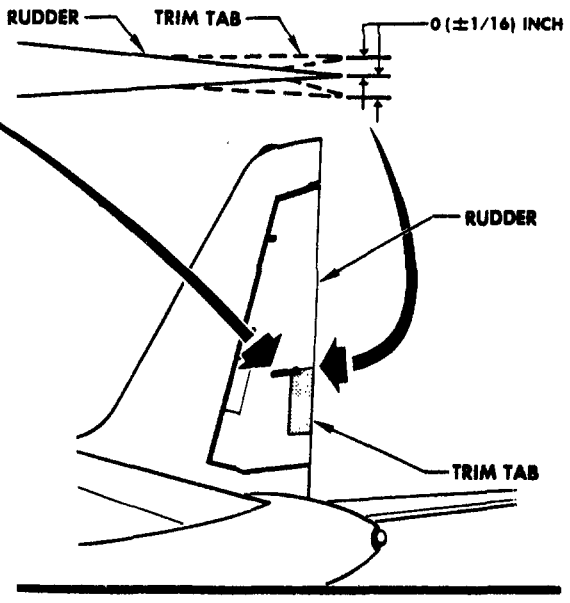
5 Connect bonding wire to rudder as shown.

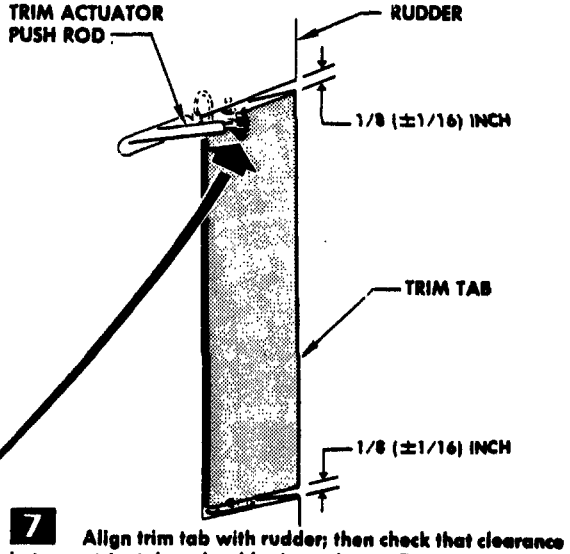




6 Connect trim actuator push rod end to clevis of trim tab upper hinge fitting.

4 Align trim tab lower trailing edge with rudder trailing edge; then check that trim tab upper trailing edge is aligned with rudder trailing edge within ±1/16 inch. If trim tab upper trailing edge is out of alignment with rudder trailing edge more than 1/16 inch, replace trim tab.





7 Align trim tab with rudder; then check that clearance between trim tab and rudder is as shown. Trim as necessary.

8 Check rudder trim. (Refer to "Checking Rudder Trim System.")

9 Operate trim tab through its full range of travel. There should be no evidence of binding.

5370-2-52-20

Installing Rudder Trim Tab
Figure 27-20-402

NOTE

Use 5/16-inch diameter rods or bolts as rig pins. Maintain a "drop fit" on all rig pins, to eliminate forced aligment of rods or bellcranks.

CAUTION

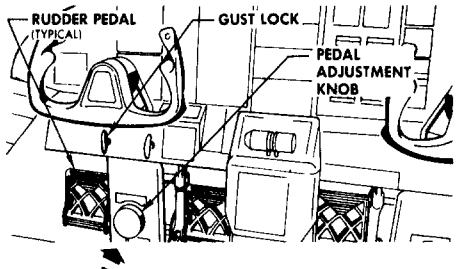
Do not apply loads to the control system when rig pins are installed. Damage to system parts can occur.

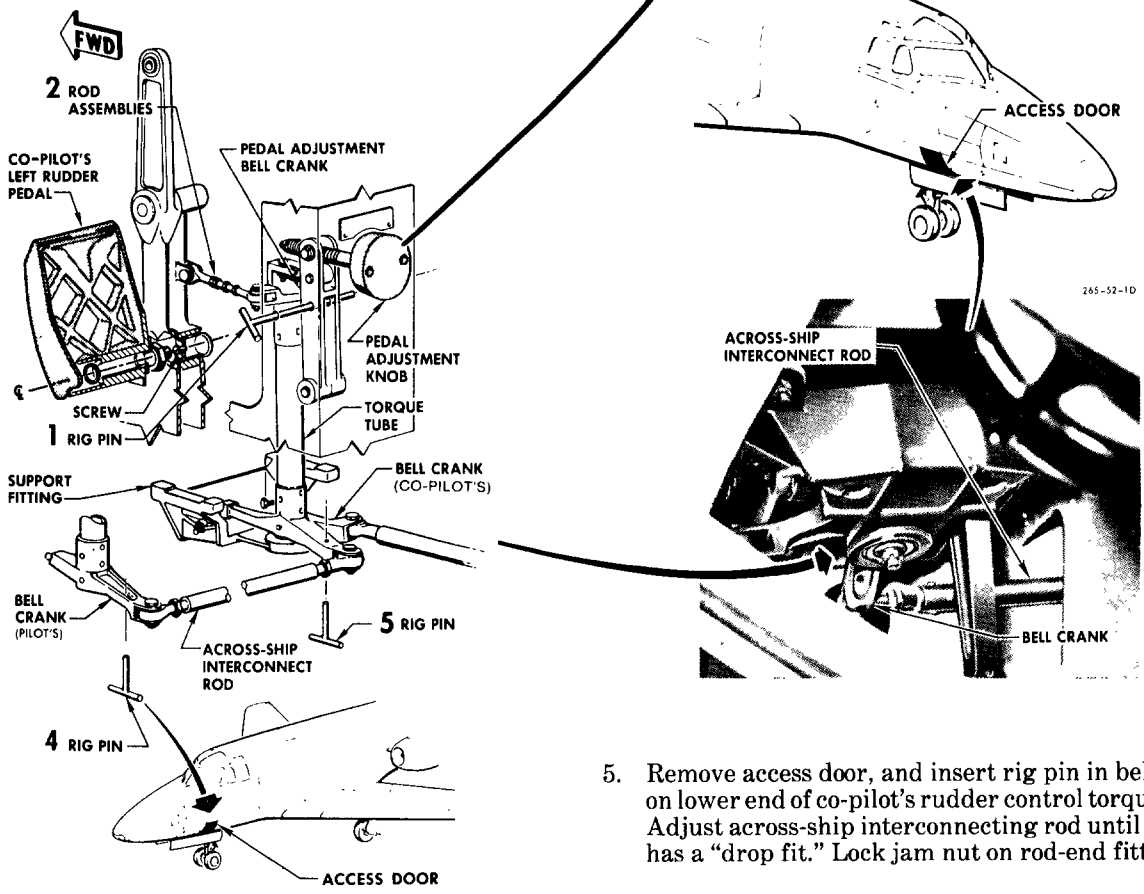
1. Insert rig pin into pedal adjustment bellcrank. Rotate pedal adjustment knob on co-pilot's control pedestal until rig has a "drop fit."
2. Adjust rod assemblies until centerline of rudder pedals aligns with centerline of screws in support. Lock jam nut on rod-end fittings.

NOTE

Clockwise rotation of pedal adjustment knob moves pedals forward; counterclockwise rotation moves pedals aft.

3. Repeat steps 1 and 2 on pilot's rudder controls.
4. Remove access door and, ensuring that gust lock handle is in the disengaged position, insert rig pin in bellcrank on lower end of pilot's torque tube.





5. Remove access door, and insert rig pin in bellcrank on lower end of co-pilot's rudder control torque tube. Adjust across-ship interconnecting rod until rig pin has a "drop fit." Lock jam nut on rod-end fitting.

Figure 27-20-403 — Rigging Rudder Control System (Sheet 1 of 3)

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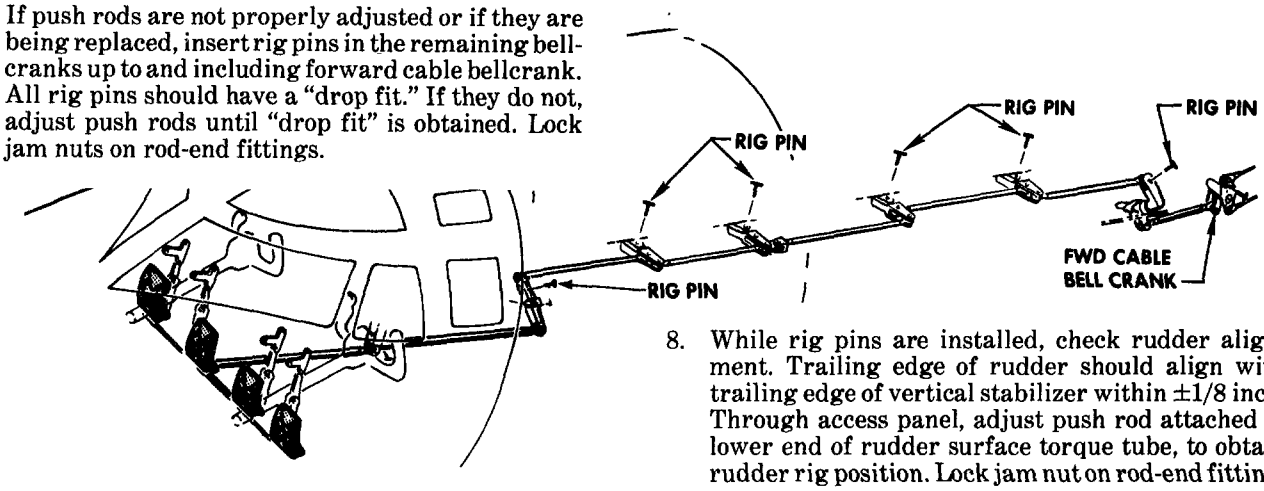
6. If push rods are not properly adjusted or if they are being replaced, insert rig pins in the remaining bellcranks up to and including forward cable bellcrank. All rig pins should have a "drop fit." If they do not, adjust push rods until "drop fit" is obtained. Lock jam nuts on rod-end fittings.



7. Through aft fuselage access doors, insert rig pin in aft cable bellcrank. Rig pin should have a "drop fit," and cable tension should be 300 (± 30) pounds at 70°F (see Figure 201). If this "drop fit" and cable tension are not obtained, alternately adjust turnbuckles until cable tension is as stated, and until rig pins have a "drop fit." Turnbuckle adjusting nut should be no less than 1/8 inch from edge of turnbuckle. Secure turnbuckles with safetywire.

NOTE

Check that rudder control cables cross, from top of cable bellcranks to bottom of opposite bellcranks.

8. While rig pins are installed, check rudder alignment. Trailing edge of rudder should align with trailing edge of vertical stabilizer within ±1/8 inch. Through access panel, adjust push rod attached to lower end of rudder surface torque tube, to obtain rudder rig position. Lock jam nut on rod-end fitting.

9. Through access panel, check that rudder position indicator pointer at lower end of rudder surface torque tube is set at zero degrees. If it is not, adjust rudder position indicator protractor until pointer is at zero degrees. Tighten screws and secure with safetywire.

10. Check that clearance between bungee end cap and rod stop is 1/4 (±1/32) inch. If it is not, loosen rod-end jam nuts, and back off lock washer; then rotate rod inside rod end until this clearance is met. Engage lock washer with rod end, tighten jam nut, and secure nut to lock washer with safetywire.

CAUTION

Do not break safetywire on bungee housing or on rod stop.

11. Remove all rig pins. Bungee should align rudder within ±2 degrees of rig position. Repeat steps 9 through 11 as necessary to align rudder within tolerance.

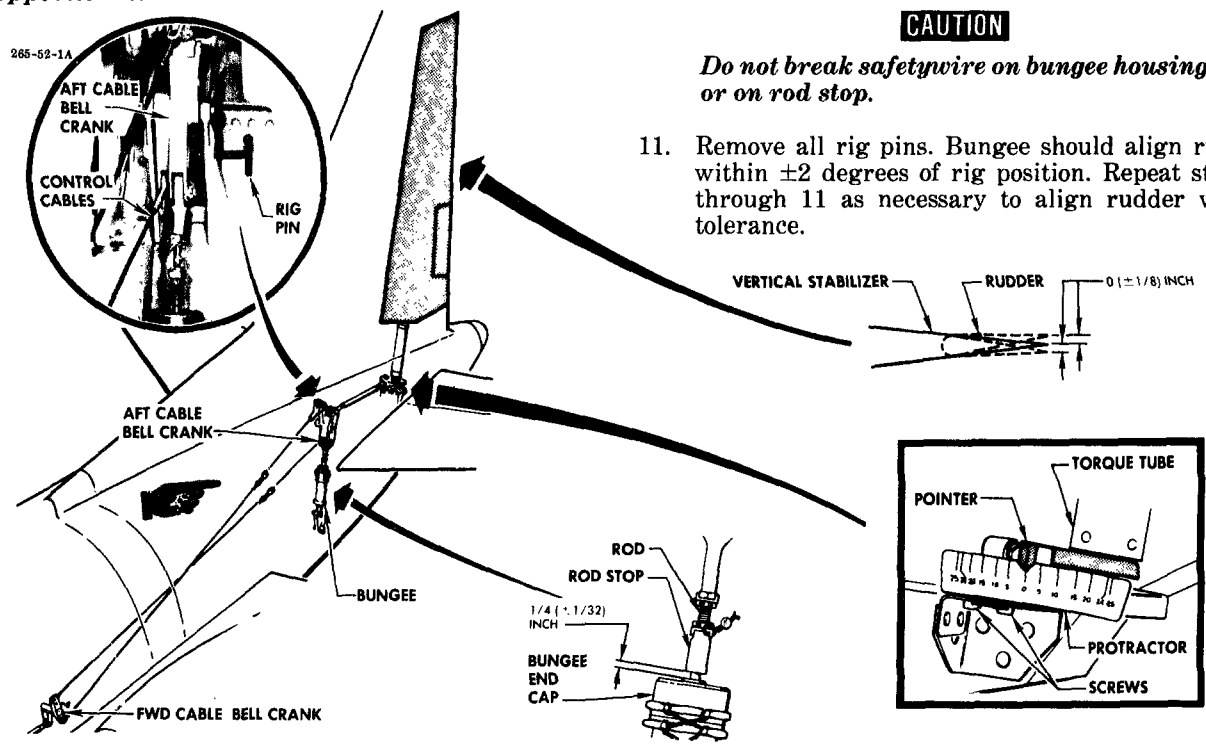


Figure 27-20-403 — Rigging Rudder Control System (Sheet 2 of 3)

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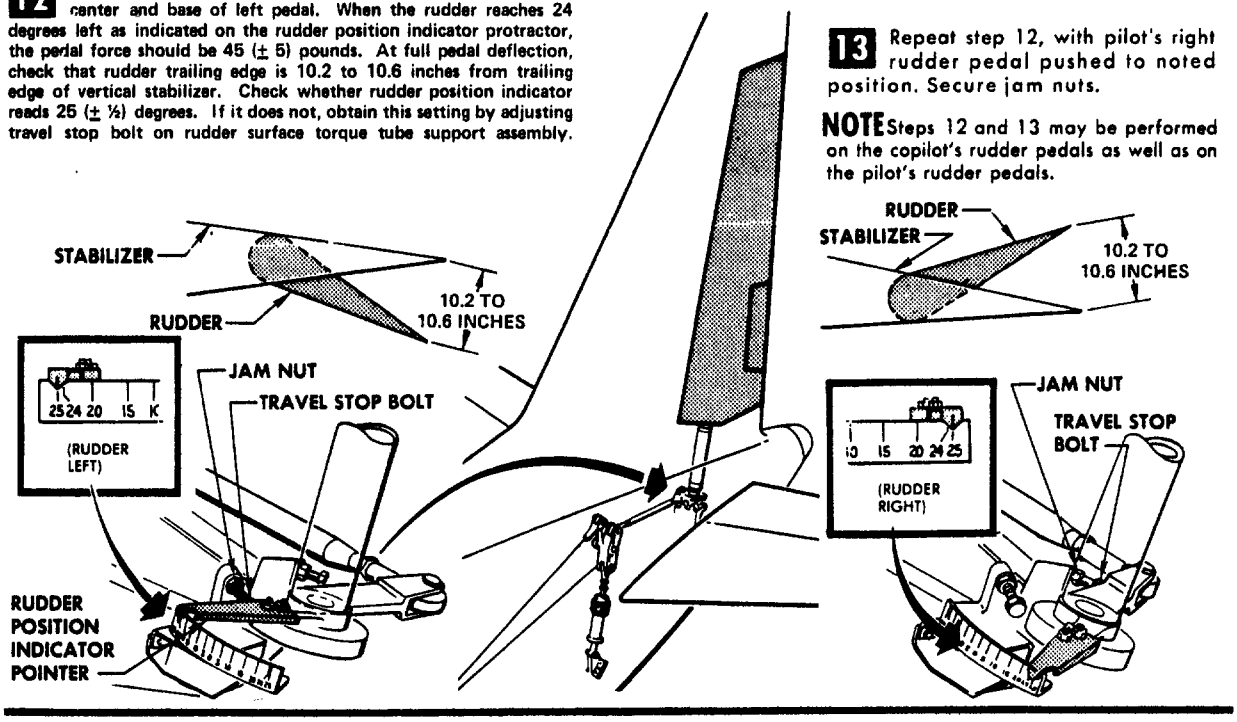
12 Using a precision scale, apply force in a forward direction, to center and base of left pedal. When the rudder reaches 24 degrees left as indicated on the rudder position indicator protractor, the pedal force should be 45 (± 5) pounds. At full pedal deflection, check that rudder trailing edge is 10.2 to 10.6 inches from trailing edge of vertical stabilizer. Check whether rudder position indicator reads 25 (± ½) degrees. If it does not, obtain this setting by adjusting travel stop bolt on rudder surface torque tube support assembly.

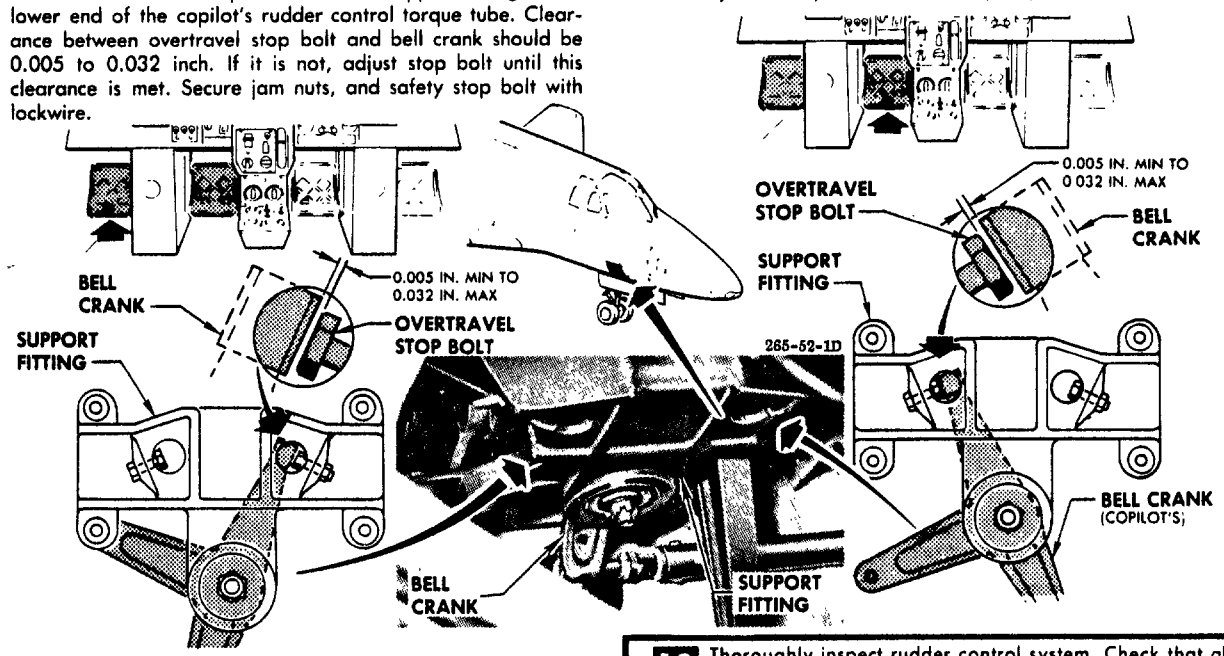
13 Repeat step 12, with pilot's right rudder pedal pushed to noted position. Secure jam nuts.

NOTE Steps 12 and 13 may be performed on the copilot's rudder pedals as well as on the pilot's rudder pedals.



14 Push pilot's left rudder pedal to full-forward position. Using a precision scale, apply 180-pound force, in a forward direction, to center and base of left rudder pedal. Maintain this force while checking clearance of overtravel stop bolt. The overtravel stop bolt is on the support fitting at the lower end of the copilot's rudder control torque tube. Clearance between overtravel stop bolt and bell crank should be 0.005 to 0.032 inch. If it is not, adjust stop bolt until this clearance is met. Secure jam nuts, and safety stop bolt with lockwire.

15 Repeat procedure, applying 180-pound force in a forward direction to the pilot's right rudder pedal. Clearance between remaining overtravel stop bolt and bell crank should be 0.005 to 0.032 inch. Adjust stop bolt as necessary. Secure jam nuts and safety stop bolt with lockwire.



16 Repeat steps 14 and 15 on copilot's rudder pedals.

17 Do procedures in "Checking Rudder Control System."

18 Thoroughly inspect rudder control system. Check that all moving parts, through full range of travel, do not bind or touch structure. Check security of all jam nuts. Check proper safetying of all bolts and turnbuckles. Be sure that areas are free of tools, rags, and loose hardware. Install access doors.

S370-2-52-23A

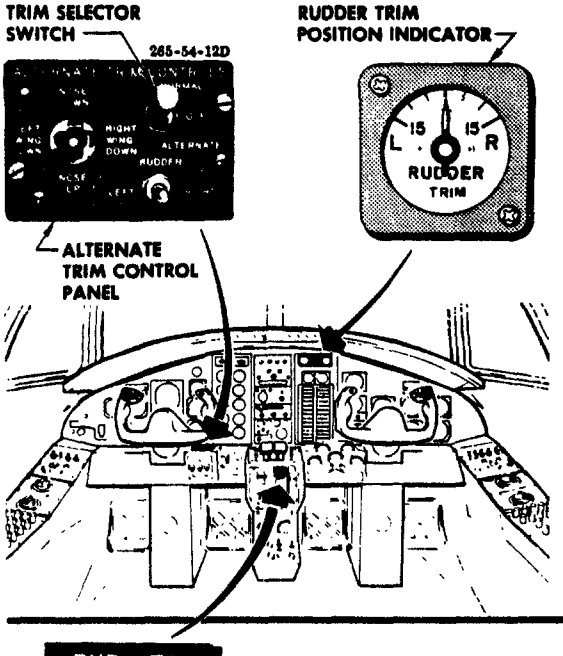
ORIGINAL
As Received By
ATP

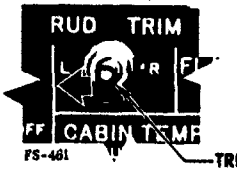
NOTE The object of this procedure is to synchronize, within tolerances and through the full range of travel, the travel of the rudder trim tab trailing edge with the travel of the rudder trim position indicator needle. The total travel of the rudder trim tab trailing edge should be equally divided between the full left and full right positions from the rudder trailing edge.

1 Connect 28-volt dc utility external power source to airplane.

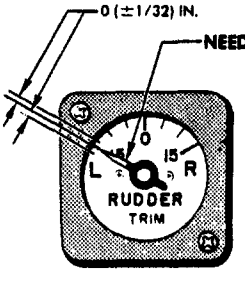
2 Move trim selector switch on alternate trim control panel to NORMAL.

3 Operate trim switch, on front of center pedestal, until rudder trim position indicator reads zero degrees.



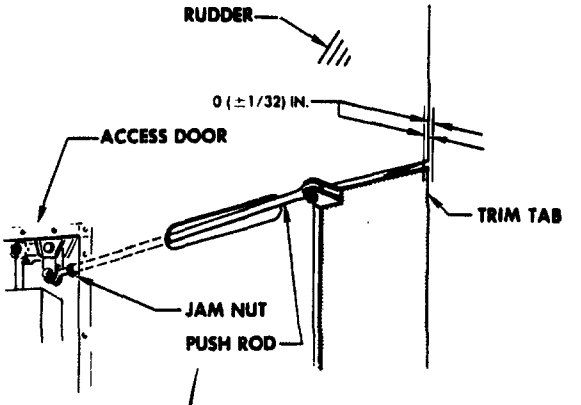


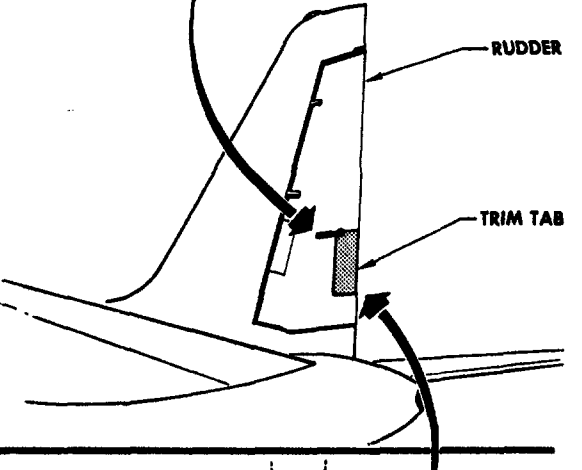
5 Move trim switch to L (left). Hold switch at L until rudder trim tab stops moving.

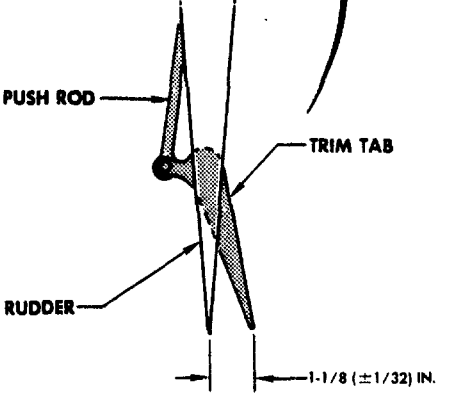


6 Check full travel position of rudder trim position indicator needle. The needle should align, within ±1/32 inch, with the 15-degree rudder left index line. If indicator needle does not align within this tolerance, refer to "Trouble Shooting"

4 Check that trim tab trailing edge is aligned, within ±1/32 inch, with rudder trailing edge. If it is not, adjust push rod until trailing edges align within this tolerance.







7 Check that trim tab trailing edge is 1-1/8 (±1/32) inches right of rudder trailing edge. If it is not, adjust push rod until it is.

S370-2-52-24

Rigging Rudder Trim System
Figure 27-20-404 (Sheet 1 of 2)

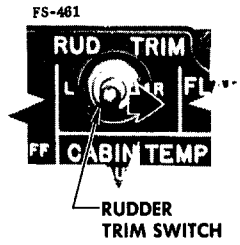
8. Move trim switch to R (right). Hold trim switch at R until rudder trim tab stops moving.

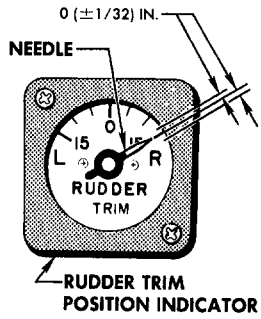
9. Check that rudder trim position indicator needle aligns, within ±1/32 inch, with the 15-degree rudder right index line. If indicator needle does not align within this tolerance, refer to Trouble Shooting.

10. Check that trailing edge of trim tab is 1-1/8 (±1/32) inches left of rudder trailing edge. Readjust push rod as necessary to equally divide total travel of trim tab.

11. Repeat steps 3 and 4. Secure and safety jam nut on push rod with safetywire.

12. Do procedures in Checking Rudder Trim System.





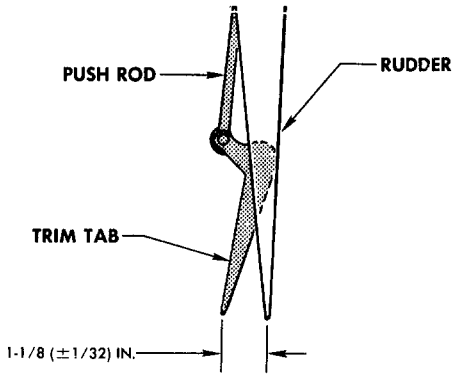


Figure 27-20-404 — Rigging Rudder Trim System (Sheet 2 of 2)

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REMOVAL AND INSTALLATION (Continued)

Removing Rudder Trim Actuator

1. Remove trim actuator access panel (refer to chapter 6).
2. Disconnect electrical plug.
3. Disconnect actuator shaft to bellcrank attach bolt.
4. Disconnect actuator attach bolt and remove actuator from rudder.

Installing Rudder Trim Actuator

1. Position actuator in rudder and install attaching bolts, washers, nuts and cotter pins.
2. Connect actuator electrical plug. Lockwire electrical plug.
3. Check rudder trim rigging. Refer to Rigging Rudder Trim System.
4. Install rudder trim actuator access panel.

Rigging Rudder Control System

For adjustment and rigging of rudder control system, see Figure 27-20-403.

Rigging Rudder Trim System

For adjustment and rigging of rudder trim system, see Figure 27-20-404.

RUDDER CONTROL CABLE REMOVAL

NOTE

Use 5/16 inch dia rods or bolts as rig pins. Maintain a "drop fit" on all rig pins to eliminate forced alignment of bellcranks.

CAUTION

Do not apply loads to the control system when rig pins are installed. Damage to system parts can occur.

1. Remove electrical power from airplane.
2. Insert rig pins in forward and aft rudder cable bellcranks, located in aft fuselage area (see Figure 27-20-403).
3. Cut safetywire from rudder control cable turnbuckles and disassemble turnbuckles.
4. Remove cables from bellcranks and remove from airplane.

RUDDER CONTROL CABLE INSTALLATION AND TENSION CHECK

1. Ensure that rig pins are installed in the forward and aft rudder bellcanks (see Figure 27-20-403).
2. Install cables on bellcranks and assemble turnbuckles.

NOTE

Check that rudder control cables cross from top of the cable bellcrank to the bottom of the opposite bellcrank. Also, ensure that turnbuckles be located closer to the aft bellcrank (see Figure 27-20-403).

3. Alternately adjust the turnbuckles to obtain a cable tension of 300 (±30) pounds at 70°F (see Figure 27-00-201). Turnbuckle adjusting nut should be no less than 1/8 inch from edge of turnbuckle.
4. Ensure that the rig pins still have a "drop fit."
5. Safety turnbuckles using .040 in. safetywire.
6. Remove all rig pins.
7. Perform operational test of rudder control system.
8. Install access panels previously removed to access bellcranks.
9. Secure airplane as necessary.

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1. Checking Rudder Control System

A. Before check-out, the rudder control system must be properly rigged. (See figure 27-20-403.) If check-out does not meet requirements, refer to trouble shooting procedures. The parts referred to in the following procedure are identified in figures 27-20-1 and 27-20-403.

(1) Check that gust lock is disengaged; then move rudder pedals through full travel and check whether they operate freely. There should be no sign of binding.

(2) Push left rudder pedal to full-forward position. The rudder trailing edge should move left. When the pedal is released, it should freely return to neutral.

(3) Repeat procedure with right rudder pedal pushed to full-forward position. The rudder should move right. When the pedal is released, it should freely return to neutral.

(4) Check rudder travel positions on rudder position indicator at lower end of rudder surface torque tube as follows:

(a) To gain access to rudder position indicator, remove access panel 8.

(b) Align rudder trailing edge with vertical stabilizer lower trailing edge. The rudder position indicator should read 0 (±1/2) degrees.

(c) Push pilot's left rudder pedal to full-forward position. The position indicator should read 25 (±1/2) in full-travel position. When the pedal is released, it should freely return to neutral position within ±5 degrees.

(d) Repeat procedure with right rudder pedal pushed to full-forward position. The rudder position indicator should read 25 (±1/2) degrees in full-travel position. When the pedal is released, it should freely return to neutral within ±5 degrees.

(5) Check rudder pedal forces by pushing base of left rudder pedal. Deflect rudder until rudder position indicator at lower end of rudder torque tube is positioned at 24 degrees. The scale should read 45 (±5) pounds. Repeat procedure on right rudder pedal. The results should be the same.

(6) Install access panel 8.

2. Checking Rudder Trim System

A. Before check-out, the rudder trim system must be properly rigged. (See figure 27-20-404.) If trim system does not meet check-out requirements, refer to trouble shooting procedures. The parts referred to in the following procedure are identified in figures 27-20-2 and 27-20-404.

(1) Connect 28-volt dc utility external power source to airplane.

(2) Push in the following circuit breakers on left console circuit-breaker panel:

- "ALTERNATE TRIM - SYSTEM"
- "NORMAL TRIM - SYSTEM"
- "NORM TRIM - CONT"
- "TRIM IND"

(3) Place trim selector switch, on alternate trim control panel, at NORMAL.

(4) Place rudder trim switch, on front of center pedestal under throttle control levers, at L (left). Hold until rudder trim tab stops moving. The rudder trim tab should move right. The needle of the rudder trim position indicator should align within ±1/32 inch of the 15-degree rudder-left index line. Check that trailing edge of trim tab is 1-1/8 (±1/32) inches right from trailing edge of rudder.

(5) Hold rudder trim switch at R (right) until rudder trim tab stops moving. The rudder trim tab should move left. The needle of the rudder trim position indicator should align within ± 1/32 inch of the 15-degree rudder-right index line. Check that trailing edge of trim tab is 1-1/8 (±1/32) inches left from trailing edge of rudder.

(6) Operate rudder trim switch until rudder trim position indicator reads zero degrees. Check that trailing edge of trim tab is aligned with trailing edge of rudder within ± 1/32 inch.

(7) Position trim selector switch, on alternate trim control panel, to ALTERNATE.

(8) Place rudder alternate trim switch, on alternate trim control panel, at LEFT. Hold until rudder trim tab stops moving. The rudder trim tab should move right. The rudder trim position indicator should read rudder left 15 degrees.

(9) Hold rudder alternate trim switch at RIGHT until rudder trim tab stops moving. The rudder trim tab should move left. The rudder trim position indicator should read rudder right 15 degrees.

(10) Check timing of rudder trim actuator. Use a stop watch to check time it takes for rudder trim actuator to operate from one extreme position to the other extreme trim position. Trim rate should be no less than 2.18 degrees per second (13.8 seconds total travel time) and no more than 3.5 degrees per second (about 8.6 seconds total travel time).

(11) Disconnect 28-volt dc external electrical power source from airplane.

INSPECTION/CHECK (Continued)

INSPECT — VERTICAL STABILIZER AND RUDDER

Inspect the vertical stabilizer and rudder as follows:

1. Vertical stabilizer skin for cracks, buckles, dents, abrasions, wrinkles, and loose rivets or screws. Fiberglass antenna fairing for condition. Antennas for condition and security.
2. Rudder and trim tab for security, tears, and missing rivets. Bonding straps for condition and security.
3. Rudder hinges for cracks, hinge bolts for looseness. Bonding straps for condition and security.
4. Rudder torque tube jo-bolts for looseness. (Do not check with wrench or attempt to tighten.)
5. Rudder torque tube bearing for cleanliness and freedom of rotation.

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ELEVATOR CONTROL SYSTEM - DESCRIPTION AND OPERATION

1. General (See figure 27-30-1.)

The elevator control system consists of two interconnected forward and aft moving control columns in the cockpit and a bobweight mechanically linked by push rods, bell cranks, and cables to aerodynamically balanced elevators. The elevators are hinged to the trailing edge of the horizontal stabilizer, to provide longitudinal (pitch) control of the airplane. A spring-type bias bungee is incorporated to establish a no-load center position for the elevator controls by interconnecting the elevators and horizontal stabilizer. This no-load position varies automatically with stabilizer trim position. The complete elevator control system is balanced by a bobweight balance bungee.

2. Elevators

The elevators are interchangeable and are interconnected by an aluminum alloy torque tube. The basic structure consists of stringers, conventional ribs, and a honeycomb trailing edge covered by stressed aluminum alloy skin. The elevators are hinged to the horizontal stabilizer trailing edge at three places on the left panel and three places on the right panel. Balance is obtained by means of weights mounted in the leading edge. Two elevator horns, one on each end of the torque tube, connect the elevators to the elevator actuating linkage.

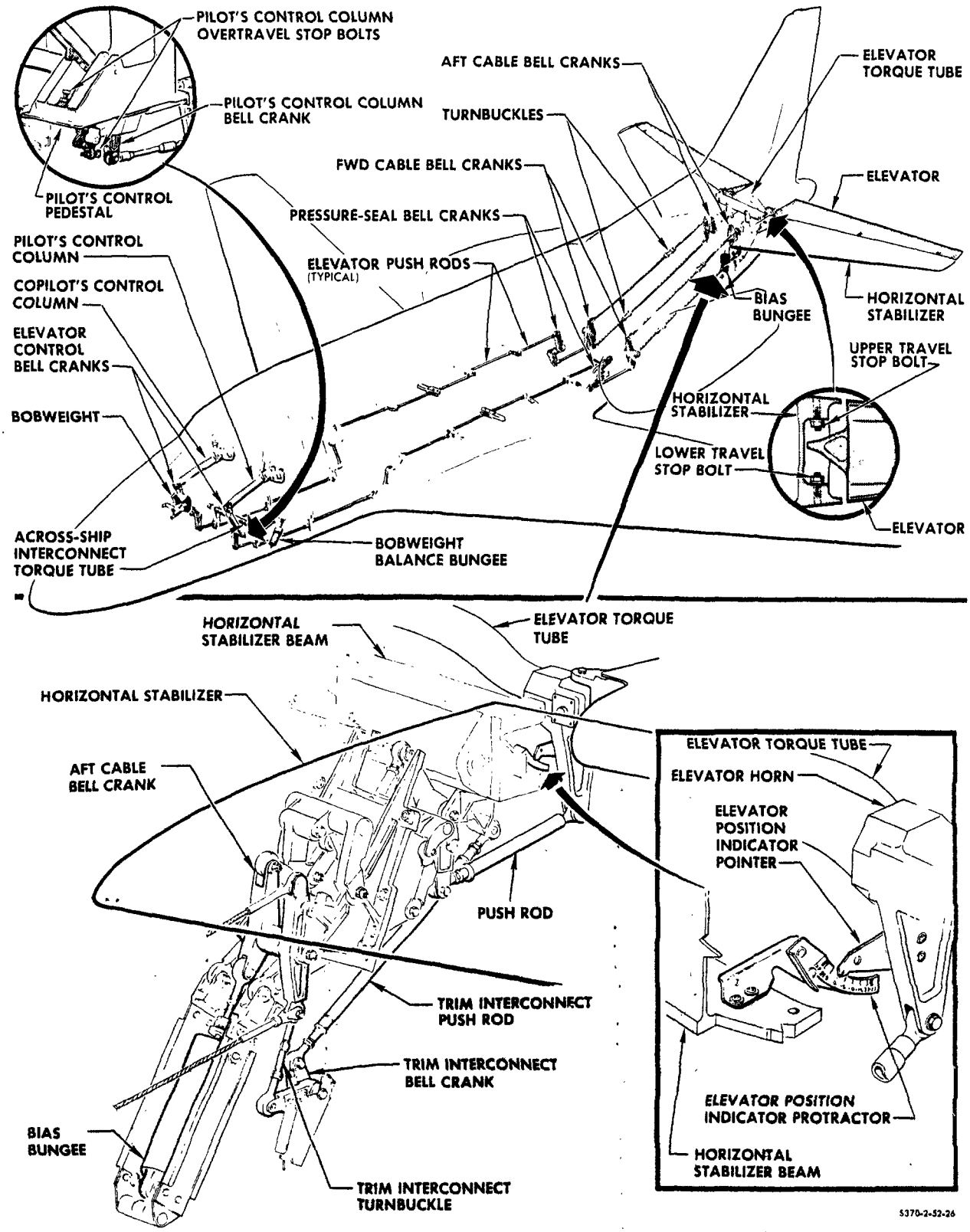
3. Elevator Control Bobweight Balance Bungee

The elevator control bobweight balance bungee is on the left side of the lower part of the fuselage forward section at station 108. The bungee is a single-action type and has a movable piston rod and a preloaded spring. It is spring-loaded toward its retracted position, and is connected to the elevator control columns. For this reason, when the system is in rigged neutral position, the bungee is partially extended to counteract the bobweight and control linkage weight. When the control columns are moved forward from neutral, the bungee is extended farther, and the spring is compressed accordingly. If the control columns are then released (hands off), the bungee repositions the controls to neutral. When the control columns are moved aft from neutral, the bungee is retracted, imposing less counteracting force. Releasing the controls at this point then allows the weight of the control linkage and bobweight to reposition the controls to neutral and reload the balance bungee.

4. Elevator Control Bias Bungee

The elevator control bias bungee is on the left side of the fuselage rear section at canted station 457. The bungee is a single-action type and has a movable piston rod and a preloaded spring. It is spring-loaded in its retracted position. When the control columns are moved, the bungee spring is compressed, creating a force against the control columns. When the control columns are released, the bungee spring returns the bungee to its retracted position. The lower end of the bungee is attached to a truss that is mechanically linked to the horizontal stabilizer. The rod end of the

bungee is attached to the aft cable bell crank. Movement of the stabilizer operates the mechanical linkage and repositions the lower end of the bungee. Motion of the bungee rotates the aft cable bell crank. The bell crank positions the elevator controls in a no-load trim position.



Elevator Control System
Figure 27-30-1

ELEVATOR CONTROL SYSTEM - TROUBLE SHOOTING

1. Trouble Shooting Elevator Control System

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. CONTROLS DO NOT MOVE.		
(1) Gust lock mechanism in engaged position.	Check aileron controls and rudder controls. If they cannot be moved, gust lock handle is engaged.	Disengage gust lock handle by moving handle down and forward from engaged position.
(2) Mechanical obstruction in control system linkage.	Check control system linkage to determine cause of mechanical interference. Check especially at bell cranks for rig pins.	
B. CONTROLS MOVE, BUT SURFACE DOES NOT FOLLOW.		
(1) Push rods, cables, or bell cranks disconnected.	Check control system linkage to see that it is not disconnected at any point.	Reconnect linkage and secure jam nuts.
(2) Cable disconnected or broken.	Check cables in fuselage rear section.	Reconnect or replace cable.
C. FULL SURFACE TRAVEL CANNOT BE OBTAINED.		
(1) Improper adjustment of elevator stop bolts or overtravel stop bolts.	Refer to adjustment procedure in figure 27-30-404.	
(2) Mechanical interference or binding in control system.	Thoroughly inspect control system to determine cause of interference or binding.	Make certain that control system has adequate clearance from all structure, plumbing, wire bundles, etc.
(3) Improperly rigged elevator control system.	Remove access and check position indicators. Position indicator should read 11 degrees trailing edge full down and 22 degrees trailing edge full up. If travel	Rerig elevator control system. (See figure 27-30-404.)

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
	is short in one direction and if there is overtravel in opposite direction, linkage is probably misrigged.	
D. CONTROLS BIND MOMENTARILY DURING OPERATION.		
(1) Mechanical interference in control system	While slowly moving controls back and forth over point where binding occurs, check entire system to determine cause of interference.	Make certain that control system has adequate clearance from all structure, plumbing, wire bundles, etc.
(2) Faulty bungee.	Remove bias bungee from airplane. Check bungee. (See figure 27-30-602.) If bias bungee is not faulty, check bobweight balance bungee. (See figure 27-30-601.)	Replace faulty bungee.
E. FRICTION FORCES TOO HIGH.		
(1) Cable tension too high.	Check cable tension.	Adjust cable tension to 275 (±25) pounds at 70°F.
(2) Mechanical interference in system.	While slowly moving controls back and forth, inspect entire system, including control columns, to determine whether it is properly installed, and whether it has adequate clearance.	Make certain that control system has adequate clearance from all structure, plumbing, wire bundles, and other objects.
F. ELEVATOR CONTROL FORCES TOO HIGH OR TOO LOW.		
(1) Improperly rigged bungees.	Check bungee installations.	Adjust bungee.
(2) Faulty bungee.	Remove bungee from airplane. Check bungee.	Replace faulty bungee.
G. ELEVATOR CONTROLS WILL NOT RETURN TO TRIM POSITION.		
(1) Mechanical interference in control system.	While slowly moving controls back and forth, inspect entire system to determine	Make certain that control system has adequate clearance

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
	whether it is properly installed and whether it has adequate clearance.	from all structure, plumbing, wire bundles, and other objects.
(2) Improperly rigged bungees.	Check bungee installations.	Adjust bungee.
(3) Faulty bungee.	Remove bungee from airplane. Check bungee.	Replace faulty bungee.

ELEVATOR CONTROL SYSTEM - MAINTENANCE PRACTICES

1. Removing Elevators

A. To remove elevators, proceed as follows:

NOTE: The removal procedures for the left and right elevators are the same.

(1) Remove two bolts at outboard hinge and two bolts at intermediate hinge by rotating elevator up to remove the bottom bolts, and down to remove the top bolts.

(2) When replacing elevators with new elevators, and when reinstalling all elevators, remove two screws on top and two screws on bottom of elevator from inboard hinge fitting. Remove two screws from aft face of inboard hinge fitting.

(3) Disconnect bonding wire at inboard and outboard hinge, and remove elevator.

2. Installing Elevators

A. If the same unaltered elevator is to be installed, do procedures in figure 27-30-401. If a new elevator is to be installed, do the following procedure.

(1) Temporarily install elevator and mark correct gap as shown in figure 27-30-401.

NOTE: Spare elevator assemblies are furnished with a small amount of material on the skin inboard end for finish trimming. The excessive skin material will not interfere with installation.

(2) Remove elevator. Trim and file to mark.

NOTE: The elevator has a honeycomb trailing edge and the end is filled with Corefil 615. The trailing edge is pretrimmed to the correct dimension at the time of manufacture and should not require additional trimming. If additional trimming is required, the Corefil 615 may be filed. To prevent delamination, file with the skin and not across it.

(3) If a new, repaired or a newly painted elevator is being installed, perform a static balance check as per the Structural Repair Manual NA-66-1032.

(4) Do procedures in figure 27-30-401.

3. Removing Elevator Hinge Pin

A. To remove elevator hinge pin, proceed as follows:

(1) Remove elevator tip from elevator.

(2) Remove lockwire from retainer bolt and remove bolt.

(3) Remove center hinge pin access door at bottom of left-hand elevator panel and top of right-hand elevator panel).

(4) Remove lockwire from hinge pin retainer bolts and remove bolts.

(5) Remove hinge pins from center hinge and outboard hinge fittings.

(6) Remove bonding braid from center hinge fitting.

4. Installing Elevator Hinge Pin

A. Install elevator hinge pin as follows:

(1) Install bonding braid to center hinge fitting. Make sure bonding braid is in good condition and that surfaces at bonding braid attach points are free of paint or any material that would not permit proper bonding.

(2) Insert outboard hinge pin through outboard fitting. Align center hinge fitting, and insert center hinge pin.

(3) Install hinge pin retainer bolts and safety with MS20995F41 lockwire.

(4) Install elevator tip and center hinge pin access door.

5. Removing Elevator Control Bobweight Balance Bungee

CAUTION: MAINTENANCE OF BUNGEE IS LIMITED TO CHECKS AND ROD-END ADJUSTMENTS REFERRED TO IN THIS CHAPTER. THE LOCKWIRE ON THE BUNGEE HOUSING MUST NOT BE BROKEN. INTERNAL ADJUSTMENT OF THE BUNGEE MUST NOT BE ATTEMPTED.

A. Parts referred to in the following procedure are identified in figure 27-30-402.

(1) Disconnect bungee rod end from bobweight balance bungee bell crank.

(2) Disconnect bungee end fitting from bracket on fuselage frame.

6. Installing Elevator Control Bobweight Balance Bungee

See figure 27-30-402.

7. Removing Elevator Control Bias Bungee

CAUTION: MAINTENANCE OF BUNGEE IS LIMITED TO CHECKS AND ROD-END ADJUSTMENTS REFERRED TO IN THIS CHAPTER. THE LOCKWIRE ON THE BUNGEE HOUSING OR ON THE ROD STOP MUST NOT BE BROKEN. INTERNAL ADJUSTMENT OF THE BUNGEE MUST NOT BE ATTEMPTED.

A. To remove elevator control bias bungee, proceed as follows:

(1) Through access doors, loosen bungee rod end, and rotate rod to allow rod stop to ride on housing of bungee, to relieve preload.

(2) Disconnect bungee rod end from aft cable bell crank. Remove rig pins, and rotate bell crank to pull bolt.

(3) Disconnect bungee end fitting from bracket on truss.

8. Installing Elevator Control Bias Bungee

See Figure 27-30-403.

9. Rigging Elevator Control System

See Figure 27-30-404.

10. Repair and Replacement of Elevator Universal Joint Hinge Support

Elevator inboard hinge supports Part No. 285-220401-3 or 285-220401-4 must be replaced when worn beyond acceptable limits. (Refer to "Elevator Universal Joint Dimensions and Tolerances" chart.) Elevator inboard hinge supports Part No. 285-220401-7 and -8 have a steel bushing riveted in the upper bearing area to reduce wear. When wear limits are exceeded in this area, the steel bushing may be removed from the hinge support and replaced. If only the hinge support is to be removed and replaced, refer to items A. and D. in the following procedures. If steel bushing or lower bearing assembly of hinge support is to be removed and replaced, refer to items B. and C. in the following procedures.

A. To remove elevator universal joint hinge support, see figure 27-30-405 and proceed as follows:

(1) Remove fuselage access panels above and below elevator to gain access to elevator torque tube.

(2) Remove applicable elevator.

(3) Remove cotter pin, nut, washers, and AN4-41A vertical bolt securing elevator torque tube universal joint to elevator inboard support hinge.

(4) Remove through bolt attaching support hinge to horizontal stabilizer root rib bearing. Remove support hinge and retain universal joint bushing.

B. To replace inboard hinge support upper bushing, see figure 27-30-405 and proceed as follows:

(1) With universal joint hinge support removed from airplane, remove upper bearing and carefully drill out rivet securing steel bushing in upper bearing area of hinge support.

MAINTENANCE PRACTICES (Continued)

NOTE

Use extreme care when drilling out attach rivet. If the rivet hole becomes oversize or elongated when the rivet is being removed, the hinge support must be replaced.

2. Remove steel bushing by adequately supporting lug areas of hinge support and tapping lightly with hammer and punch, or use suitable tool to press bushing out.
3. Clean area of any rivet chips and install a new bushing Part No. 285-220406-3 or -5.

NOTE

The 285-220406-5 bushing is of the latest design and is made of stainless steel for improved corrosion resistance.

The bushing is a light press fit to the hinge support. Adequately support hinge lug area when installing bushing. If 285-220406-5 bushing is being installed, coat MS20470AD5-10 attach rivet with wet zinc-chromate primer and install. Oversize rivets are not to be used; squeeze rivet to fill hole if required.

4. Install new AN201KP8A bearing in bushing.

C. To replace inboard hinge support lower bearing, see Figure 27-30-405 and proceed as follows:

1. With universal joint hinge support removed from airplane, remove the four nuts and washers securing lower bearing retainer to hinge support.
2. Remove 265-220404 bearing retainer and bearing.
3. Clean bearing area of any foreign matter and install new AN200KP8 bearing.
4. Install bearing retainer, washers, and the four NAS129-3 nuts. Torque retainer attach screws to 25-35 inch-pounds. With screws torqued, a gap of 0.000 to 0.020 inch between the retainer and hinge support is acceptable.

D. To install elevator universal joint hinge support, see figure 27-30-405 and proceed as follows:

1. Install through bolt which attaches support hinge to horizontal stabilizer root rib bearing. With bolt head outboard, install one washer under bolt head and one washer under nut. Tighten nut to torque value of 480 to 690 inch pounds, and install cotter pin.
2. Install bushing in universal joint and align support hinge fitting with torque tube assembly. Install vertical AN4-41A bolt, washers, and retaining nut. Install one 2W25P-17-26-125 washer under bolthead. Install one 2W18-416 washer between remaining 2W25P-17-26-125 washer and retaining nut. Tighten nut to 50-70 inch-pounds and install cotter pin.
3. Install elevator. (See Figure 27-30-401.)
4. Check completed installation for proper safetying and security of attachment. Install fuselage access panels as required. Operate elevator and check for proper clearances.

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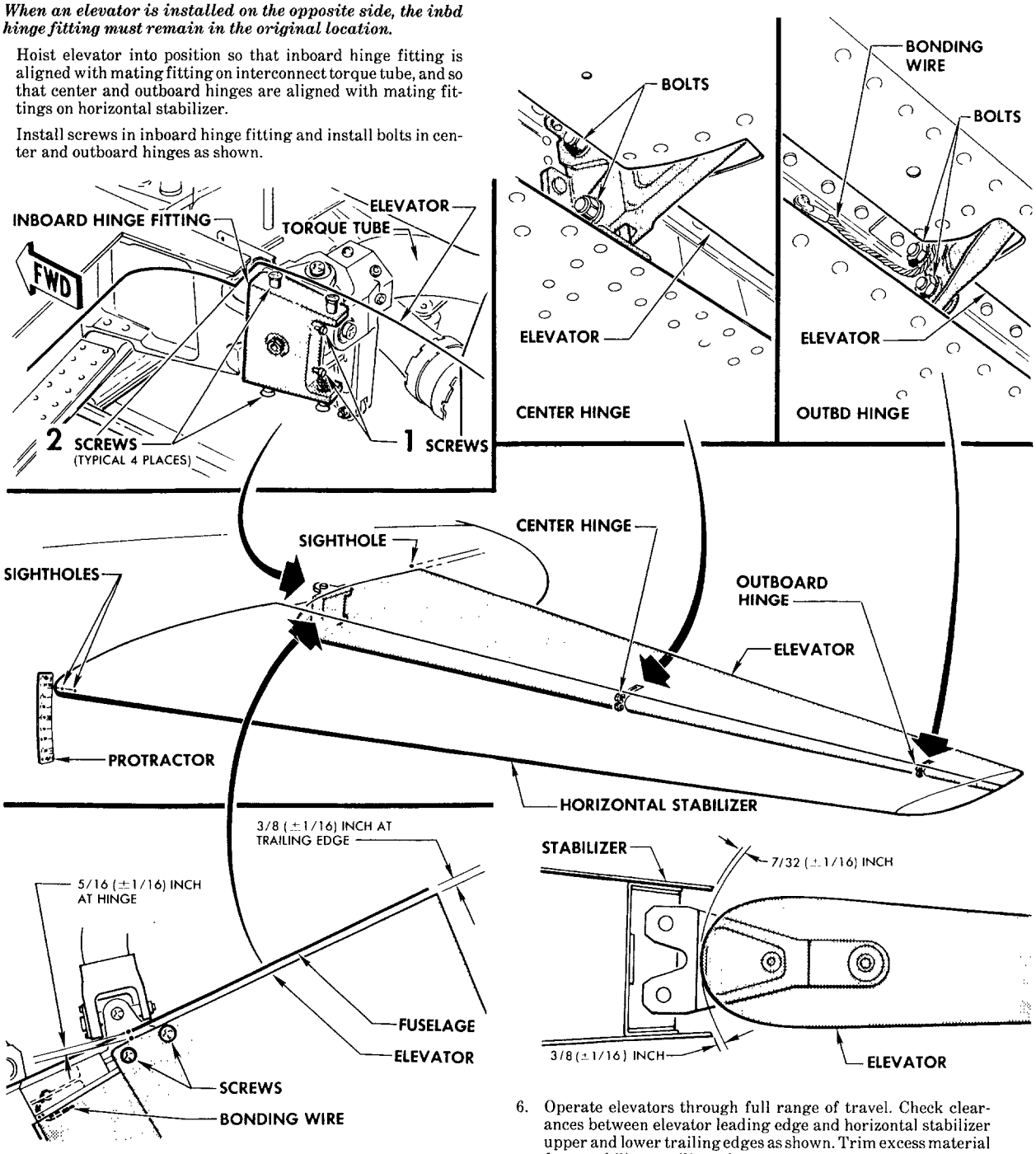
NOTES

- ***The left and right elevator installation procedures are the same.***
- ***When an elevator is installed on the opposite side, the inbd hinge fitting must remain in the original location.***

1. Hoist elevator into position so that inboard hinge fitting is aligned with mating fitting on interconnect torque tube, and so that center and outboard hinges are aligned with mating fittings on horizontal stabilizer.
2. Install screws in inboard hinge fitting and install bolts in center and outboard hinges as shown.
3. Torque screws and bolts.
4. Connect bonding wire at inboard and outboard hinges.



5. Operate horizontal stabilizer to 0° on protractor. Align elevator trailing edge with sight-hole on fuselage. Check that clearances between elevator and fuselage are as shown. Trim excess material from elevator inboard edge.
6. Operate elevators through full range of travel. Check clearances between elevator leading edge and horizontal stabilizer upper and lower trailing edges as shown. Trim excess material from stabilizer trailing edges.
7. Inspect all connections. Make sure that area is free of tools, rags, and loose hardware.

S370-2-52-37A

Figure 27-30-401 — Installing Elevators

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NOTE The elevator control system must be properly rigged before bungee is installed.

1 Disconnect elevator control bias bungee on left side of lower part of fuselage at rear section, canted station 457.

2 Remove bolt connecting push rod to bobweight balance bungee bell crank.

3 Connect end of bungee to bracket on frame by installing bolt.

4 Connect bungee rod end to bungee bell crank by installing bolt.

5 Reconnect push rod to bungee bell crank as follows: With control wheels held in full-aft position, rotate bungee bell crank until push rod end is aligned with hole in bell crank. Install bolt removed in step 2.

6 Release control wheels. Alternately apply small push-pull forces to pilot's control wheel, noting where wheel comes to rest; then place control wheel in center of "dead band."

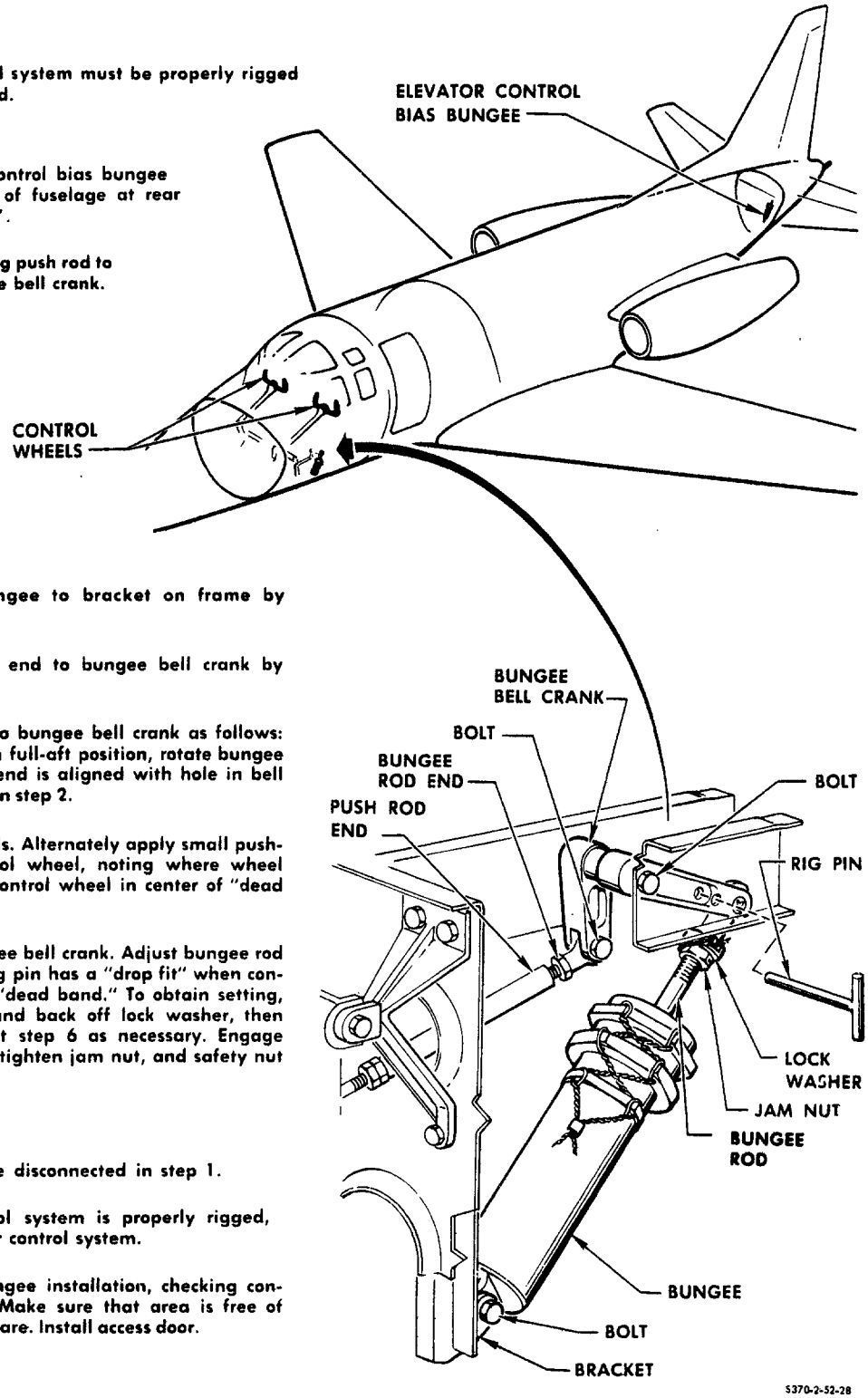
7 Insert rig pin in bungee bell crank. Adjust bungee rod end as necessary so that rig pin has a "drop fit" when control wheel is in center of "dead band." To obtain setting, loosen rod-end jam nut, and back off lock washer, then rotate rod end and repeat step 6 as necessary. Engage lock washer with rod end, tighten jam nut, and safety nut to lock washer.

8 Remove rig pin.

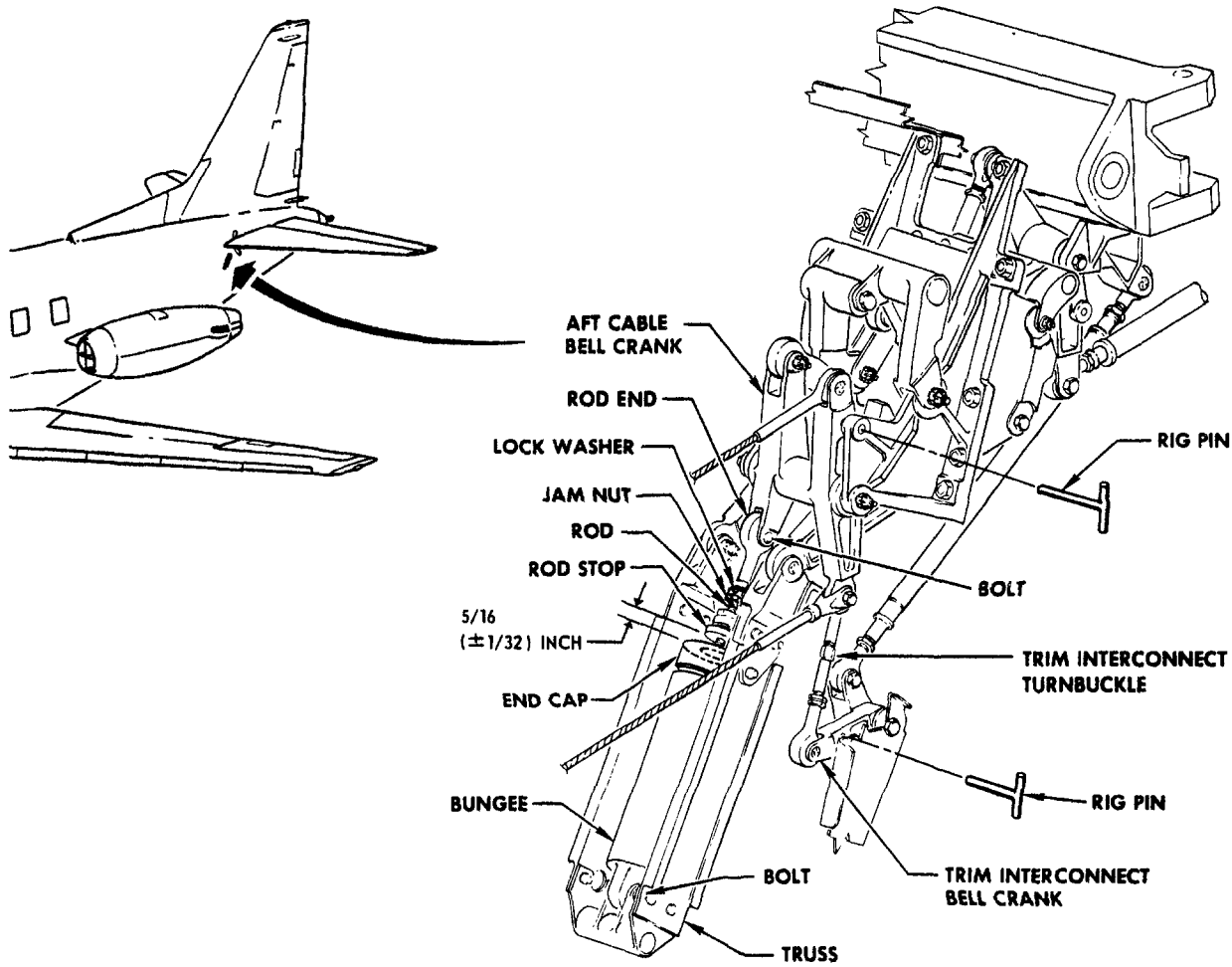
9 Reinstall bias bungee disconnected in step 1.

10 After elevator control system is properly rigged, check operation of elevator control system.

11 Inspect complete bungee installation, checking connections and adjustment. Make sure that area is free of rags, tools, and loose hardware. Install access door.



Installing Elevator Control Bobweight Balance Bungee
Figure 27-30-402



1 Operate horizontal stabilizer leading edge to 4-1/2 degrees (4°30') on protractor.

2 Through accesses, connect end of bungee to truss on bulkhead by installing bolt.

3 *With bungee rod end extended as necessary, connect* bungee to aft cable bell crank by installing bolt.

4 Through access, insert rig pin in trim interconnect bell crank.

5 Insert rig pin in aft cable bell crank.

6 Rotate bungee rod inside rod end to provide 5/16 (± 1/32) inch gap between bungee end cap and rod stop.

Caution Do not break lockwire on bungee housing or on rod stop.

7 Check that rig pin in aft cable bell crank has a "drop fit." To obtain "drop fit," adjust trim interconnect turnbuckle and lockwire.

8 Recheck gap on bungee. To obtain setting, adjust rod end again as necessary. Engage lock washer with rod end, tighten jam nut, and safety nut to lock washer.

9 Remove rig pins.

10 Inspect complete bungee installation, checking connections and adjustment. Make sure that area is free of rags, tools, and loose hardware. Install access door.

S370-2-52-29

Installing Elevator Control Bias Bungee
Figure 27-30-403

NOTE Use 5/16-inch diameter rods or bolts as rig pins. Maintain a "drop fit" on all rig pins, to eliminate forced alignment of rods or bell cranks.

- Before rigging elevator control system, the horizontal stabilizer trim system must be properly rigged. (Refer to "Rigging Horizontal Stabilizer Trim System.")

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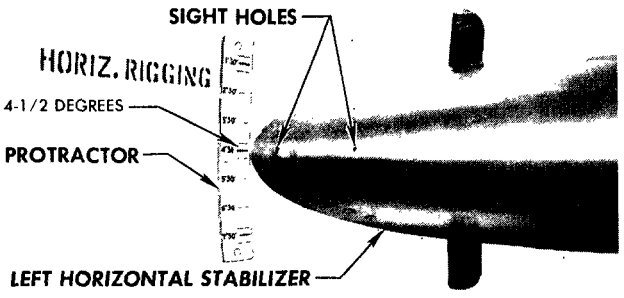
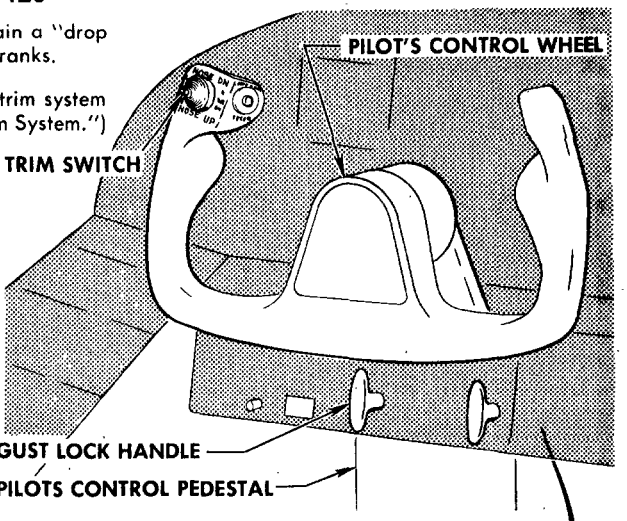
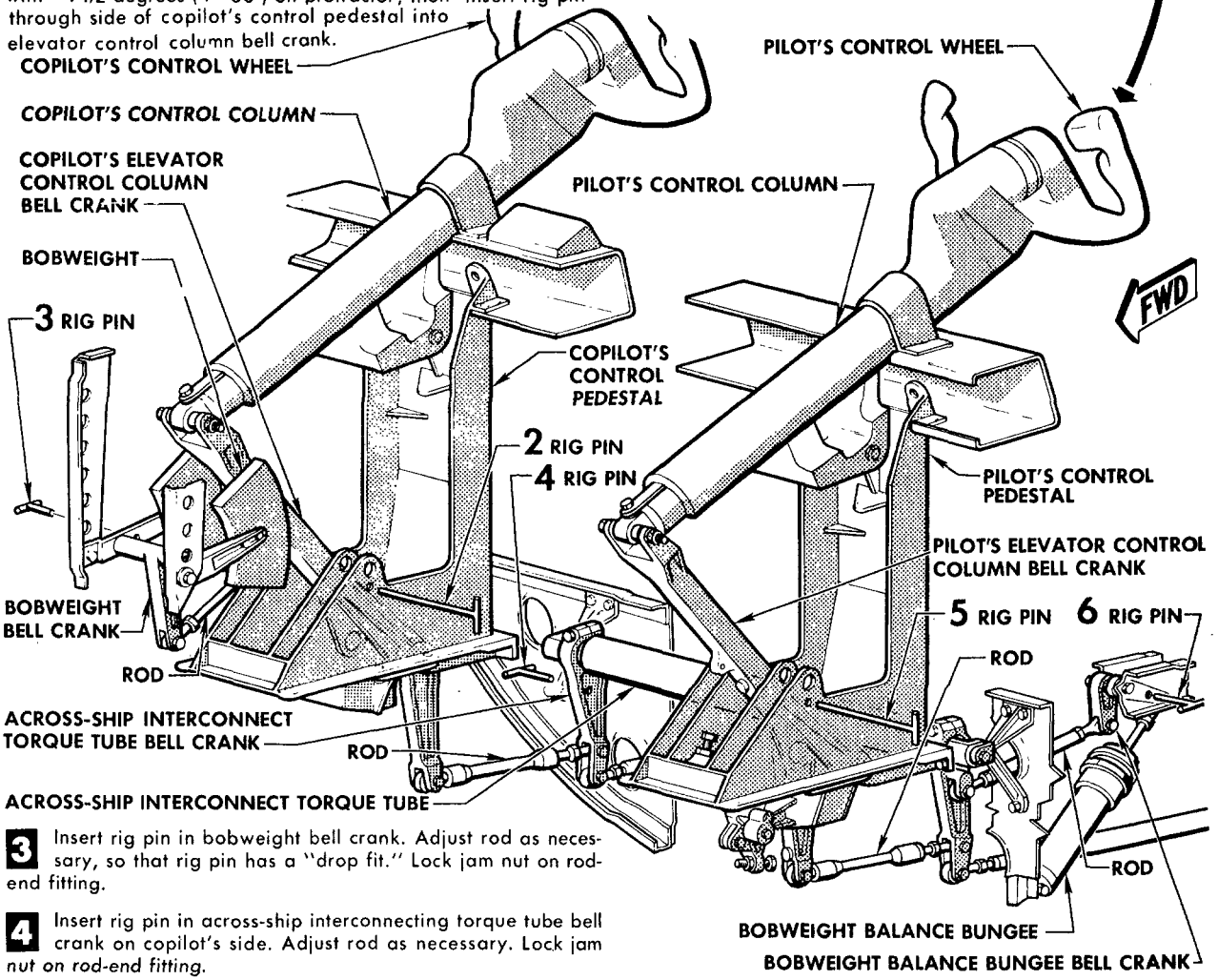




Caution Before operating horizontal stabilizer, check that all rig pins are removed from elevator control system and trim interconnect bell crank.

1 Move gust lock handle to the disengaged position.

2 Operate horizontal stabilizer leading edge until sight holes on inboard leading edge of left horizontal stabilizer align with -- 4 1/2 degrees (4° 30') on protractor; then insert rig pin through side of copilot's control pedestal into elevator control column bell crank.



3 Insert rig pin in bobweight bell crank. Adjust rod as necessary, so that rig pin has a "drop fit." Lock jam nut on rod-end fitting.

4 Insert rig pin in across-ship interconnecting torque tube bell crank on copilot's side. Adjust rod as necessary. Lock jam nut on rod-end fitting.

5 Insert rig pin through outboard side of pilot's control pedestal into elevator control column bell crank. Adjust rod as necessary. Lock jam nut on rod-end fitting.

6 Insert rig pin in bobweight balance bungee bell crank. Adjust rod as necessary to install rig pin. A "drop fit" is not necessary at this time. Lock jam nut on rod-end fitting.

S370-2-52-30 A

Rigging Elevator Control System
Figure 27-30-404 (Sheet 1 of 7)

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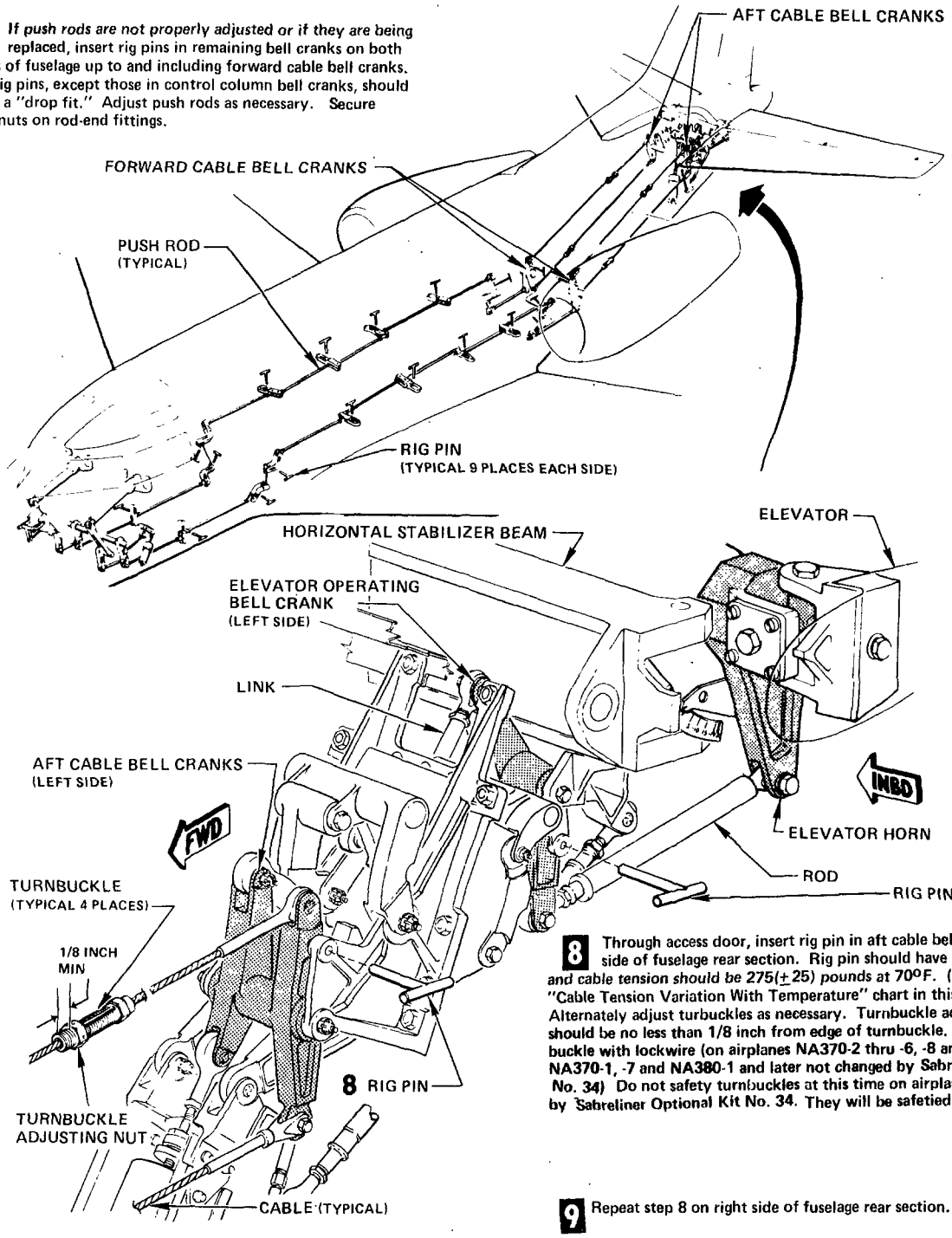
7 *If push rods are not properly adjusted or if they are being* replaced, insert rig pins in remaining bell cranks on both sides of fuselage up to and including forward cable bell cranks. All rig pins, except those in control column bell cranks, should have a "drop fit." Adjust push rods as necessary. Secure jam nuts on rod-end fittings.



8 Through access door, insert rig pin in aft cable bell crank on left side of fuselage rear section. Rig pin should have a "drop fit" and cable tension should be 275(±25) pounds at 70°F. (Refer to "Cable Tension Variation With Temperature" chart in this Chapter. Alternately adjust turbuckles as necessary. Turnbuckle adjusting nut should be no less than 1/8 inch from edge of turnbuckle. Safety turnbuckle with lockwire (on airplanes NA370-2 thru -6, -8 and -9, and NA370-1, -7 and NA380-1 and later not changed by Sabreliner Optional Kit No. 34) Do not safety turnbuckles at this time on airplanes changed by Sabreliner Optional Kit No. 34. They will be safetied in a later step.

9 Repeat step 8 on right side of fuselage rear section.

10 Through access doors, insert rig pins in elevator operating bell cranks on left and right sides. Adjust links as necessary. Lock jam nuts on rod end fittings.

S370-2-52-31A

Rigging Elevator Control System
Figure 27-30-404 (Sheet 2 of 7)

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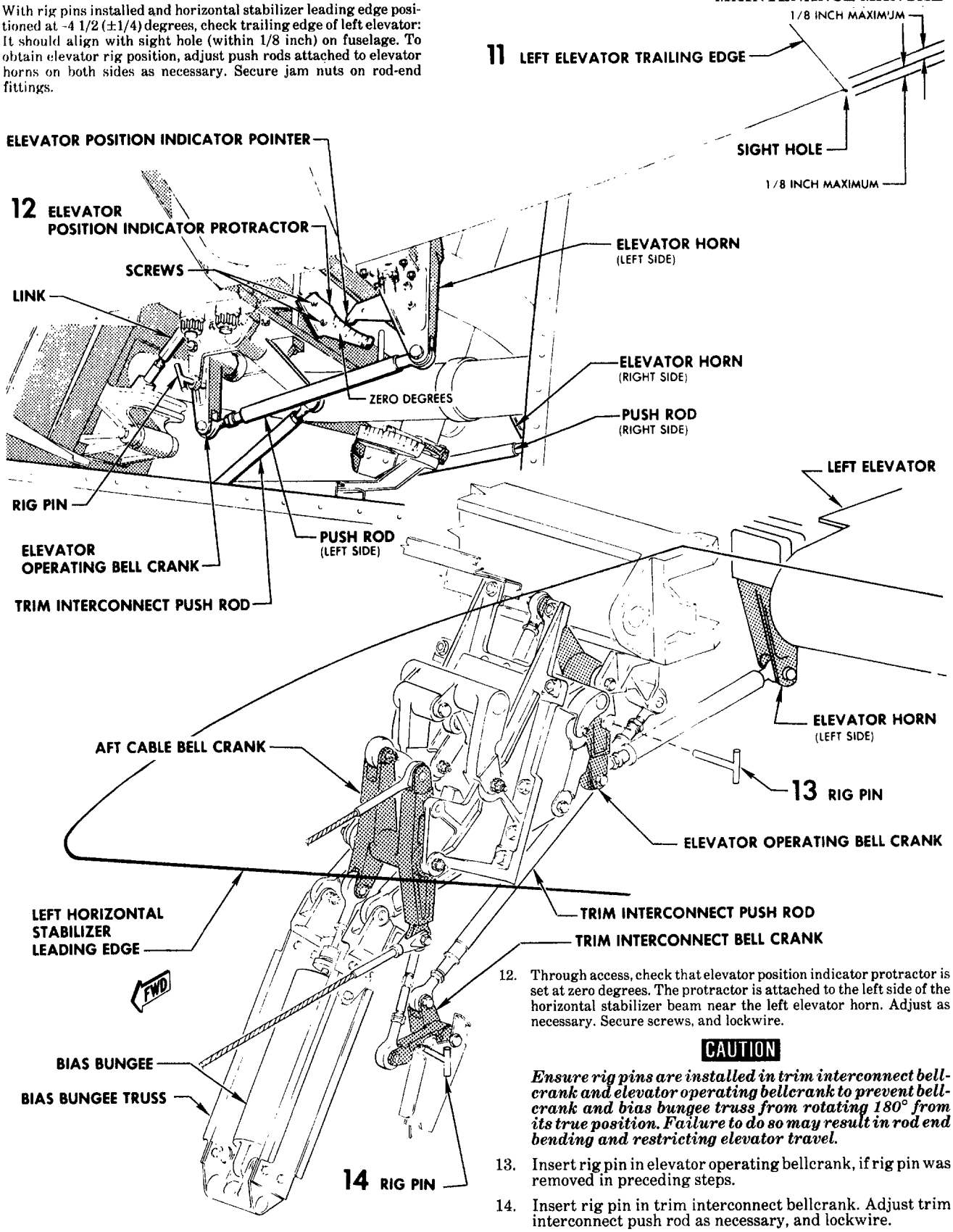
11. With rig pins installed and horizontal stabilizer leading edge positioned at -4 1/2 (±1/4) degrees, check trailing edge of left elevator: It should align with sight hole (within 1/8 inch) on fuselage. To obtain elevator rig position, adjust push rods attached to elevator horns on both sides as necessary. Secure jam nuts on rod-end fittings.



12. Through access, check that elevator position indicator protractor is set at zero degrees. The protractor is attached to the left side of the horizontal stabilizer beam near the left elevator horn. Adjust as necessary. Secure screws, and lockwire.

CAUTION

Ensure rig pins are installed in trim interconnect bellcrank and elevator operating bellcrank to prevent bellcrank and bias bungee truss from rotating 180° from its true position. Failure to do so may result in rod end bending and restricting elevator travel.

13. Insert rig pin in elevator operating bellcrank, if rig pin was removed in preceding steps.

14. Insert rig pin in trim interconnect bellcrank. Adjust trim interconnect push rod as necessary, and lockwire.

Figure 27-30-404 — Rigging Elevator Control System (Sheet 3 of 7)

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NOTE

The trim interconnect push rod may be adjusted without disconnecting its ends.

15. With rig pin in aft cable bellcrank, check that gap between bias bungee end cap and rod stop is 5/16 (±1/32) inch.

    To obtain dimension "A", proceed as follows:

    A. Loosen bungee rod-end jam nut and back off lock washer; then rotate rod inside rod end as necessary to obtain setting.

CAUTION

Do not break lockwire on bungee housing or on rod stop.

    B. Remove all rig pins.

    C. Adjust trim interconnect turnbuckle so that a rig pin inserted in aft cable bellcrank has a "drop fit." Lockwire turnbuckle assembly.

    D. Recheck gap on bungee. Adjust bungee rod end again as necessary, to obtain setting. Engage lock washer with rod end, tighten jam nut, and safety nut to lock washer.

    E. Remove rig pin.

16. Check that bobweight balance bungee is installed; then insert rig pin in bobweight balance bungee bellcrank. Adjust rod end of bobweight balance bungee so that rig pin has a "drop fit." Remove rig pin.

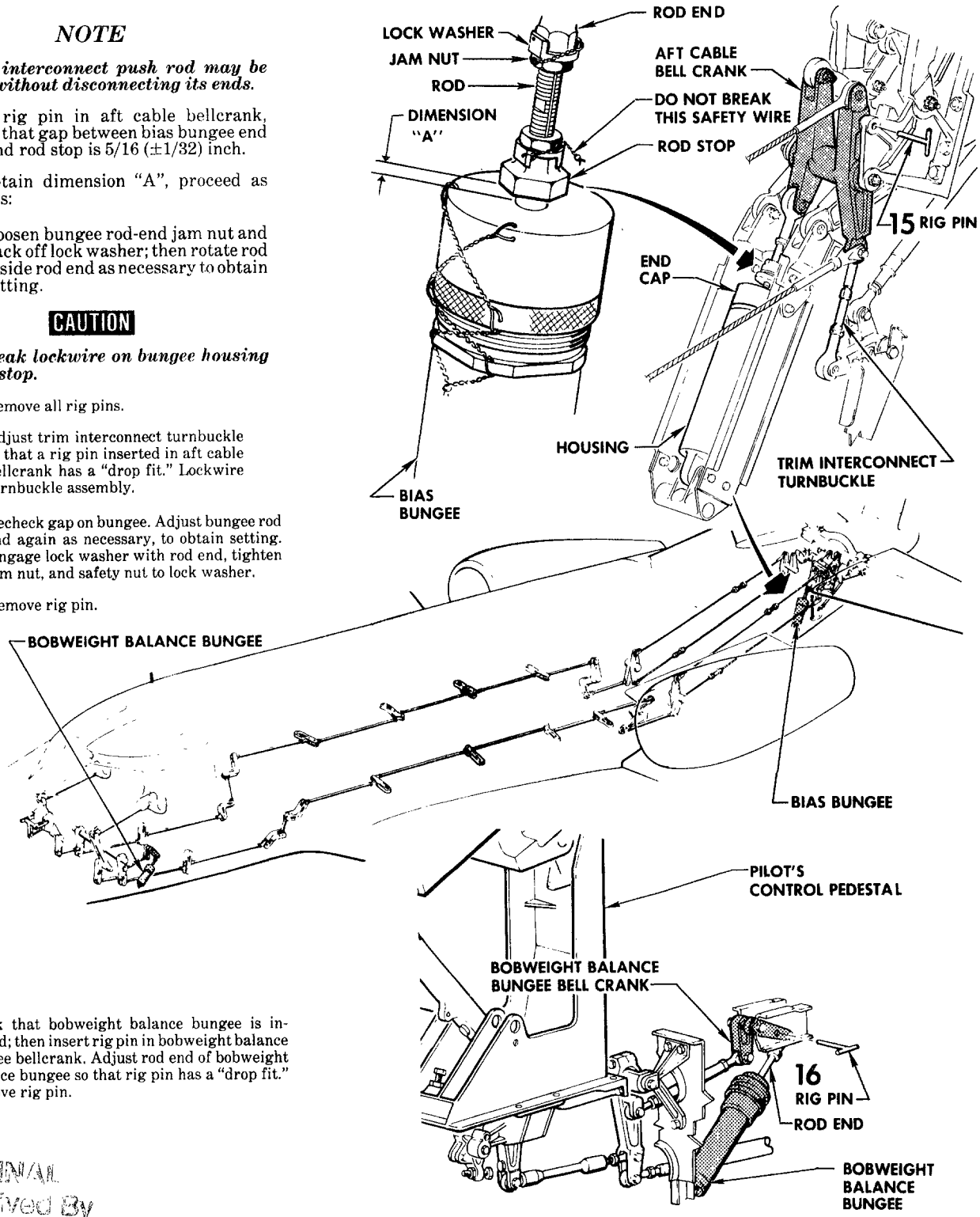

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Figure 27-30-404 — Rigging Elevator Control System (Sheet 4 of 7)

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As Received By
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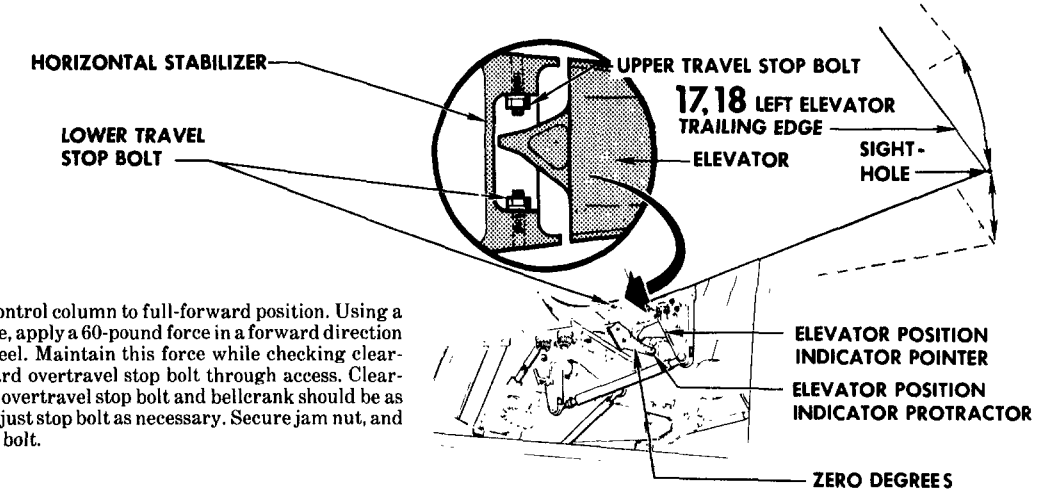
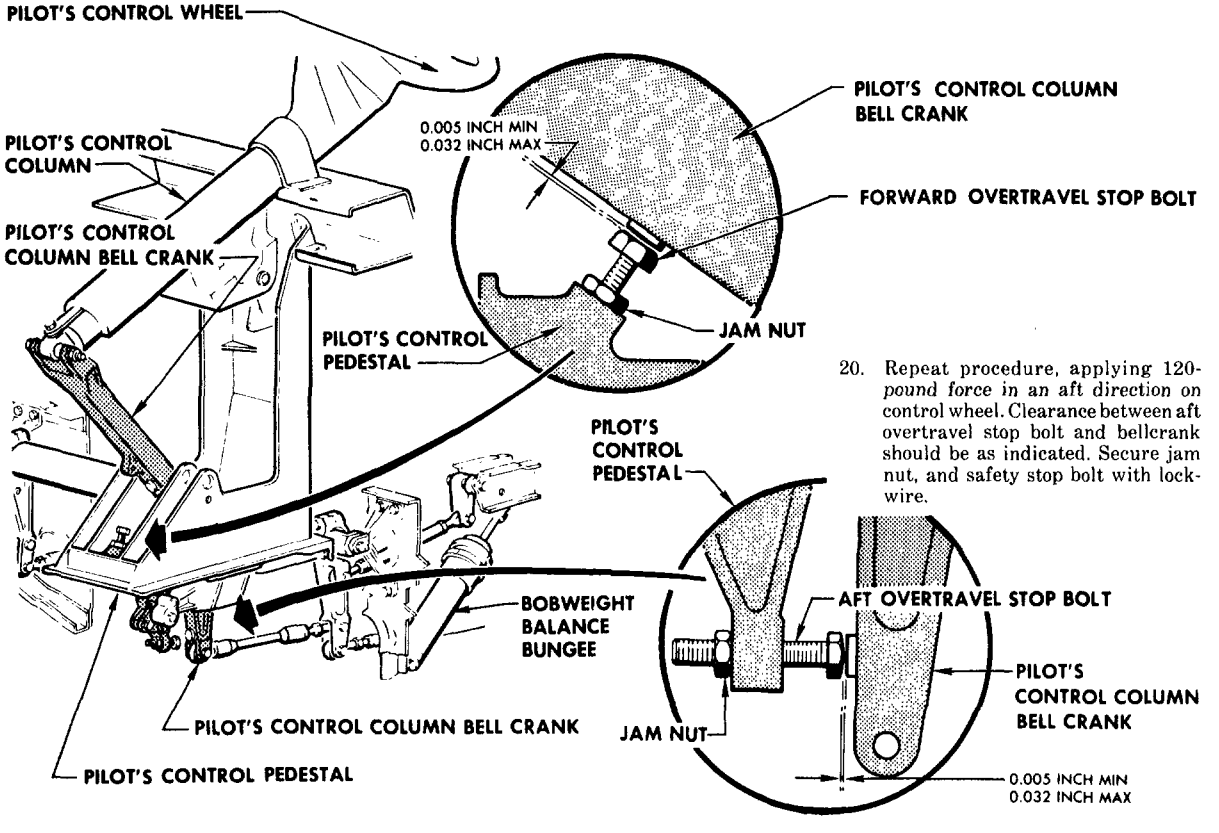
17. Operate elevator trailing edge to full up position. Adjust lower stop bolts on stabilizer as necessary to limit elevator trailing edge uptravel to 22 (±1/2) degrees (6-31/32 inches minimum to 7-3/8 inches maximum, measured from elevator trailing edge centerline to sighthole on fuselage). Secure jam nuts on stop bolts; then check that elevator indicator indicates surface position within ±1/2 degree.

18. Operate elevator trailing edge to full down position. Adjust upper stop bolts on stabilizer as necessary to limit elevator trailing edge down-travel to 11 (±1/2) degrees (3-19/32 inches min to 3-15/16 inches maximum, measured from elevator trailing edge centerline to sighthole on fuselage). Secure jam nuts on stop bolts; then check that elevator position indicator indicates surface position within ±1/2 degree.



19. Push pilot's control column to full-forward position. Using a precision scale, apply a 60-pound force in a forward direction to control wheel. Maintain this force while checking clearance of forward overtravel stop bolt through access. Clearance between overtravel stop bolt and bellcrank should be as indicated. Adjust stop bolt as necessary. Secure jam nut, and lockwire stop bolt.



20. Repeat procedure, applying 120-pound force in an aft direction on control wheel. Clearance between aft overtravel stop bolt and bellcrank should be as indicated. Secure jam nut, and safety stop bolt with lockwire.

370-2-52-34

Figure 27-30-404 — Rigging Elevator Control System (Sheet 5 of 7)

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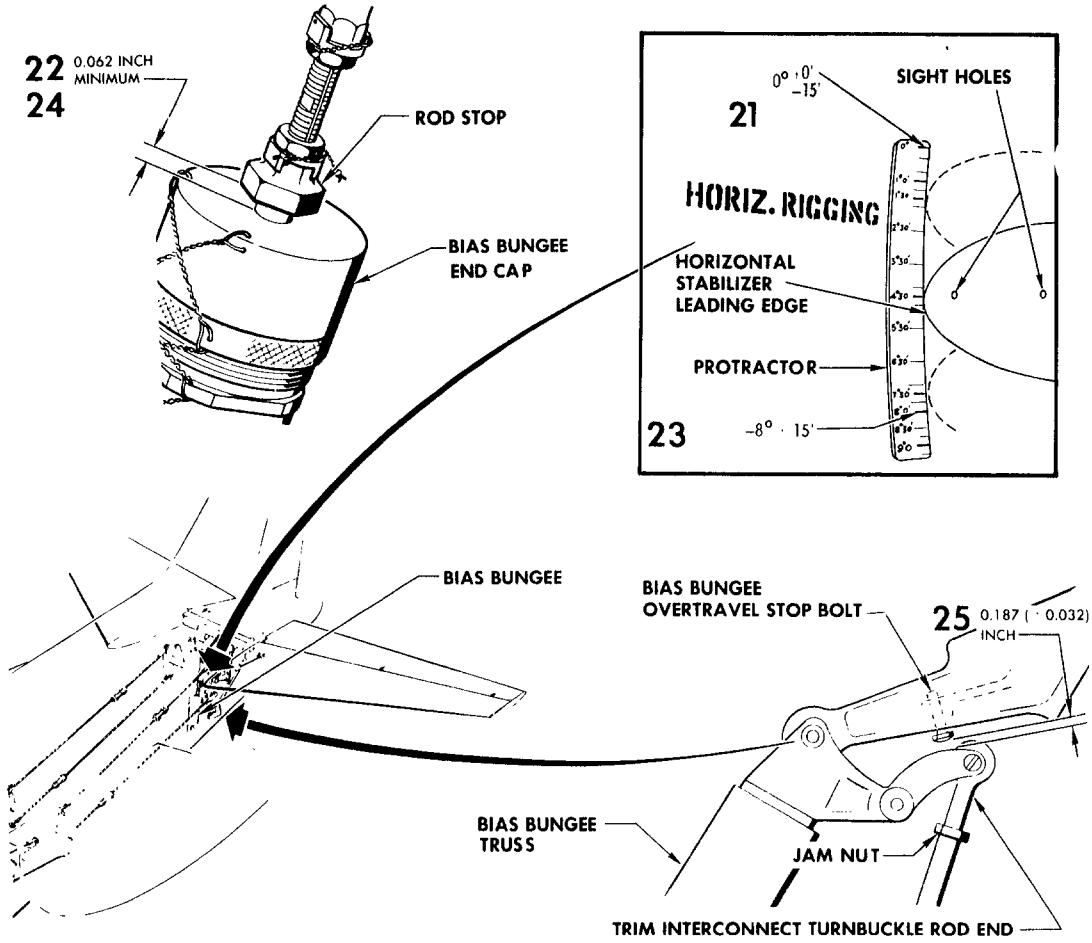

CAUTION

Before operating horizontal stabilizer, check that all rig pins are removed from elevator control system and trim interconnect bellcrank.

CAUTION

Ensure a minimum clearance of 0.062 in. is obtained to prevent rod stop from bottoming out against the bungee end cap and bending the bungee rod before operating horizontal stabilizer.

21. Operate horizontal stabilizer to full NOSE DN position. Check that sight holes on leading edge of left horizontal stabilizer align with protractor as shown.
22. Move elevator controls through full range of travel. Check that gap between bias bungee end cap and rod stop is no less than 0.062 inch. If adjustment is necessary, repeat steps 1 and 15.
23. Operate horizontal stabilizer to full NOSE UP position. Check that sightholes on leading edge of horizontal stabilizer align with protractor as shown.
24. Move elevator controls through full range of travel. Check that gap between bias bungee end cap and rod stop is not less than 0.062 inch. If adjustment is necessary repeat steps 1 and 15.
25. With stabilizer at 8° and elevator trailing edge up 8°, adjust bias bungee overtravel stop bolt to 0.187 (±0.032) inch from upper rod end of trim interconnect turnbuckle. Secure jam nuts and lockwire stop bolt.
    - S/N 370-2 thru -6, -8 and -9, S/N 370-1, -7 and S/N 380-1 and Subs not changed by SOK 34. Check elevator free play as follows:
      A. Install LH aft cable bellcrank rig pin 27.
      B. Mount an inclinometer on the center hinge fitting on the elevator.
      C. Apply a 5-pound up-and-down load to trailing edge of elevator and record both deflections. Difference should not exceed 0°40′. Remove rig pin.
      D. Proceed to step 34.
    - S/N 370-1 thru -9, and S/N 380-1 and Subs changed by SOK 34, perform the following steps:

Figure 27-30-404 — Rigging Elevator Control System (Sheet 6 of 7)

S370-2-52-35B

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26. Trim stabilizer to -4°30′.
27. Insert aft cable bellcrank rig pins (both LH and RH sides). Adjust upper LH cable to 300+ 30 pounds tension and lower LH cable to 250+ 25 pounds tension. Adjust upper RH cable to 275± 25 pounds tension and the lower RH cable to 395± 40 pounds tension. Refer to cable tension chart for temperature variations. Check that turnbuckle adjusting nut is no less than 1/8 inch from edge of turnbuckle. Safety turnbuckle with lockwire.

NOTE

All cable tensions are at 70°F (Refer to Cable Tension Variation with temperature chart in this chapter.)

28. Check forward bellcrank rig pins for free fit. A 1/8 (.125) diameter rig pin may be used for this check only. Remove all rig pins.
29. With stabilizer trimmed to -4°30′, insert LH and RH rig pins 29. Remove and extend LH rod one-half turn. Remove and shorten RH rod one-half turn. Reconnect both rods.
30. Check that cable tensions are as depicted in step 27. The final bias between the upper LH cable and the lower LH cable should be between 20 and 60 pounds, and 60 to 180 pounds between the RH upper and RH lower cables.

Check elevator free play as follows:

A. Install LH aft cable bellcrank rig pin 27.
B. Mount an inclinometer on the center hinge fitting on the elevator.
C. Apply a 5 pound up-and-down load to trailing edge of elevator and record both deflections. Difference should not exceed 0° 20′. Remove rig pin.

31. Repeat step 30 on RH side.
32. If difference exceeds 0° 20′, insert both LH and RH elevator operating bellcrank rig pins and extend the LH (lengthen) rod (29) one-half turn and shorten the RH rod one-half turn. Remove all rig pins.
33. Repeat step 30.
34. Do procedures in Checking Elevator Control System.
35. Conduct a complete inspection of elevator control system. Check that, through full range of travel, all moving parts do not bind or touch structure. Check security of all jam nuts. Check proper safetying of all bolts and turnbuckles. Make sure that areas are free of tools, rags, and loose hardware. Install access doors.

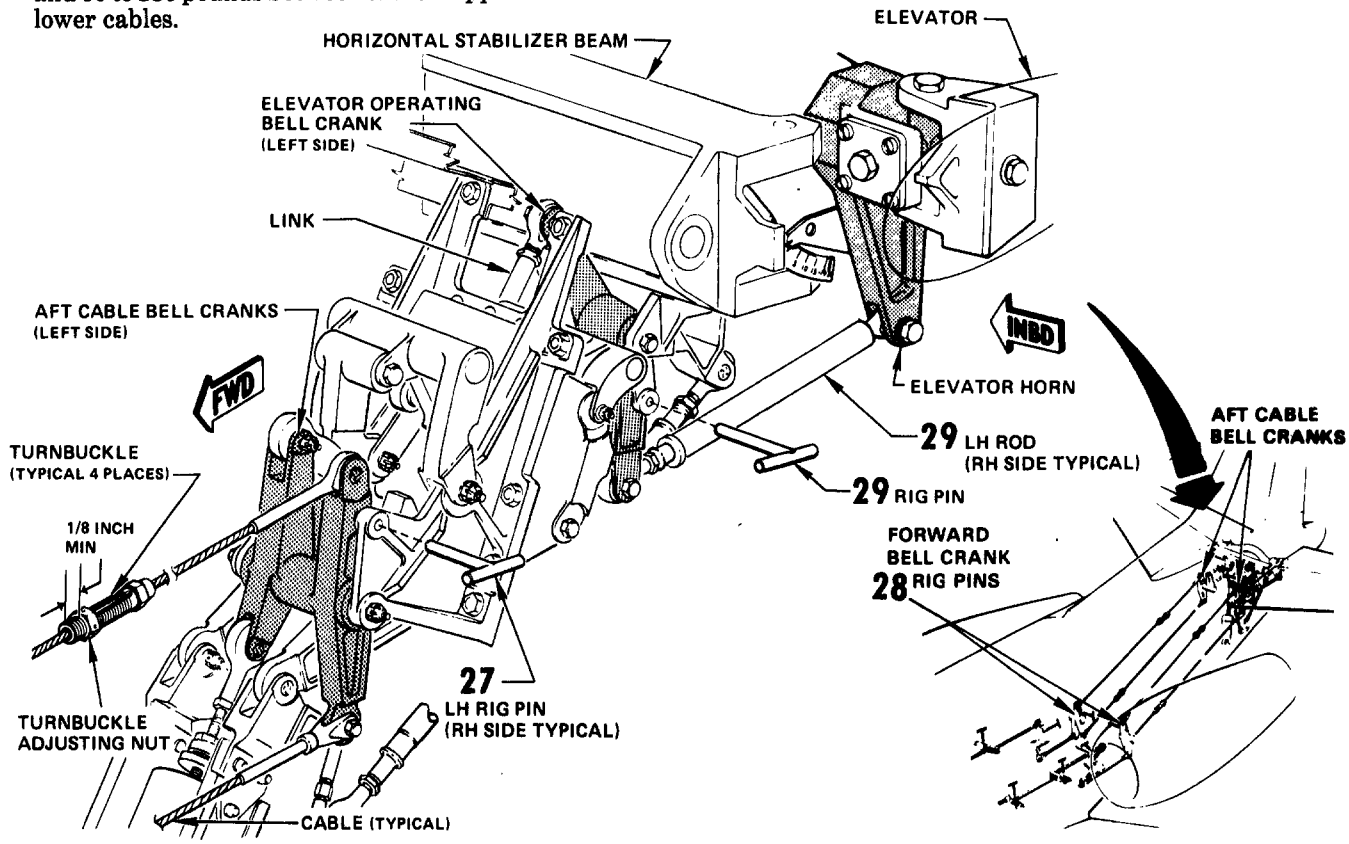

Figure 27-30-404 — Rigging Elevator Control System (Sheet 7 of 7)

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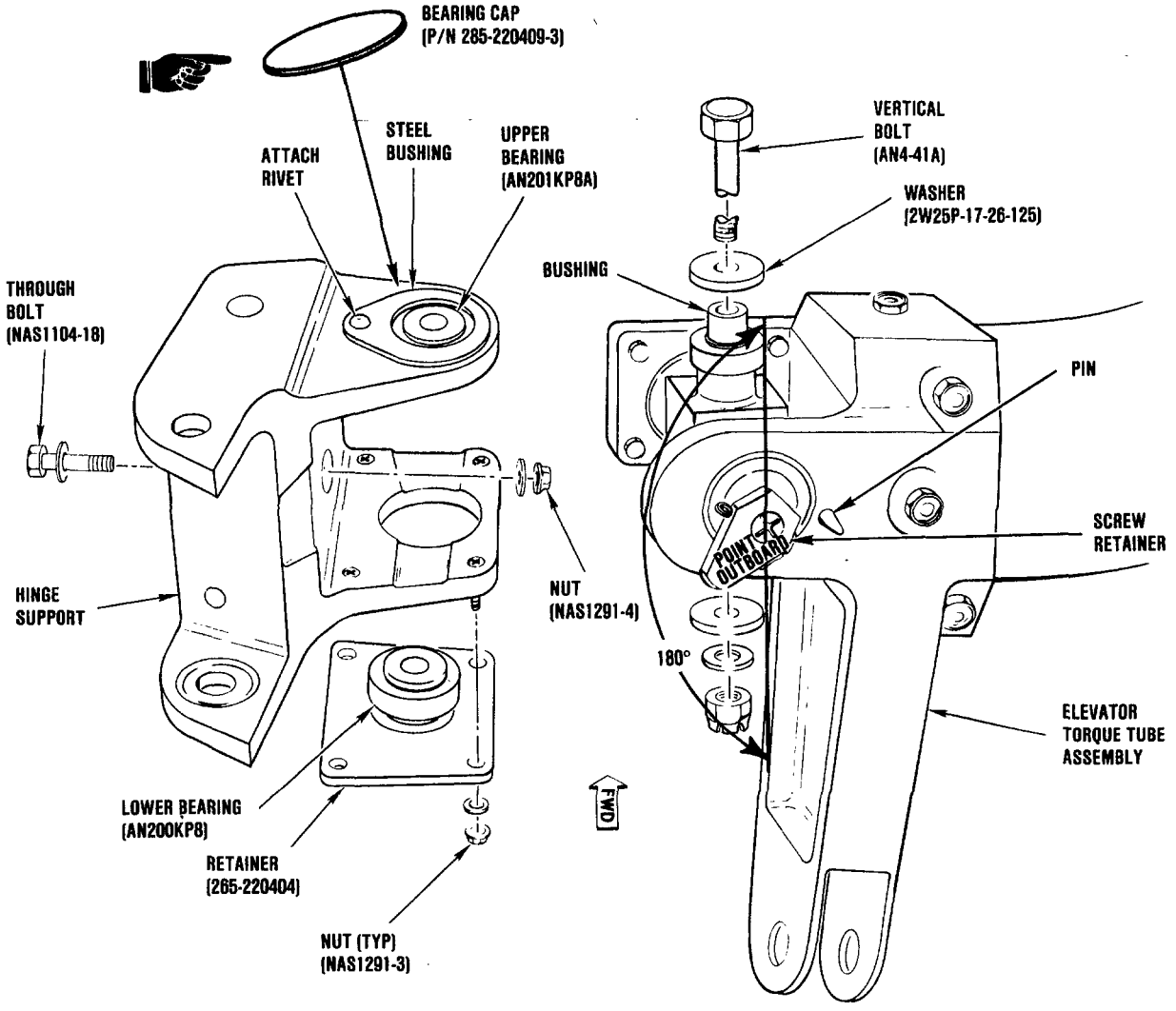

Figure 27-30-405 — Elevator Universal Joint Hinge Support

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1. Checking Elevator Control System

A. Before check-out, the elevator control system and horizontal stabilizer trim system must be properly rigged. (See figures 27-30-404 and 27-40-405.) If check-out does not meet requirements, refer to trouble shooting procedures for the elevator system.

(1) Make sure that gust lock is disengaged; then move elevator controls through full range of travel, and check that they operate freely. There should be no sign of binding.

(2) Operate horizontal stabilizer leading edge until sight holes on inboard leading edge align with 4-1/2 degrees (4°30′) on protractor.

CAUTION: BEFORE OPERATING HORIZONTAL STABILIZER, MAKE SURE THAT ALL RIG PINS ARE REMOVED FROM ELEVATOR CONTROL SYSTEM AND THE TRIM INTERCONNECT BELL CRANK.

(3) Push control wheel to full-forward position. Check full travel of elevator: The elevator trailing edge should move down.

(4) Check elevator position indicator on left elevator horn. With the elevator trailing edge full down, the elevator position indicator should read $11 \pm \frac{1}{2}$ degrees. When the wheel is slowly released, the control system should return to neutral position.

(5) Pull control wheel to full-aft position, and repeat procedure. With the elevator trailing edge full up, the elevator position indicator should read $22 \pm \frac{1}{2}$ degrees. When the wheel is slowly released, the control system should return to neutral position.

(6) Trim horizontal stabilizer to position indicated in the following chart; then, using an accurate-reading push-pull type scale, push or pull control wheel to position elevators as indicated. Control system forces should fall within specified limits.

TRIM POS (DEG)	ELEV TRG EDGE POS (DEG)	CONT SYS FORCES (LB)
4-1/2	Up 21	48 (±7)
	Down 10	19 (±3)
	Up 21	62 (±8)
0	Down 10	0 (±5)
	Up 21	35 (±5)
8	Down 10	33 (±5)

(7) Install access.

2. Checking Elevator Control System Friction Forces

A. Friction forces may be checked with bias bungee disconnected from aft cable bell crank. Remove all rig pins; then, using an accurate-reading

push-pull type scale and a flexible tape rule, check friction forces as follows:

(1) Move pilot's control column to full-forward position.

(2) Attach scale to center of control wheel.

(3) By slowly and continuously pulling on scale, move control wheel in an aft direction.

(4) Record scale readings at one-inch increments from 1/2 inch of start position to 1/2 inch of full-travel position. Record all aft moving forces of wheel as "A" and all forward moving forces of wheel as "B." Record forces in a vertical row.

(5) Place control wheel in full-aft position and repeat check in opposite direction.

(6) By slowly and continuously pushing on scale, move control wheel forward.

(7) Record forces at same points. Record all forward moving forces of wheel as "B" and all aft moving forces of the wheel as "A." Record forces at same points in a vertical row from bottom to top, next to first row of forces.

(8) Compute friction forces as follows: For positions where force recorded in one direction is "A" and in opposite direction is "B," add these two forces. Then divide by two. Where both forces are either "A" or "B," subtract smaller force from larger force. Then divide by two. The results will be friction forces. Friction forces at any position must not exceed 6 pounds.

3. Checking Elevator Control Bobweight Balance Bungee

See figure 27-30-601.

4. Checking Elevator Control Bias Bungee

See figure 27-30-602.

5. Checking Elevator Universal Joint Fittings Free Play

A. Refer to Chapter 5 for intervals at which this check will be performed. See figure 27-30-603 for identification of components.

(1) Remove upper and lower flight control access panels on both sides of fuselage. (Refer to Chapter 6, figure 6-00-3.)

(2) Hold one elevator in a fixed position, and using a force gage, apply a force, not to exceed one pound, to move the opposite elevator trailing edge up and down.

(3) Measure up and down travel (free play) at trailing edge of elevator. Total deflection of the elevator trailing edge should not exceed 1/8 inch, and play should be evenly distributed among all joints in the left and right universal joint assemblies.

NOTE: Make sure fixed elevator does not move during measurement of opposite elevator free play.

(4) If free play at elevator trailing edge exceeds 1/8 inch, or if, while watching the universal joints, the measured play seems to be in only one or two of the joints in the assemblies, continue with step (5). If free play at the trailing edge is within 1/8 inch and the play seems to be evenly distributed among all the four joints in each assembly, skip steps (5) through (7) and proceed to step (8).

(5) Mount a dial indicator on one joint of the affected universal joint assembly so that free play between the shaft, bearing, and bearing housing can be measured (indicator clamped to bearing housing fitting and indicator contact point touching shaft nut or bolt).

(6) Using a force gage, apply a force, not to exceed one pound, to move elevator trailing edge up and down. Free play in the joint being measured should not exceed dimensions in column "G" of figure 27-30-604.

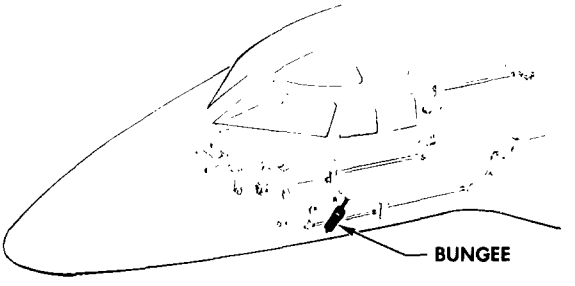
(7) Repeat steps (5) and (6) for each of the other joints in the assembly. If maximum tolerances are exceeded, the universal joint assembly must be removed, disassembled, and repaired or replaced as necessary.

NOTE: If free play in the entire joint assembly exceeds the total of the maximum figures in column "G" of figure 27-30-604, it may be necessary to replace several components in the assembly which do not themselves exceed the maximum limits in column "F" of the chart.

(8) If disassembly of one or both universal joints was necessary, steps (2) through (7) must be repeated after reassembly.

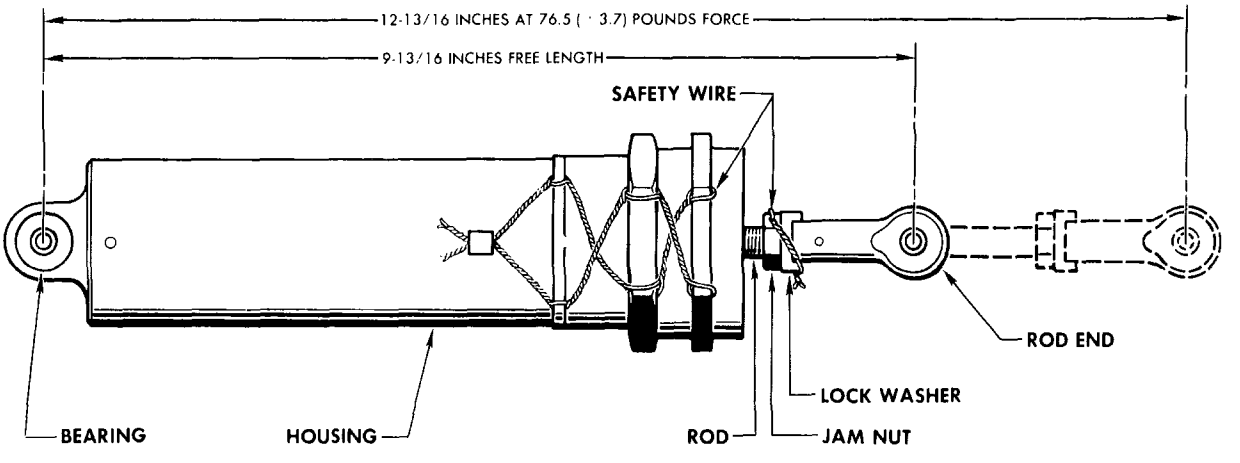
(9) Install access panels.

NOTE With the exception of the procedure in step 4, the following information is not to be used when the bobweight balance bungee is being adjusted. It is intended to furnish check-out information only, to determine whether the bungee is serviceable. The bungee is to be checked in the condition it is in when it is removed from the airplane. Safety wire on bungee housing must not be broken. Internal adjustments of bungee must not be attempted. If the bungee fails to meet any of the checks, it should be replaced with a new unit.



To check bungee, proceed as follows:

1. Visually check bungee assembly to determine that housing, rod, bearings, etc., are not damaged.
2. Check that bungee is free of dirt, grease, oil, and foreign matter.
3. Shake bungee to determine that loose objects are not inside bungee housing.
4. Check that free length of bungee is 9-13/16 inches.
5. Extend bungee (10 times) to 12-13/16 inches.
6. Check bungee force. The force required to extend the bungee to 12-13/16 inches should be 76.5 (±3.7) pounds.



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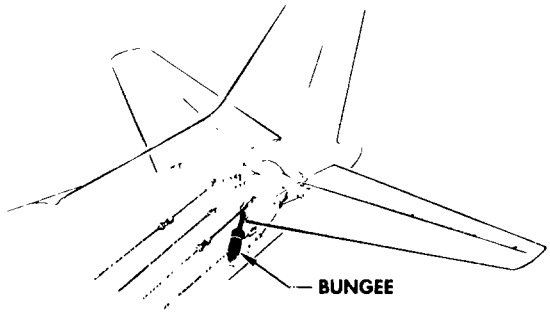
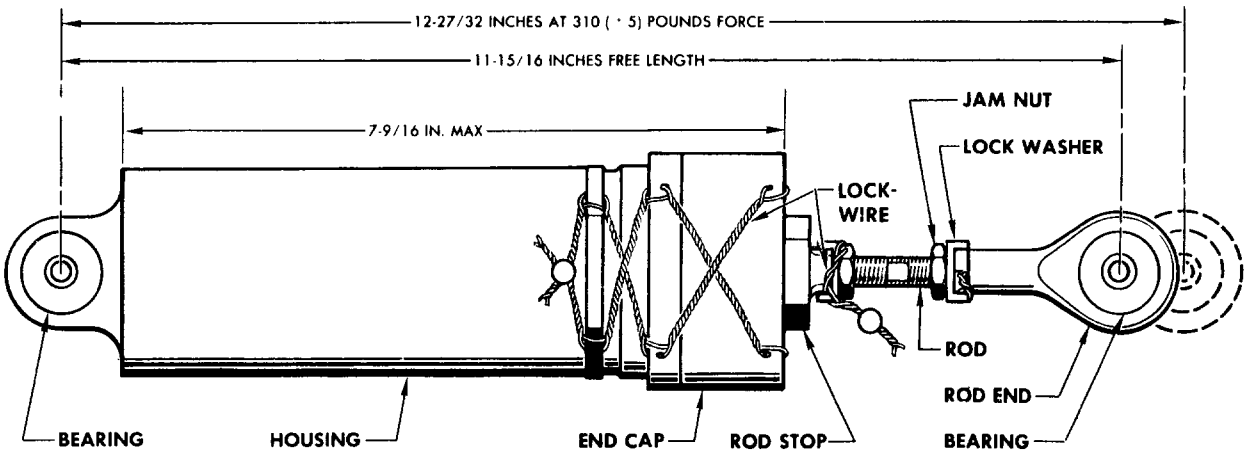
Checking Elevator Control Bobweight Balance Bungee
Figure 27-30-601

NOTE With the exception of the procedure in step 4, the following information is not to be used when the bias bungee is being adjusted. It is intended to furnish check-out information only, to determine whether the bungee is serviceable. The bungee is to be checked in the condition it is in when it is removed from the airplane. Lockwire on bungee body must not be broken. Internal adjustments of bungee must not be attempted. If the bungee fails to meet any of the checks, it should be replaced with a new unit.



To check bungee, proceed as follows:

1. Visually check bungee assembly to determine that housing, rod, bearings, etc., are not damaged.
2. Check to determine whether bungee is free of dirt, grease, oil, and foreign matter. If it is not, clean it.
3. Shake bungee to determine that loose objects are not inside bungee housing.
4. Check that free length of bungee is 11-15/16 inches. If it is not, adjust it to that measurement.
5. Ten times, extend length of bungee to 12-27/32 inches.
6. Check bungee force. The force required to extend the bungee to 12-27/32 inches should be 310 (± 5) pounds.
7. Measure distance from top of end cap to bottom of outer rim of housing; distance should not exceed 7-9/16 inches. If end cap requires adjustment, remove safety wire and adjust cap to obtain dimension, and safety end cap with MS20995F41 lockwire.



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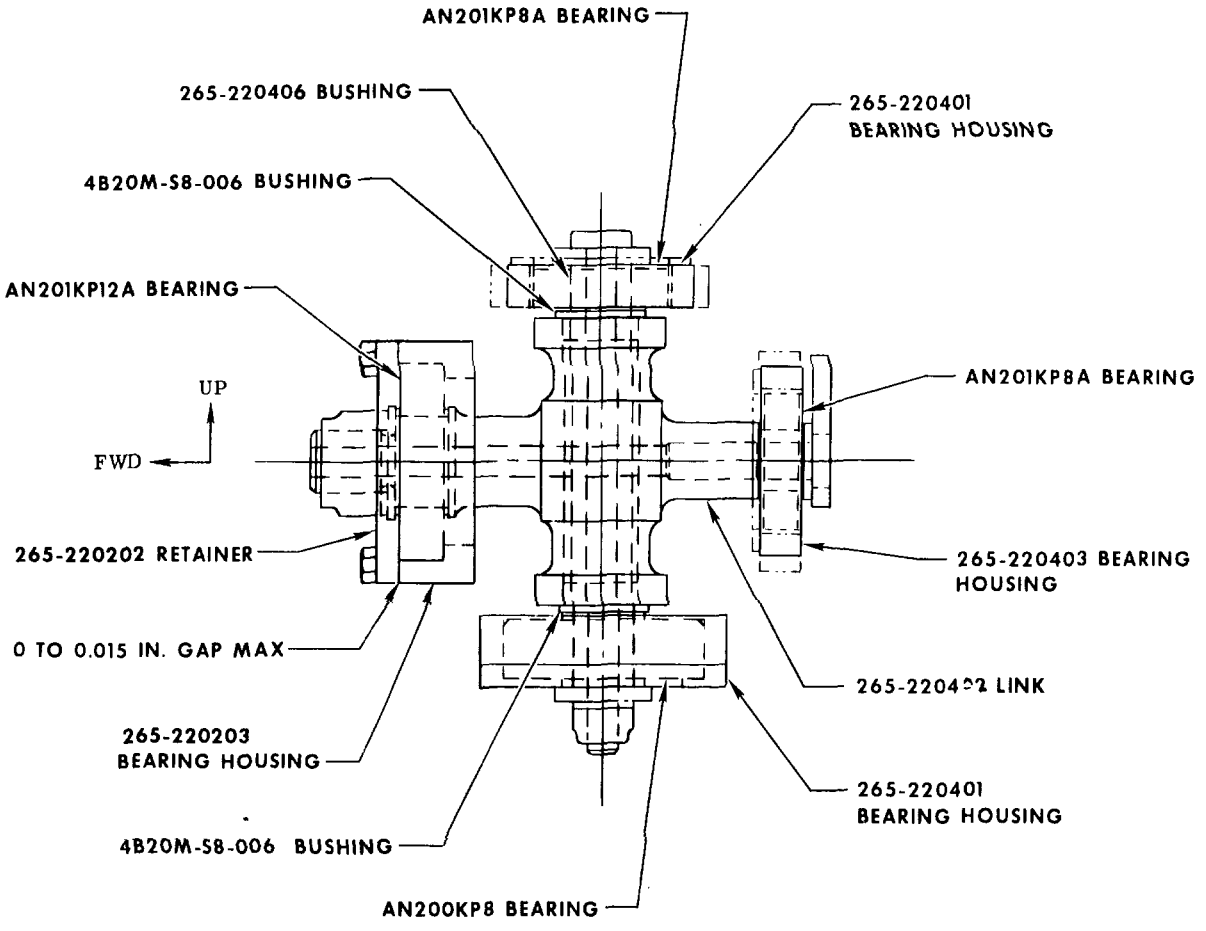
Checking Elevator Control Bias Bungee
Figure 27-30-602



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Elevator Universal Joint Component Identification
Figure 27-30-603

Table 601. Universal Joint Dimensions and Tolerances

		A	B	C	D	E	F	G
JOINT (SEE FIGURE 27-30-603)	JOINT COMPONENTS	MFG OR NEW DIM.	GAP BETWEEN COMPONENTS DUE TO COL A	JOINT GAP DUE TO COL B	MFG OR NEW BEARING RADIAL PLAY	NEW OR MFG TOTAL JOINT GAP (C + D)	MAX ALLOWABLE GAP BETWEEN JOINT COMPONENTS DUE TO WEAR (COL B MAX + WEAR)	ALLOWABLE JOINT GAP DUE TO WEAR (COL E MAX + WEAR)
FWD	265-220402 shaft OD	0.7475 nom 0.7465 min	0.0020 nom 0.0035 max	0.0020 nom 0.0048 max	0.000 nom 0.001 max	0.0020 nom 0.0058 max	0.0100 max	0.0108 max
	AN201KP12A bearing ID	0.7495 nom 0.7500 max						
	AN201KP12A bearing OD	1.6250 nom 1.6245 min	0.0000 nom 0.0013 max				0.0080 max	
	265-220403 bearing housing ID	1.6250 nom 1.6258 max						
AFT	265-220402 shaft OD	0.4975 nom 0.4965 min	0.0020 nom 0.0035 max	0.0020 nom 0.0046 max	0.000 nom 0.001 max	0.0020 nom 0.0056 max	0.0100 max	0.0106 max
	AN201KP8A bearing ID	0.4995 nom 0.5000 max						
	AN201KP8A bearing OD	1.1250 nom 1.1245 min	0.0000 nom 0.0011 max				0.0080 max	
	265-220403 bearing housing ID	1.1250 nom 1.1256 max						
UPPER	265-220406 shaft OD	0.4985 nom 0.4979 min	0.0010 nom 0.0021 max	0.0020 nom 0.0058 max	0.000 nom 0.001 max	0.0020 nom 0.0068 max	0.0090 max	0.0118 max
	AN201KP8A bearing ID	0.4995 nom 0.5000 max						
	AN201KP8A bearing OD	1.1250 nom 1.1245 min	0.0000 nom 0.0011 max				0.0080 max	
	265-220401 bearing housing ID	1.1250 nom 1.1256 max						
	265-220406 shaft OD	0.4985 nom 0.4979 min	0.0010 nom 0.0026 max				0.0080 max	
	4B20M-S8-006 bushing ID	0.4995 nom 0.5005 max						
LOWER	265-220406 shaft OD	0.4985 nom 0.4979 min	0.0010 nom 0.0021 max	0.0020 nom 0.0060 max	0.000 nom 0.001 max	0.0020 nom 0.0070 max	0.0090 max	0.0120 max
	AN200KP8 bearing ID	0.4995 nom 0.5000 max						
	AN200KP8 bearing OD	1.6875 nom 1.6870 min	0.0000 nom 0.0013 max				0.0080 max	
	265-220401 bearing housing ID	1.6875 nom 1.6883 min						
	265-220406 shaft OD	0.4985 nom 0.4979 min	0.0010 nom 0.0026 max				0.0080 max	
	4B20M-S8-006 bushing ID	0.4995 nom 0.5005 max						

NOTES

- *See figure 27-30-603 for identification of components.*
- *The following notes explain the letter columns in the chart:*
  - *Column A indicates allowable minimum and maximum manufactured dimensions.*
  - *Column B indicates allowable play between adjacent components in the joint, resulting from manufactured limits in column A.*
  - *Column C indicates allowable free play for the entire joint assembly, resulting from manufactured limits in column A.*
  - *Column D indicates allowable manufactured radial play in the bearings.*
  - *Column E indicates allowable play for the entire joint assembly with bearing radial play resulting from manufactured allowable limits (column C + column D).*
  - *Column F indicates maximum allowable play between adjacent components in the joint assembly, resulting from wear.*
  - *Column G indicates maximum allowable play between all components in the joint, resulting from wear. The dimensions given in this column are not the total of the dimensions given in column F. If maximum limits in column F for adjacent components are exceeded, then the maximum limits in column G will be exceeded.*
- *Total wear in all joints is not to exceed 0.010 in., even though wear in adjacent components within each individual joint may be within tolerance.*

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INSPECT — ELEVATOR HONEYCOMB TRAILING EDGE — S/B 48, AD 80-08-07

EQUIPMENT/TOOLS

- Metal Object, Coin or Washer

CONSUMABLES

- None

INSPECTION

CAUTION

Do not use a sharp metal object or tap too hard during this inspection or the honeycomb structure may be damaged.

NOTES

- ***Inspect the top and bottom surfaces of the trailing edge on each elevator for delamination.***
- ***Delamination may also be detected by looking across the trailing edge for wrinkles or buckles in the otherwise smooth surface.***

1. Lightly tap the upper and lower surfaces of the elevator trailing edge.
2. Listen for a sharp metallic click indicating sound honeycomb structure. A noticeably non-metallic click indicates delamination has occurred.

NOTE

Installation of non-honeycomb trailing edges will terminate additional inspection requirements.

3. Replace defective trailing edge(s). (Refer to S/B 48.)
4. Establish a continuing inspection cycle if trailing edges are sound. (Refer to chapter 5.)

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HORIZONTAL STABILIZER CONTROL - DESCRIPTION AND OPERATION

1. Horizontal Stabilizer

The entire horizontal stabilizer assembly is movable and is used for longitudinal trim. It is hinged to the upper part of the fuselage rear section at two places and is operated by two synchronized electromechanical actuators. The horizontal stabilizer assembly consists of two panels interconnected by a center section beam and an interconnect torque tube. The basic structure consists of a rear beam, stringers, and conventional ribs covered by stressed aluminum alloy skin.

The panels are bolted onto the center section beam at their inboard aft ends. The center section beam is attached to the fuselage at two pivot points. The inboard forward ends of the panels are attached to the bell-crank type ends of the interconnect torque tube. The torque tube is mounted to the fuselage by two idler links, one on each end of the torque tube. The two actuators are also attached to the inboard forward ends of the panels. The torque-tube/bell-crank arrangement provides a fail-safe horizontal stabilizer actuator system. In case one actuator should fail, the other will continue to operate at a reduced speed.

2. Horizontal Stabilizer Trim System

Longitudinal trimming of the airplane is controlled by pushing a trim switch, on the outboard grip of either control wheel, in an up or down direction. (See figure 27-40-1.) The trim switch controls two electromechanical actuators interconnected by a flexible shaft to provide single-actuator operation. The actuators mechanically position the entire horizontal stabilizer up or down. The actuators operate the stabilizer at two speeds: low speed when the landing gear is up, and high speed when the main landing gear is down and locked. The stabilizer and elevators are interconnected through the bias bungee. Movement of the stabilizer repositions the elevators in a no-load position through the bias bungee. A horizontal stabilizer trim position transmitter is mechanically operated by a rod connected to the left end of the stabilizer torque tube. The transmitter electrically operates a horizontal stabilizer trim position indicator in the cockpit. On NA-380-38 and later airplanes, an audible warning horn will sound whenever the horizontal stabilizer trim actuators are energized.

3. Horizontal Stabilizer Trim Position Indicator

The horizontal stabilizer trim position indicator, on the copilot's inboard instrument panel, is powered by the 28-volt dc essential bus. A rotating pointer indicates nose-up trim and is electrically positioned by a horizontal stabilizer trim position transmitter. The indicator is calibrated in increments of 1/2 degree.

4. Horizontal Stabilizer Trim Actuators

Two linear-type, 28-volt dc, electromechanical actuators operate the horizontal stabilizer. The actuators are connected to support fittings on the inboard forward end of each stabilizer panel. The actuators are interconnected by a flexible shaft to provide single-actuator operation. Each

actuator can drive the opposite actuator through the synchronizing flexible shaft. Therefore, in case of power failure to one actuator, stabilizer positioning (at a reduced rate) is still possible.

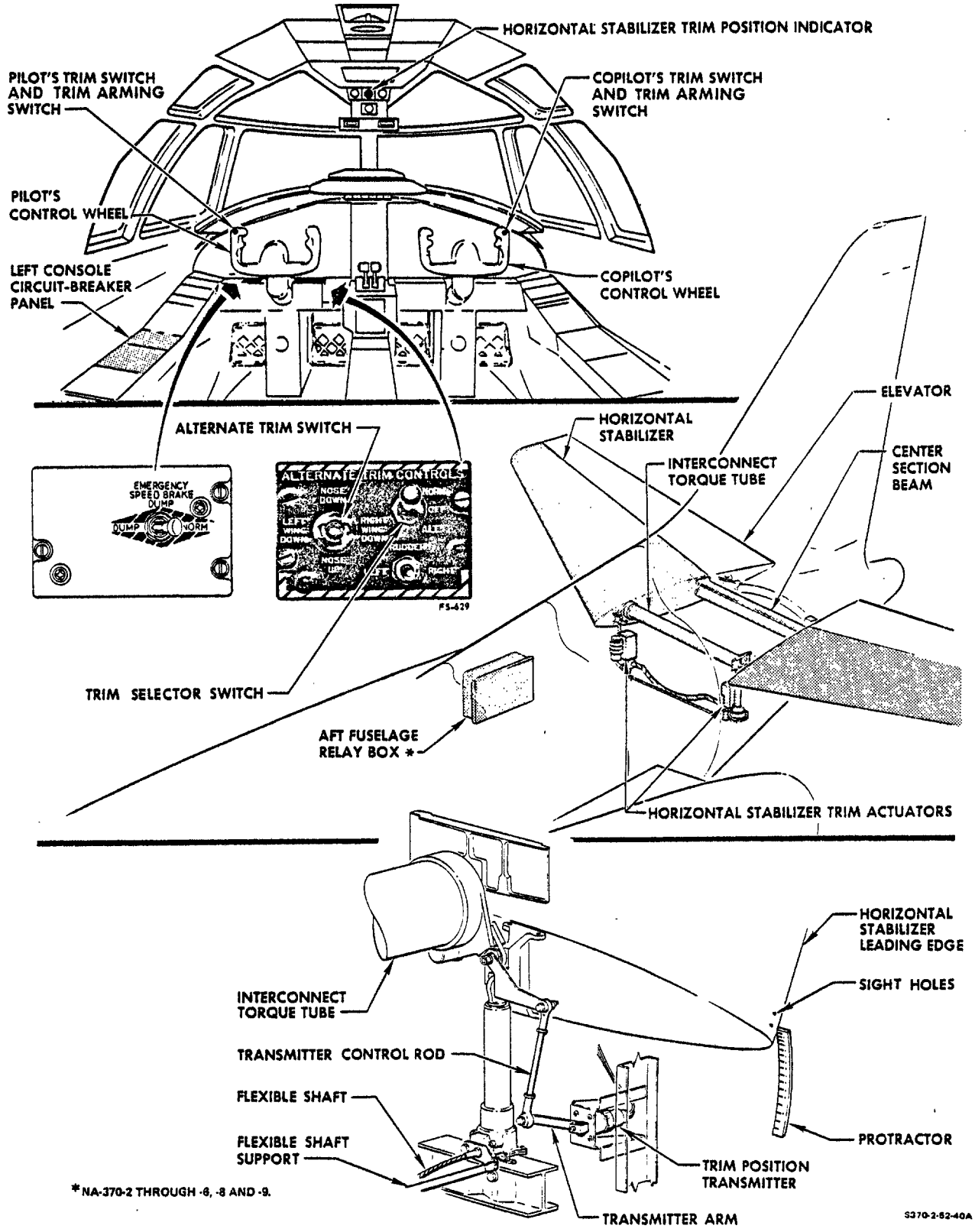
The actuators operate at high or low speed depending on landing gear position. The actuator operates normally at low speed through the extend field coil or retract field coil. The actuator will operate at high speed when the high-speed selector relay within the actuator is energized. The relay closes contacts that complete the circuit through a resistor to the opposite field coil. The resistor reduces the voltage to the opposite field coil. When both field coils are energized, with one at a reduced voltage, the electromagnetic field will be weakened. A weaker magnetic field increases the speed of the armature, thereby operating the actuator jackscrew at a higher speed.

The actuators have three limit switches to limit the travel of the jackscrew. Nonjamming mechanical stops are employed to restrict jackscrew travel in case of travel limit switch failure. An antirotational device is used to prevent the jackscrew end fitting from rotating.

5. Horizontal Stabilizer Trim Position Transmitter

The horizontal stabilizer trim position transmitter electrically operates the horizontal stabilizer trim position indicator. The transmitter is forward of the left horizontal stabilizer trim actuator. The transmitter is mechanically connected by a rod to the left end of the stabilizer interconnect torque tube. Movement of the torque tube by the horizontal stabilizer actuators operates the transmitter. The transmitter electrically operates the indicator in the cockpit. The transmitter is a potentiometer type designed to operate through 360 degrees or more of shaft rotation.

The transmitter wiper is centered on tap "B" with the stabilizer in 4-1/2 degree rig position. Retracting the actuators, moving the stabilizer leading edge down, produces A,B,C direction of rotation of the wiper inside the transmitter; movement of the wiper electrically reproduces a similar movement in the indicator.



*NA-370-2 THROUGH -6, -8 AND -9.

Horizontal Stabilizer Trim System
Figure 27-40-1

HORIZONTAL STABILIZER CONTROL

TROUBLE SHOOTING

TROUBLE SHOOTING HORIZONTAL STABILIZER TRIM SYSTEM

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Trim System Inoperative		
1. Circuit breakers not pushed in.	Check the four NORMAL TRIM circuit breakers, the three ALTERNATE TRIM circuit breakers, and TRIM IND and STAB SPEED CONT circuit breakers on left console circuit-breaker panel.	
2. Trim selector switch positioned incorrectly.	Check trim selector switch position on alternate trim control panel.	
3. Electrical system faulty.	Perform continuity check.	Repair electrical system.
4. Mechanical failure in one actuator, causing other actuator to be held immobile through synchronizing shaft.	Disconnect one actuator from torque tube. Pull one circuit breaker; operate opposite actuator. Reverse procedure for other side.	Replace defective actuator. Repair or replace faulty wiring.
5. Faulty trim switch or electrical wiring.	Position TRIM SELECTOR switch at NORM and hold trim and arm switch at NOSE UP. Check for 28 VDC at terminal X1 on K540 relay at right hand station 150. (Relay panel 107) and 28 VDC at terminal (X2) or 15 on K534 relay at right hand station 150 (relay assembly 125). Hold trim switch at NOSE DN. Check for 28 VDC at terminal X1 or K538 relay at right hand station 150. If voltage is not available at either contact, switch is faulty or electrical wiring. ***NOTE*** ***Terminal numbers not in parenthesis are for S/N 370-1 and -7, and 380-1 and Subs. Terminal numbers with parenthesis are for S/N 370-2 thru -6, -8, and -9.***	Replace trim switch or repair electrical wiring.
Trim Actuators Do Not Disable When Trim Disc Switch is Depressed — Airplanes Changed by S/B 74-6		
1. Faulty trim disconnect switch(es) or open electrical wiring.	Perform continuity check of switch(es) and electrical wiring	Replace switch(es) or repair electrical wiring.
2. Faulty aileron and rudder trim disconnect relay (K543).	Perform continuity check of relay coil.	Replace relay coil.

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TROUBLE SHOOTING (Continued)

TROUBLE SHOOTING HORIZONTAL STABILIZER TRIM SYSTEM (Cont'd)

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Horizontal Stabilizer Trim Actuator(s) Do Not Disable When Trim Disc Switch is Depressed — Airplanes Changed by S/B 74-6.		
1. Faulty RH and/or LH stabilizer trim disconnect relay(s) K541 and/or K542.	With electrical power applied to airplane, check for 28 VDC at terminals "XI" of relay coil(s). If voltage is not present, check for open circuit in electrical wiring. If voltage is present, replace relay(s).	Replace relay(s) or repair electrical wiring.
Audible Signal Absent When Trimming Horizontal Stabilizer — Airplanes Changed by S/B 74-7		
1. Faulty audible warning horn (LS505).	Check for 28 VDC at positive terminal of horn when stabilizer is trimming.	Replace warning horn.

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HORIZONTAL STABILIZER CONTROL - MAINTENANCE PRACTICES

1. Removing Horizontal Stabilizer Panel

A. The removal procedures for the left and right horizontal stabilizer panels are the same.

(1) Remove elevator from stabilizer if stabilizer panel is being replaced; otherwise, elevator does not need to be removed.

(2) Disconnect actuator rod end from forward support fitting.

(3) Through accesses, disconnect stabilizer forward support fitting from interconnect torque tube. When removing both surfaces, disconnect transmitter rod from torque tube before disconnecting support fitting; then carefully lower torque tube.

(4) Remove one bolt attaching elevator inboard hinge fitting to elevator torque tube assembly.

(5) Remove the two bolts attaching panel to center section beam, and remove horizontal stabilizer panel.

2. Installing Horizontal Stabilizer

See figure 27-40-401.

3. Installing and Adjusting Horizontal Stabilizer Center Section Beam Attach Bolts

See figure 27-40-402.

NOTE: If the horizontal stabilizer center section beam support fittings or the related splice fittings, on the forward and aft sides of the canted bulkhead at fuselage station 457, are replaced, it is necessary to inspect for excessive clearance as shown in figure 27-40-402, detail A. If clearance is excessive, add shims. If the support or splice fittings are removed and are to be reinstalled, any existing shims must be retained for reinstallation.

4. Horizontal Stabilizer Wear and Rebushing Data

Refer to Structural Repair Manual, Report No. NA-66-1032.

5. Removing Horizontal Stabilizer Trim Actuators

A. The horizontal stabilizer actuators are in the fuselage rear section, under the leading edge of the stabilizer. The removal procedures for the left and right actuators are the same.

(1) Disconnect actuator electrical plug.

(2) Disconnect interconnect flex shaft from actuator.

(3) Disconnect flex shaft support from actuator.

(4) Remove attaching bolts from both ends of actuator and remove actuator.

6. Installing Horizontal Stabilizer Trim Actuators

See figure 27-40-403.

7. Removing Horizontal Stabilizer Trim Position Transmitter

A. The horizontal stabilizer trim position transmitter is in the fuselage rear section, aft of the left horizontal trim actuator.

(1) Disconnect electrical plug through accesses.

(2) Disconnect arm from shaft.

(3) Remove the four screws from bracket and remove transmitter.

8. Installing Horizontal Stabilizer Trim Position Transmitter

See figure 27-40-404.

9. Rigging Horizontal Stabilizer Trim System

See figure 27-40-405.

10. Replacement of Horizontal Stabilizer Rotary Flex Drive

A. To remove horizontal stabilizer rotary flex drive, proceed as follows:

(1) Open aft fuselage access doors on aft fuselage lower centerline.

(2) Ensure that all electrical power is off.

(3) Disconnect drive coupling from each trim actuator.

(4) Remove clamps securing flex drive to support rod and remove flex drive.

B. To install horizontal stabilizer rotary flex drive, proceed as follows:

(1) Check the total free end play of the flex shaft is 1/32 to 3/32 inch (see figure 27-40-406). To check free play, proceed as follows:

(a) Lay the flex drive assembly on a flat surface. Push the inner shaft one way as far as possible.

(b) Measure distance between washer and splined end as depicted.

CAUTION: AFTER THE REQUIRED DIMENSION IS OBTAINED, DO NOT ALLOW THE OUTER CASING TO ROTATE DURING HANDLING OR INSTALLATION.

(c) If the dimension is not 1/32 to 3/32 inch, secure one end of outer casing and rotate opposite end as required until the required dimension is obtained.

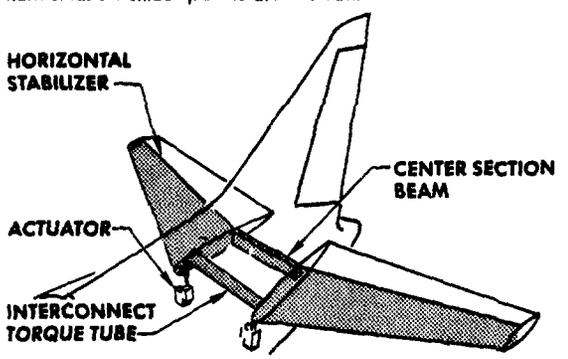
(2) Connect flex drive couplings to trim actuators. Ensure that drive is engaged properly. Hold outer casing and tighten "B" nuts. Safety with lockwire.

(3) Route flex drive as depicted in figure 27-40-405; then, install clamps securing flex drive to support rod.

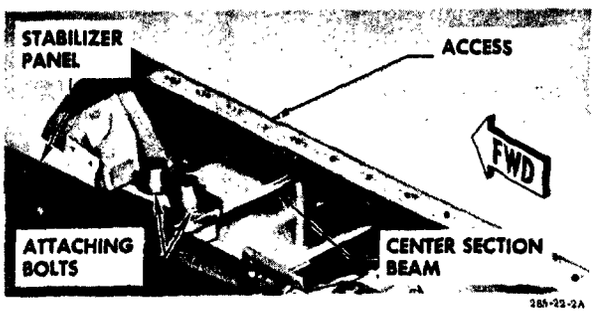
(4) Perform operational check out of stabilizer trim system.

(5) Close access doors.

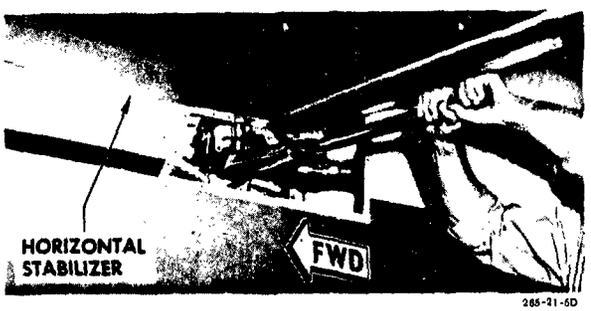
NOTE The installation procedures for the left and right horizontal stabilizer panels are the same.



1 With accesses removed, install two bolts attaching stabilizer panel to center section beam.

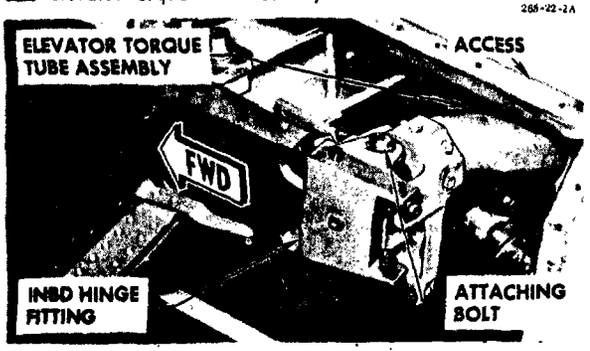


2 Install nuts and washers on bolts, and torque them to standard values of 2500 to 3000 inch-pounds.



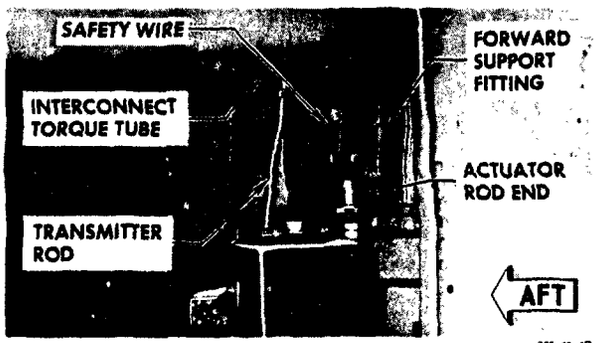
3 Install bolt attaching elevator inboard hinge fitting to elevator torque tube assembly.

265-22-2A



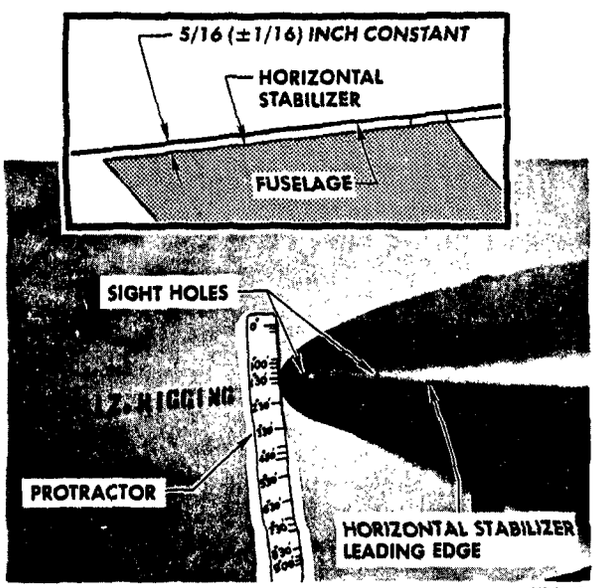
4 Through accesses, connect stabilizer forward support fitting to interconnect torque tube and safety with lockwire. Connect transmitter rod to torque tube if transmitter rod was disconnected during removal of stabilizer.

5 Connect actuator rod end to forward support fitting.



6 Install elevator if it was removed.

7 With horizontal stabilizer leading edge at 0 degree, check clearance between horizontal stabilizer and fuselage as shown. Trim as necessary.

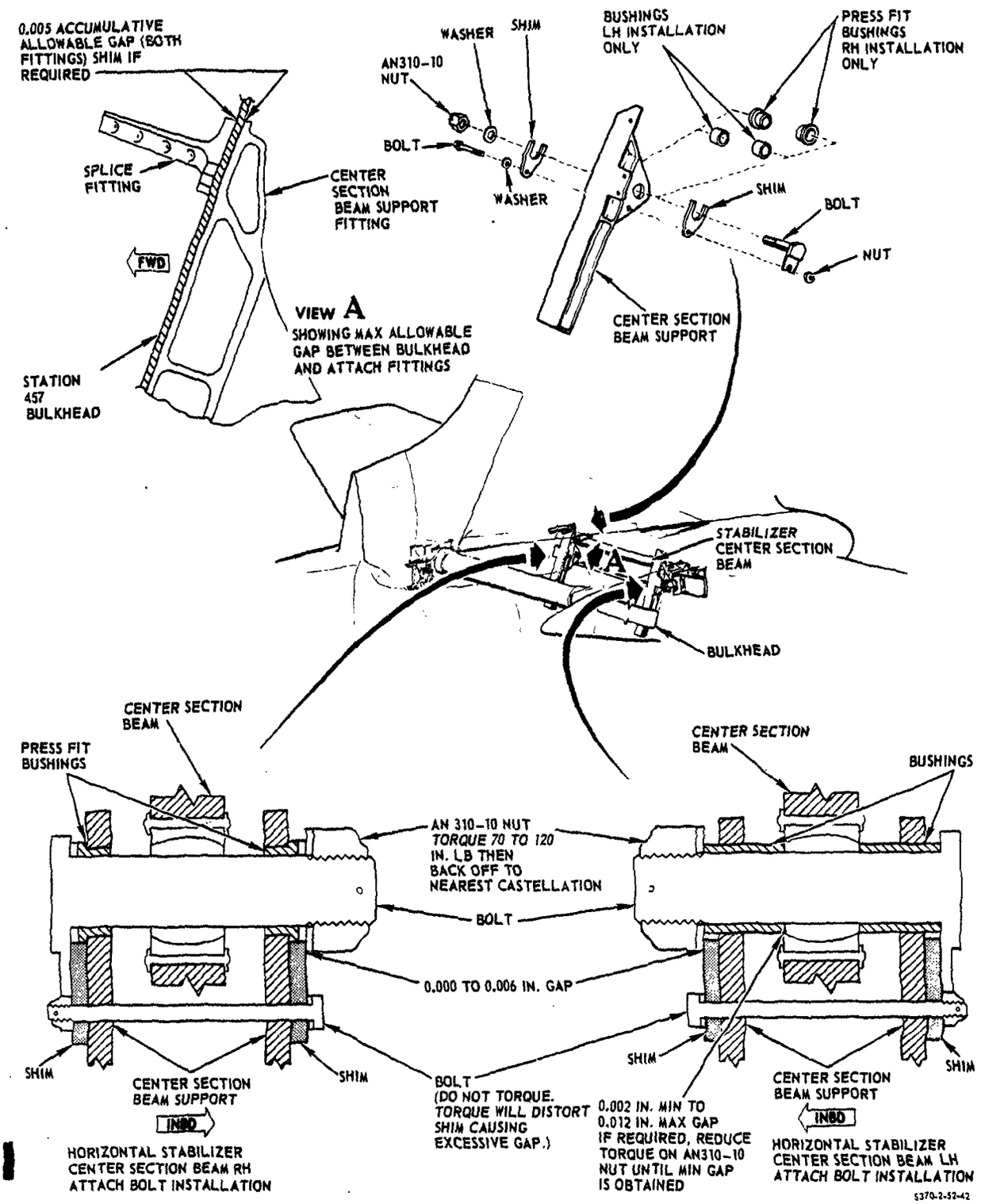


8 Slowly operate stabilizer through full range of travel. There should be no evidence of binding.

9 Inspect all connections. Make sure that area is free of tools, rags, and loose hardware.

10 Install accesses.

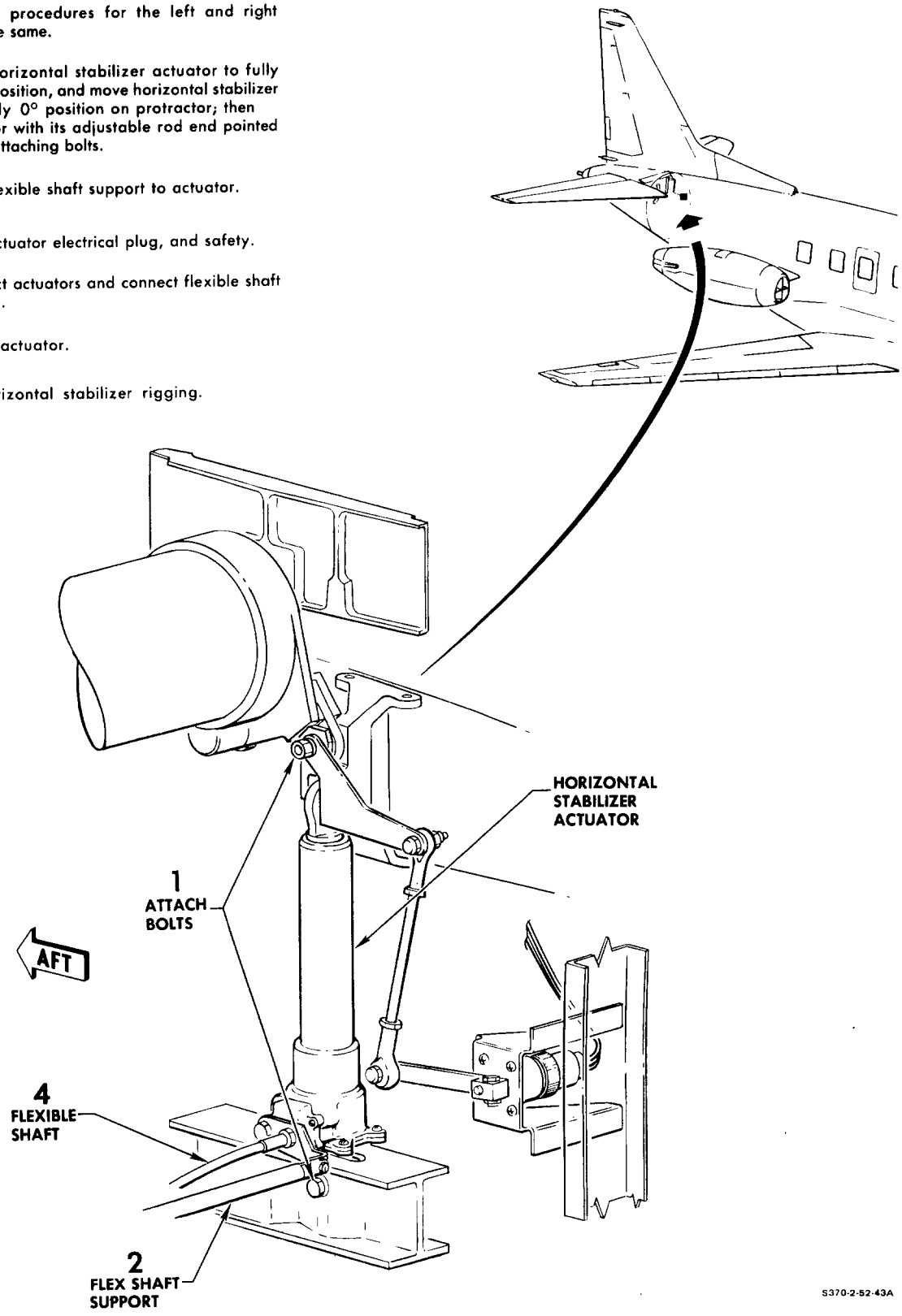
3370-2-82-41A



Horizontal Stabilizer Center Section Beam
Attach Bolt Installation
Figure 27-40-402

The installation procedures for the left and right actuators are the same.

1. Operate horizontal stabilizer actuator to fully extended position, and move horizontal stabilizer to approximately 0° position on protractor; then position actuator with its adjustable rod end pointed up, and install attaching bolts.
2. Connect flexible shaft support to actuator.
3. Connect actuator electrical plug, and safety.
4. Fully retract actuators and connect flexible shaft to actuator.
5. Lubricate actuator.
6. Check horizontal stabilizer rigging.



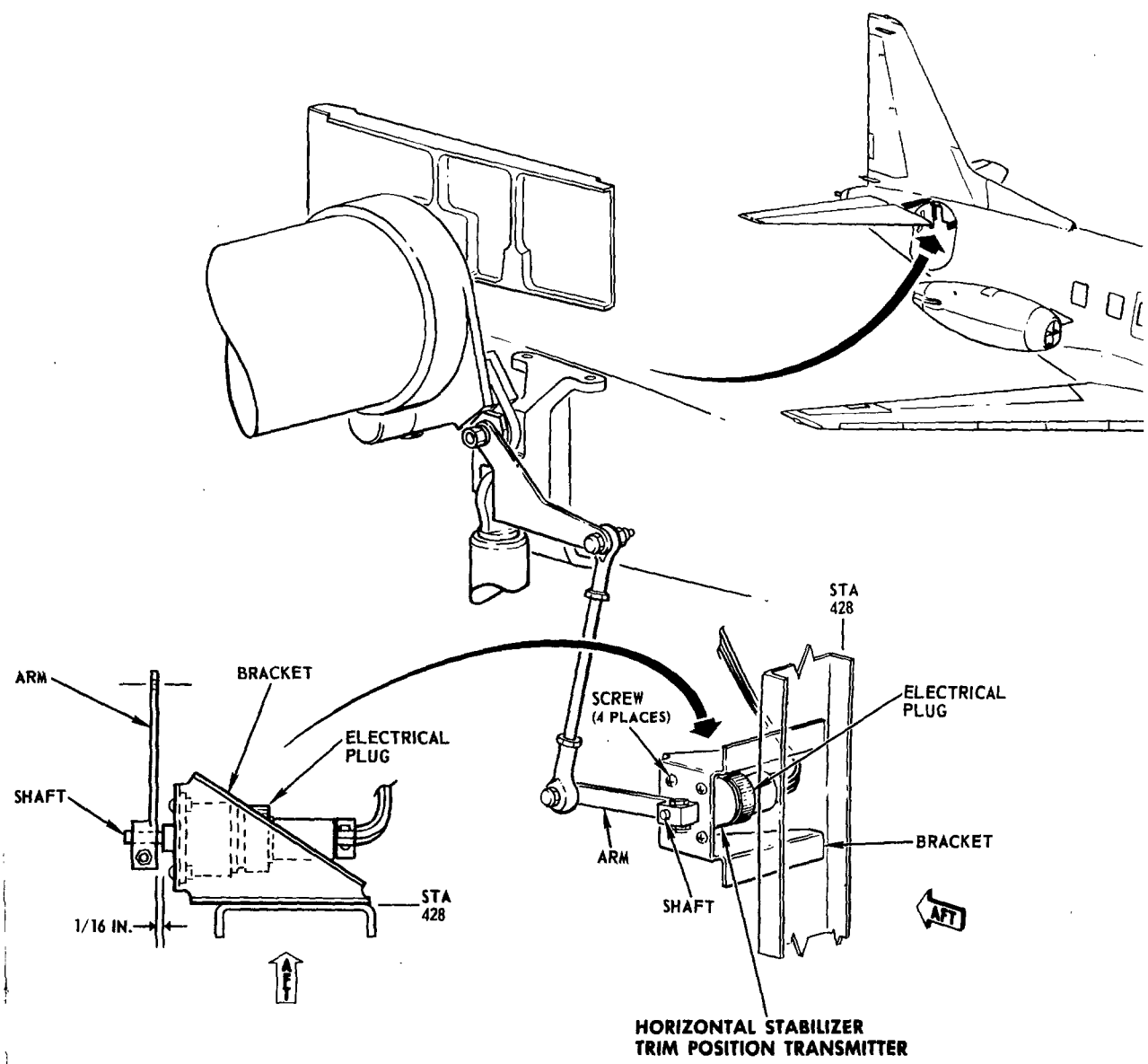
Installing Horizontal Stabilizer Trim Actuators
Figure 27-40-403

1 Through access doors 1 and 51, obtain access to bracket on left side of fuselage rear section. Using four screws, install horizontal side stabilizer trim position transmitter, its shaft inboard, on bracket.

2 Install arm on transmitter with flat side outboard, check that clearance between arm and transmitter is 1/16 inch; then connect arm to shaft

3 Connect electrical plug, and safety.

4 Connect transmitter. Do steps 2 through 5 of "Checking Horizontal Stabilizer Trim System."



Installing Horizontal Stabilizer Trim Position Transmitter
Figure 27-40-404

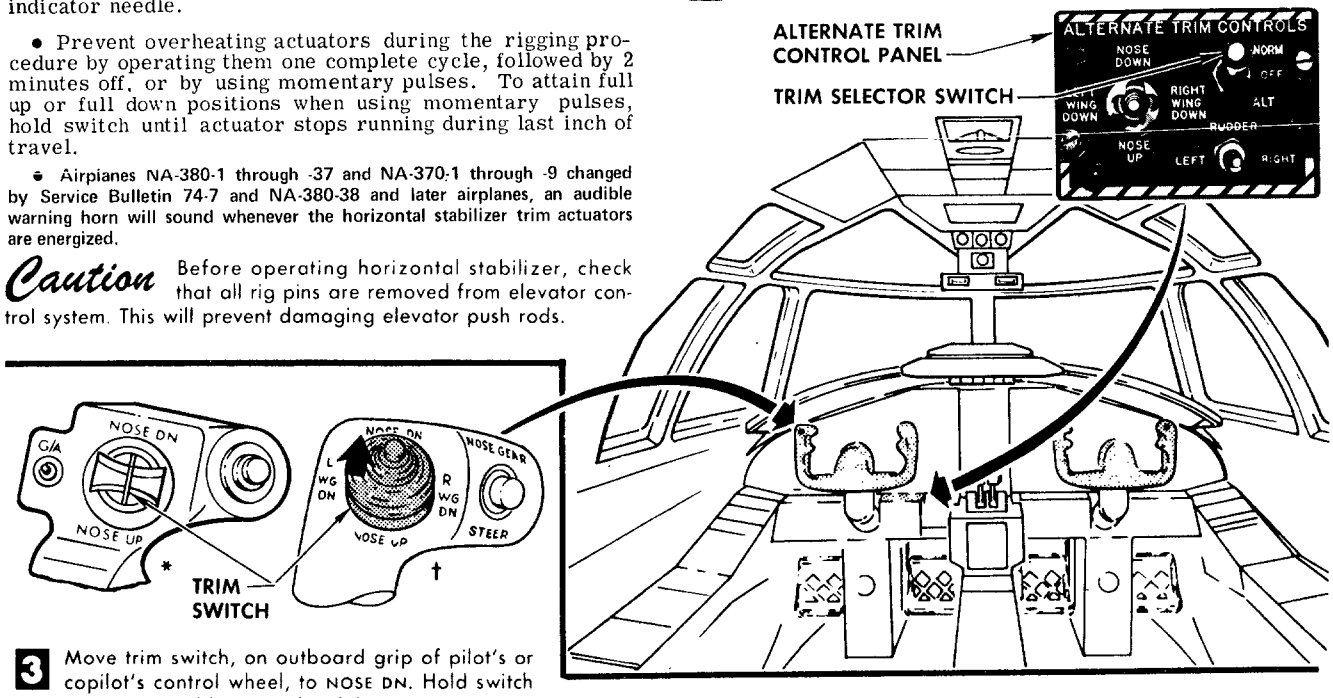
NOTE The object of this procedure is to synchronize, within tolerances, the travel of the leading edges of both panels of the horizontal stabilizer, one with the other, and with the travel of the horizontal stabilizer trim position indicator needle.

- Prevent overheating actuators during the rigging procedure by operating them one complete cycle, followed by 2 minutes off, or by using momentary pulses. To attain full up or full down positions when using momentary pulses, hold switch until actuator stops running during last inch of travel.
- Airplanes NA-380-1 through -37 and NA-370-1 through -9 changed by Service Bulletin 74-7 and NA-380-38 and later airplanes, an audible warning horn will sound whenever the horizontal stabilizer trim actuators are energized.

Caution Before operating horizontal stabilizer, check that all rig pins are removed from elevator control system. This will prevent damaging elevator push rods.

1 Connect 28-volt dc utility external power source to airplane.

2 Move trim selector switch on alternate trim control panel to NORMAL.

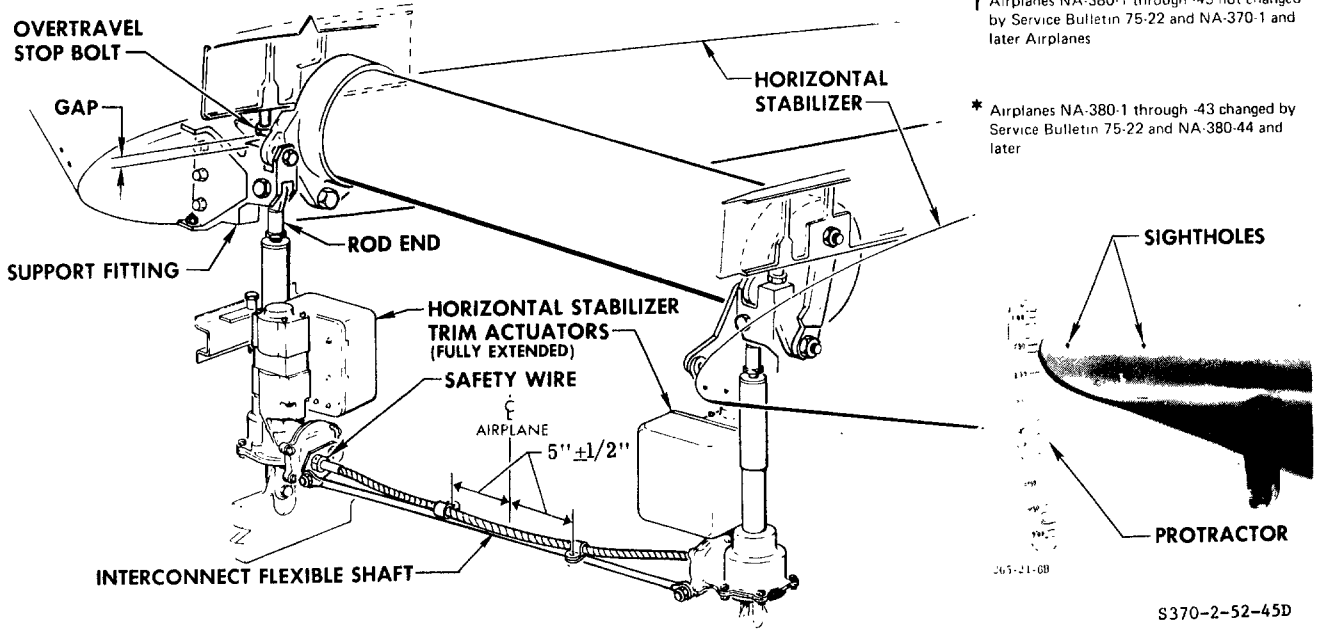


3 Move trim switch, on outboard grip of pilot's or copilot's control wheel, to NOSE DN. Hold switch at this position until horizontal stabilizer stops moving. Through access doors, check that overtravel stop bolts do not touch support fittings on horizontal stabilizer. Back off stop bolts if necessary.

4 Check that sightholes on leading edge or horizontal stabilizer align with a point 0 (+15, -0) degrees on protractors. Adjust rod ends on actuators as necessary.

Caution When the interconnect flexible shaft is connected to actuators, do not allow the outer casing to rotate while the "B" nuts are tightened.

5 Disconnect interconnect flexible shaft at one actuator to remove any possible cable twist, then reconnect it and lockwire.



Rigging Horizontal Stabilizer Trim System
Figure 27-40-405 (Sheet 1 of 4)

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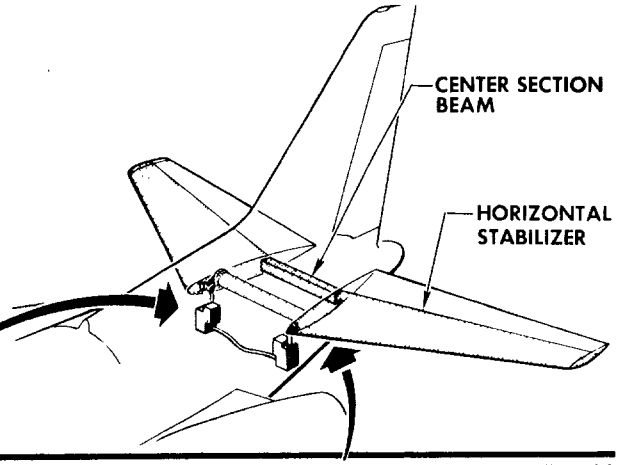
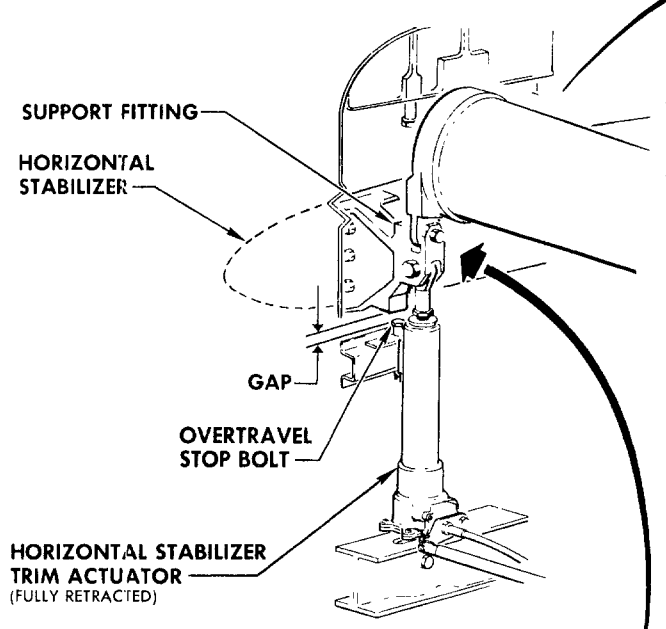
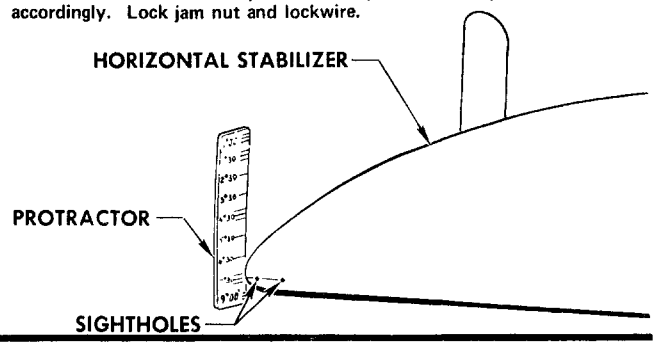
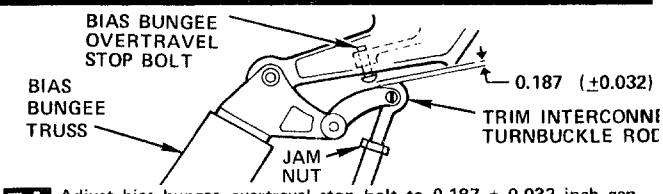
6 With main landing gear down and locked, move trim switch to NOSE UP. Hold switch at this position until horizontal stabilizer stops moving. Check that overtravel stop bolts do not touch support fittings on stabilizer. Back off stop bolts if necessary.



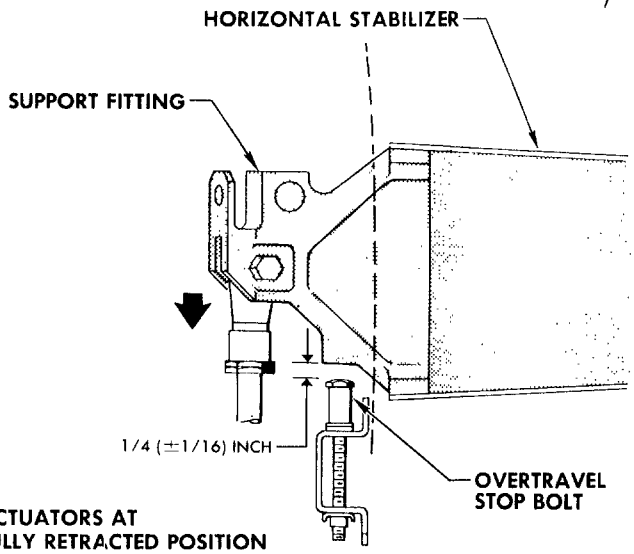


7 Check that sightholes on leading edge of horizontal stabilizer align with a point 8 (±15) degrees on protractor. Adjust rod and on left actuator to divide stroke between up and down positions. Adjust right actuator accordingly. Lock jam nut and lockwire.



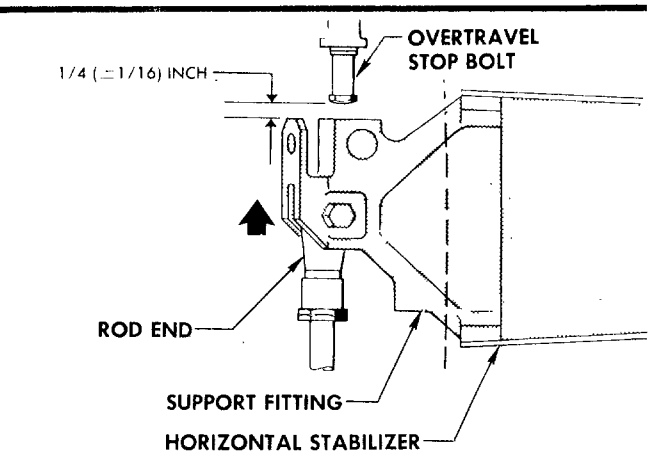


7A Adjust bias bungee overtravel stop bolt to 0.187 ± 0.032 inch gap. Secure jam nut and lockwire.



ACTUATORS AT FULLY RETRACTED POSITION

8 With actuators at full retracted position check clearance of overtravel stop bolts. Clearance between overtravel stop bolts and support fittings should be 1/4 (±1/16) inch. Adjust spacers and stop bolts until this clearance is met.



ACTUATORS AT FULLY EXTENDED POSITION

9 Energize actuators to their full extended position. Clearance between overtravel stop bolts and support fittings should be 1/4 (±1/16) inch. Adjust spacers and stop bolts until this clearance is met.

S 380-2-52-46B

Rigging Horizontal Stabilizer Trim System
Figure 27-40-405 (Sheet 2 of 4)

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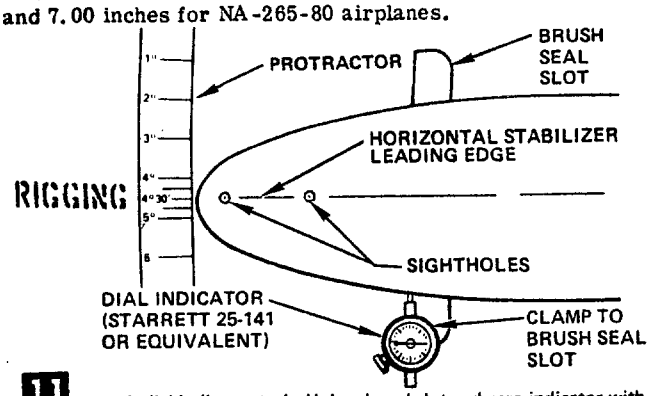
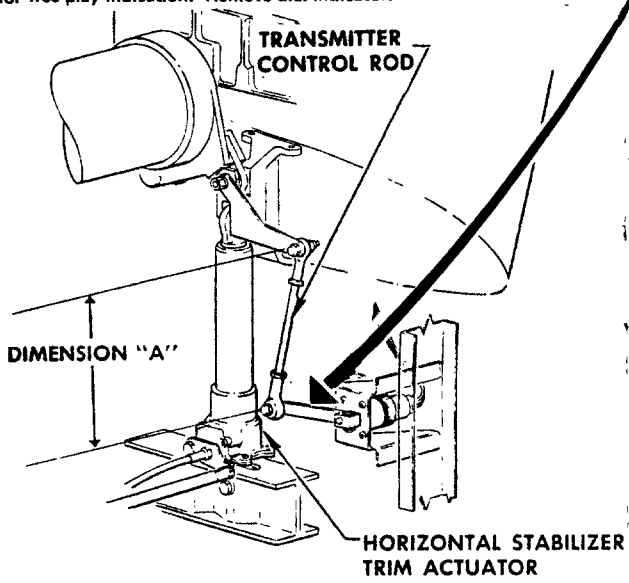
10 Operate trim switch to position stabilizer leading edge at 4-1/2 degrees (4° 30') down. Check that dimension "A" of transmitter control rod is 8.84 inches for NA-265-70 airplanes and 7.00 inches for NA-265-80 airplanes.

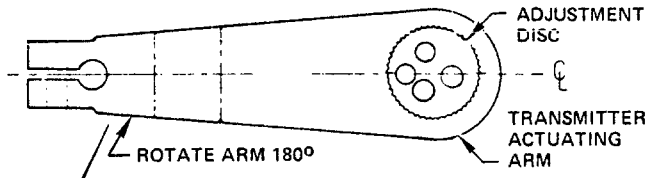


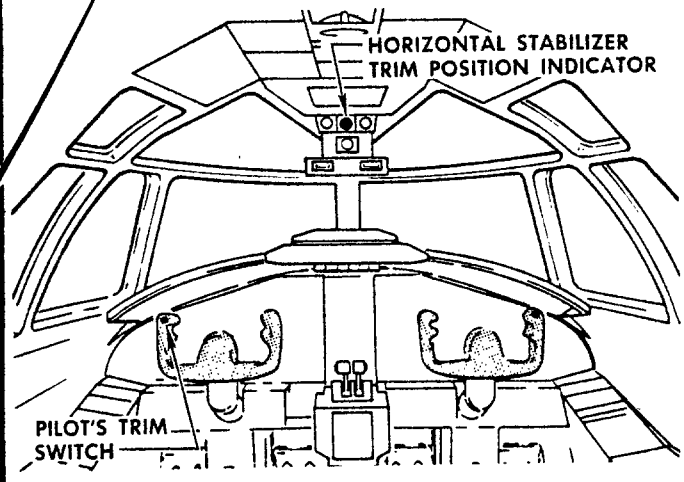
11 Attach dial indicator to L. H. brush seal slot and zero indicator with stabilizer leading edge at 4-1/2 degrees. Manually push up on the stabilizer leading edge and record indicator reading, then pull stabilizer leading edge down and record indicator reading. Add both readings and it must not be over 0.040 inch (This indicates the free-play of the horizontal stabilizer actuator and its attach points). Repeat this step on the R. H. side of fuselage for free-play indication. Remove dial indicator.



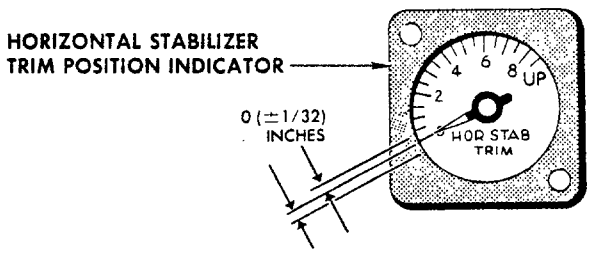
12 With trim indicator system energized and stabilizer leading edge at 4-1/2 (± 1/4) degrees (4°30'), check "HOR STAB. TRIM" position indicator in cockpit. The indicator should read 4-1/2 degrees NOSE-UP. To set indicator, adjust transmitter by rotating transmitter shaft. Install transmitter arm on shaft with flat side facing aft, and lightly clamp transmitter arm to shaft.

The transmitter actuating arm has an adjustment disc. Rotating the disc will provide an increased or decreased adjustment range.

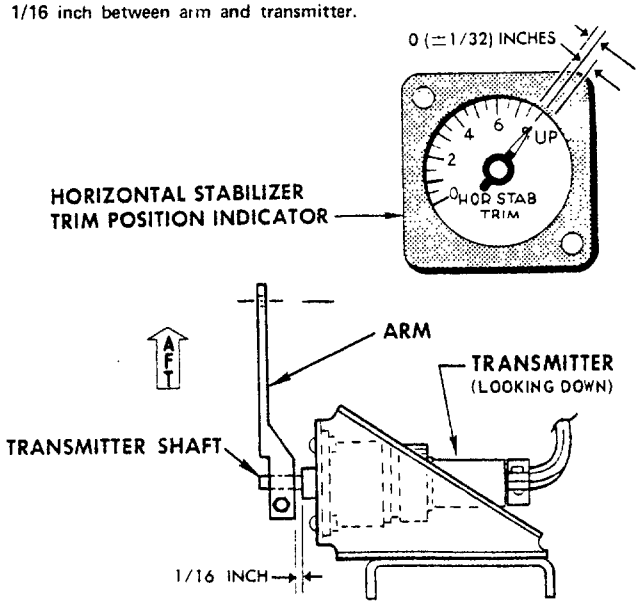




13 Hold trim switch at NOSE DN until stabilizer stops moving. The needle of the "HOR STAB. TRIM" position indicator should align, within ±1/32 inch, with the 0-degree index line. If indicator needle does not align within this tolerance, readjust transmitter shaft. Lightly clamp transmitter arm to shaft.



14 Move trim switch to NOSE UP. Hold trim switch in this position until stabilizer stops moving. The needle of the "HOR STAB. TRIM" position indicator should align, within ± 1/32 inch, with the 8-degree index line. If it does not, readjust transmitter shaft until it does. Clamp transmitter arm to shaft, allowing a clearance of 1/16 inch between arm and transmitter.



S380-2-52-47D

Rigging Horizontal Stabilizer Trim System
Figure 27-40-405 (Sheet 3 of 4)

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15 Remove top access covers over left and right horizontal stabilizer attach bolts.

16 Operate trim switch to position horizontal stabilizer leading edge at 4-1/2 degrees (4°30") down.

17 Fabricate special support angle using (Aluminum Alloy Extrusion) 1E30 7075–T6511 QQ–A–200/11 or equivalent as shown. Fabricate special clamp using 2024–T351 aluminum alloy rolled bar QQ–A–225/6 or equivalent 3/4 X 1 1/2 X 1 1/2.

Use Starrett Dial Indicator No. 25–441 or equivalent

18 Attach dial indicator to LH support at Fus. Sta. 457.406 as shown using clamp and left hand special support angle.

19 Zero indicator with horizontal stabilizer leading edge at 4–1/2 degrees (4°30") down.

20 Manually push up on the LH horizontal stabilizer outboard rib sufficiently to just begin to lift aft end of aircraft and record reading.

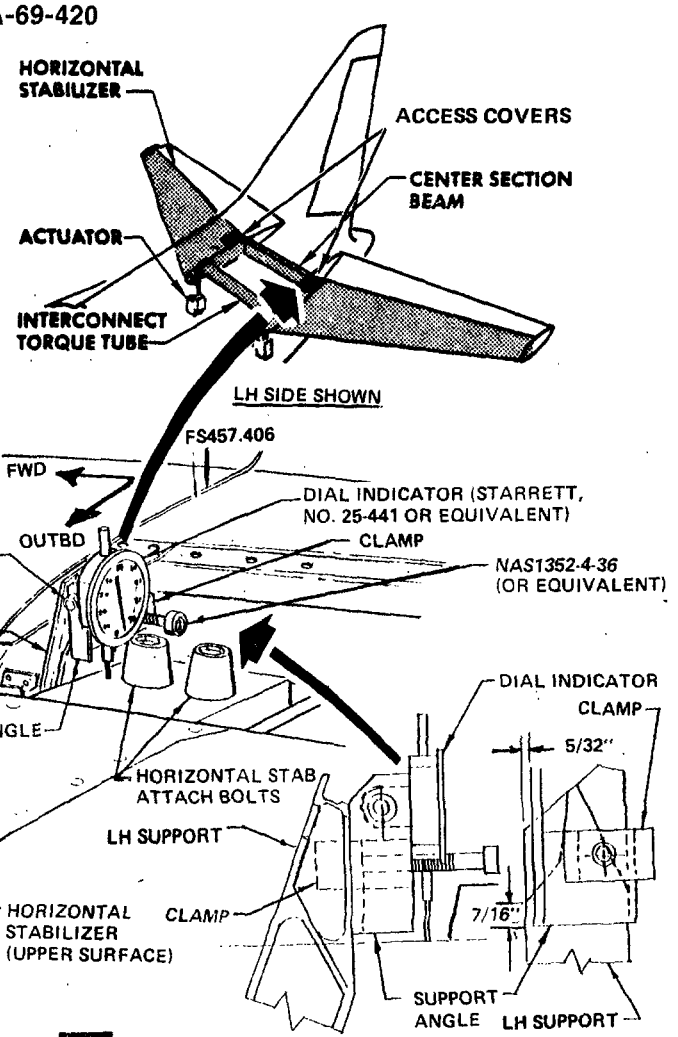
CAUTION: Do not push up on horizontal stabilizer tip. Push at outboard rib only.

21 Remove dial indicator, support angle and clamp from LH support.

22 Attach dial indicator to RH support at Fus. Sta. 457.406 as previously shown, using clamp and right hand special support angle.

23 Manually push on the RH horizontal stabilizer outboard rib sufficiently to just begin to lift aft end of aircraft and record reading.

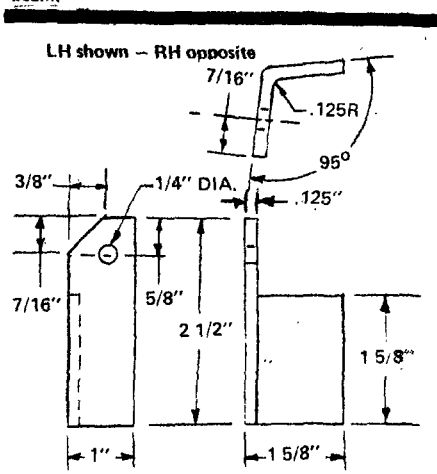
24 Readings obtained indicate play in attach joints between the LH support, the RH support, and the horizontal stabilizer center section beam.

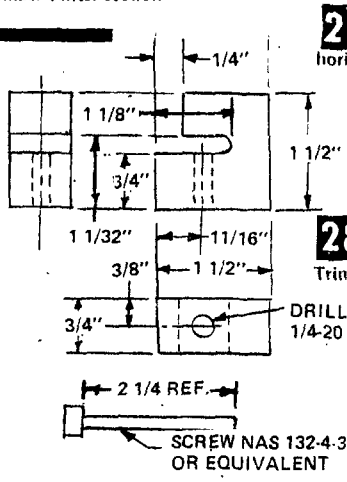


25 Readings must not exceed .0195 on the left side and .0165 on the right side.

26 If readings exceed noted diamensions, a detailed inspection of the joint components must be made to determine actual joint component wear. If component wear exceeds the maximum allowable wear, the joint must be replaced or repaired

27 Remove dial indicator, support angle and clamp from right horizontal stabilizer support and install removed access covers.





28 Comply with procedures in "Checking Horizontal Stabilizer Trim System".

Rigging Horizontal Stabilizer Trim System
Figure 27-40-405 (Sheet 4 of 4)

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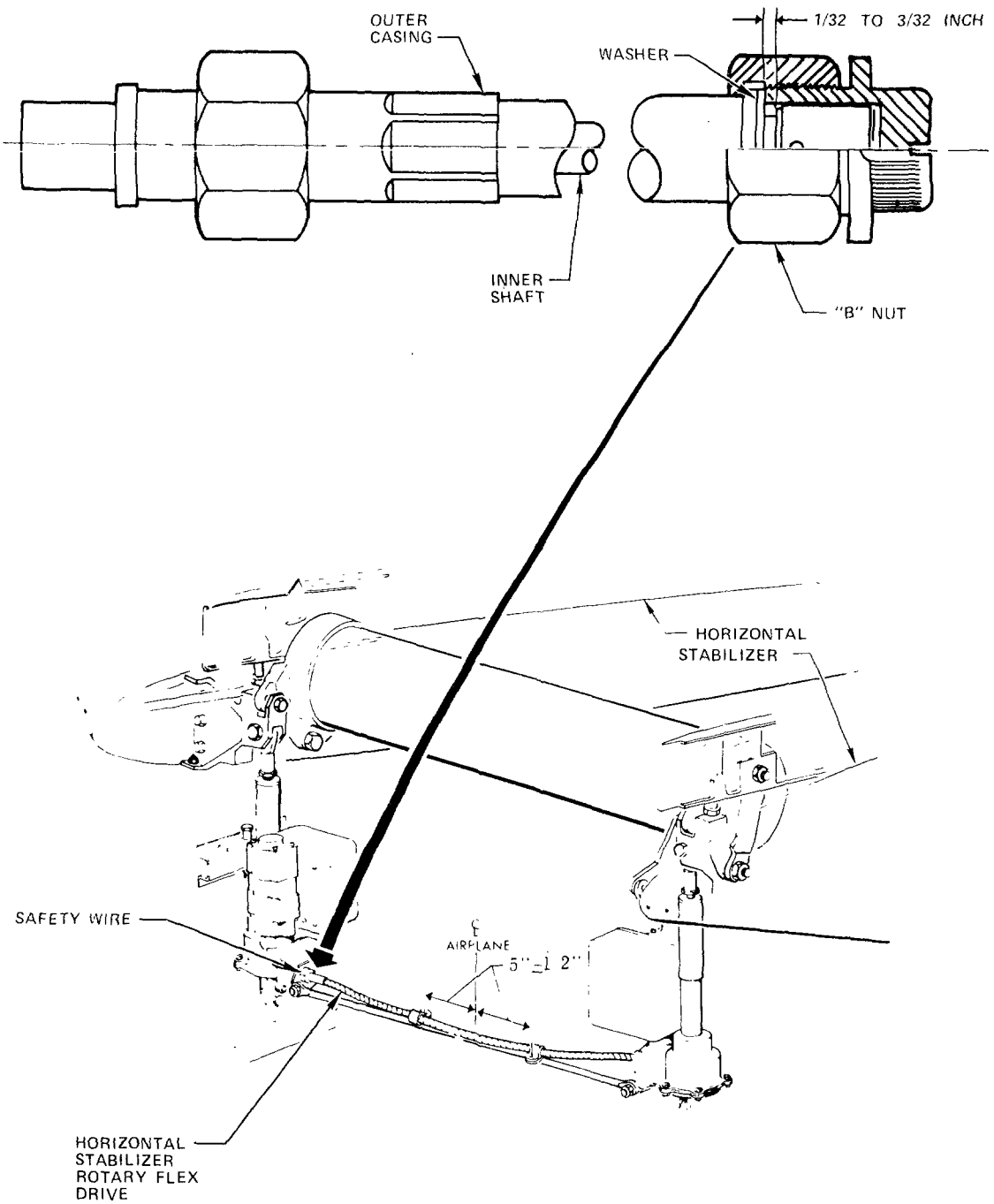

S 370-2-52-62

Horizontal Stabilizer Rotary
Flex Drive Free End Play
Figure 27-40-406

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INSPECTION/CHECK

CHECKING HORIZONTAL STABILIZER TRIM SYSTEM

Before checkout, the horizontal stabilizer trim system must be properly rigged. (See Figure 27-40-405.) If trim system does not meet checkout requirements, refer to trouble shooting procedures for the horizontal stabilizer trim system. With the airplane on jacks, proceed as follows:

1. Connect 28 VDC utility external power source to airplane.
2. Push in the following circuit breakers on left console circuit-breaker panel:
   - NORMAL TRIM (four)
   - ALTERNATE TRIM (three)
   - TRIM IND
   - STAB SPEED CONT
3. Place trim selector switch, on alternate trim control panel, at NORMAL.

CAUTION

Check that all rig pins are removed from elevator control system before operating horizontal stabilizer. This will prevent damage to elevator push rods.

4. Engage trim arming switch and hold switch at NOSE DN until horizontal stabilizer stops moving. The horizontal stabilizer leading edge should go up. The needle of the horizontal stabilizer trim position indicator should align within ±1/32 inch of the zero degree index line. Check that sightholes on the leading edge of horizontal stabilizer align between zero and -1/4 degree marks on protractors on both sides of fuselage.

NOTES

- ***On S/N 380-1 thru -37 and S/N 370-1 thru -9 changed by S/B 74-7 and S/N 380-38 and Subs, an audible warning horn will sound whenever the horizontal stabilizer trim actuators are energized.***
- ***To prevent overheating actuators during rigging procedure, actuators may be operated for one complete cycle followed by 2 minutes off, or by using momentary pulses. To attain full-up or full-down positions when using momentary pulses, hold switch until actuator stops running during last inch of travel.***

5. With main landing gear down and locked and trim arming switch engaged, move trim switch, on left grip of pilot's control wheel, to NOSE UP. Hold until horizontal stabilizer stops moving. The horizontal stabilizer edge should go down. The needle of the horizontal stabilizer trim position indicator should align within ±1/32 inch of the 8-degree index line. Check that sightholes on inboard edge of stabilizer align with 7-3/4 degrees minimum, 8-1/4 degrees maximum on protractor attached to fuselage.
6. Repeat steps 4 and 5, on co-pilot's control wheel.
7. Position trim selector switch, on ALTERNATE TRIM CONTROLS panel, to ALT.
8. Repeat steps 4 and 5 with alternate trim switch, on ALTERNATE TRIM CONTROLS panel.
9. Return trim selector switch to NORM.
10. Check high trim speed of horizontal stabilizer full trim range with main landing gear down and locked. Operate trim switch. Check one full length of trim travel (no less than 6 seconds minimum and no more than 7.5 seconds maximum).
11. Check low-speed operation of horizontal stabilizer trim system as follows:
    A. With landing gear up and locked and trim switch at NORM, move trim switch to NOSE UP. The horizontal stabilizer should move down at a slower rate of speed.
    B. Check low trim speed of horizontal stabilizer full trim range. Check one full length of trim travel (no less than 16 seconds minimum and no more than 23.5 seconds maximum.

NOTE

Trim speeds of the horizontal stabilizer should be different for high- and low-speed checks. If speed checks are the same, a break in electrical continuity is indicated.

12. Check left horizontal stabilizer actuator as follows:
    A. Pull out RH STABILIZER — NORMAL TRIM circuit breaker.
    B. Move trim switch to NOSE DN. The horizontal stabilizer should move leading-edge-up.
13. Check right horizontal stabilizer actuator as follows:
    A. Push in RH STABILIZER — NORMAL TRIM circuit breaker.
    B. Pull out LH STABILIZER — NORMAL TRIM circuit breaker.
    C. Move trim switch to NOSE UP. The horizontal stabilizer should move leading-edge-down.

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INSPECTION/CHECK (Continued)

CHECKING HORIZONTAL STABILIZER TRIM SYSTEM (Cont'd)

D. Push in LH STABILIZER — NORMAL TRIM circuit breaker.

14. Check pilot's trim arming switch, on normal trim switch on control wheel, as follows:

    A. With trim arming switch released, position trim switch to NOSE UP and NOSE DN. The horizontal stabilizer should not move.

    B. Hold trim switch at NOSE UP and NOSE DN and depress trim arming switch. The horizontal stabilizer should start moving.

15. Repeat step 14, on co-pilot's control wheel.

16. Disconnect 28 VDC utility external power source from airplane.

HORIZONTAL STABILIZER INSPECTION — CHECK

1. Open access panels. Check the following:
2. Horizontal stabilizer skin for cracks, buckles, dents, corrosion and loose rivets.
3. Horizontal stabilizer attach fittings for security.
4. Elevator skin for cracks, buckles, dents, corrosion and loose rivets. Trailing edge for damage and security. Leading edge skin for cracks around balance weight attach screws.
5. Elevator hinges for cracks, hinge bolts for looseness. Bonding straps for condition and security. Torque tube bearings AN201KP8A (installed within bushing 285-220406-3) for cleanliness and freedom of rotation.
6. Elevator drain holes for obstruction.
7. Install access panels.

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FLAPS - DESCRIPTION AND OPERATION

1. Wing Flaps

Wing flaps are provided to reduce stalling speeds of the airplane. The flaps are electrically controlled by a flap switch in the cockpit which controls two electromechanical actuators. The actuators mechanically position the flaps at the inboard trailing edge of both wings. Each flap is individually operated by an actuator through an individual electric circuit. The actuators are mechanically interconnected by a flexible shaft to synchronize the flap travel. Each actuator is capable of driving the opposite flap through the synchronizing flexible shaft, along with the flap to which it is attached. Therefore, in case of a power or motor failure to one actuator, flap positioning (at a reduced speed) is still obtainable. A flap position transmitter is mechanically connected to the left flap and electrically operates a flap position indicator in the cockpit.

Wing flap operation is controlled by a switch identified "FLAPS" on the aft face of the center pedestal in the cockpit. Movement of the wing flap switch from **PART** to either UP or DN closes the switch and directs electrical power to the wing flap actuators. (See figure 27-50-1.) When the actuators are energized, the output shaft is either extended or retracted, depending upon the position of the wing flap switch.

On the down cycle, the extend-limit switch in each actuator opens and breaks the circuit to the extend field winding when the flaps reach full-down position. On the up cycle as the actuator is retracted, a travel limit switch and a load limit switch (in each actuator) open the circuit when the flaps reach full up and the actuator operation is resisted by the flap mechanical up-stops. This results in a torque load to preload the wing flaps in the streamlined position.

The flap system (figure 27-50-1) is a three position system with internal limit switches built into the flap actuators. The three flap positions are UP, PART and DN. Selecting the PART position will place the flaps at the 60% position.

The wing flap position indicator, on the center instrument panel, is electrically operated by a transmitter attached to the trailing edge of the left wing. When the flap moves up or down, an arm rotates the shaft of the transmitter. Movement in the transmitter electrically reproduces a similar movement in the indicator. The indicator reflects the position of the flaps in percentage up or down.

2. Wing Flap Actuators

Two linear-type, 28-volt dc, electromechanical actuators operate the wing flaps. The wing flap actuators, one in each wing, are attached to the wing structure about midway between flap tracks. The actuators are interconnected by a flexible shaft to ensure synchronized flap trave. Nonjamming internal mechanical stops are incorporated in the actuators to resist jackscrew displacement in case of failure of a travel limit switch, load limit switch, or relay in the actuator. Two limit switches in each actuator serve

to stop the flaps automatically at the selected position when the flap switch is placed at PART.

3. Wing Flap Position Transmitter

The wing flap position transmitter, in the left wing trailing edge, electrically operates the wing flap position indicator. The transmitter is mechanically connected by a rod to the inboard rib near the forward leading edge of the flap. Movement of the flap operates the transmitter. The transmitter then sends a signal to the indicator in the cockpit. Retracting the flap toward the up position produces A, B, C direction of rotation of the wiper inside the transmitter. Movement of the wiper electrically reproduces a similar movement in the indicator.

4. Wing Flap Position Indicator

The wing flap position indicator is near the bottom of the center instrument panel. A rotating pointer indicates the position of the flaps in percentage up or down. The indicator is electrically positioned by a wing flap position transmitter. The instrument is calibrated in percent of flap extended and reads from "UP" (0) to "DOWN" (100) in increments of 10 percent. Full flap travel is 25 degrees (100 percent). This indicator facilitates the selection of intermediate flap positions.

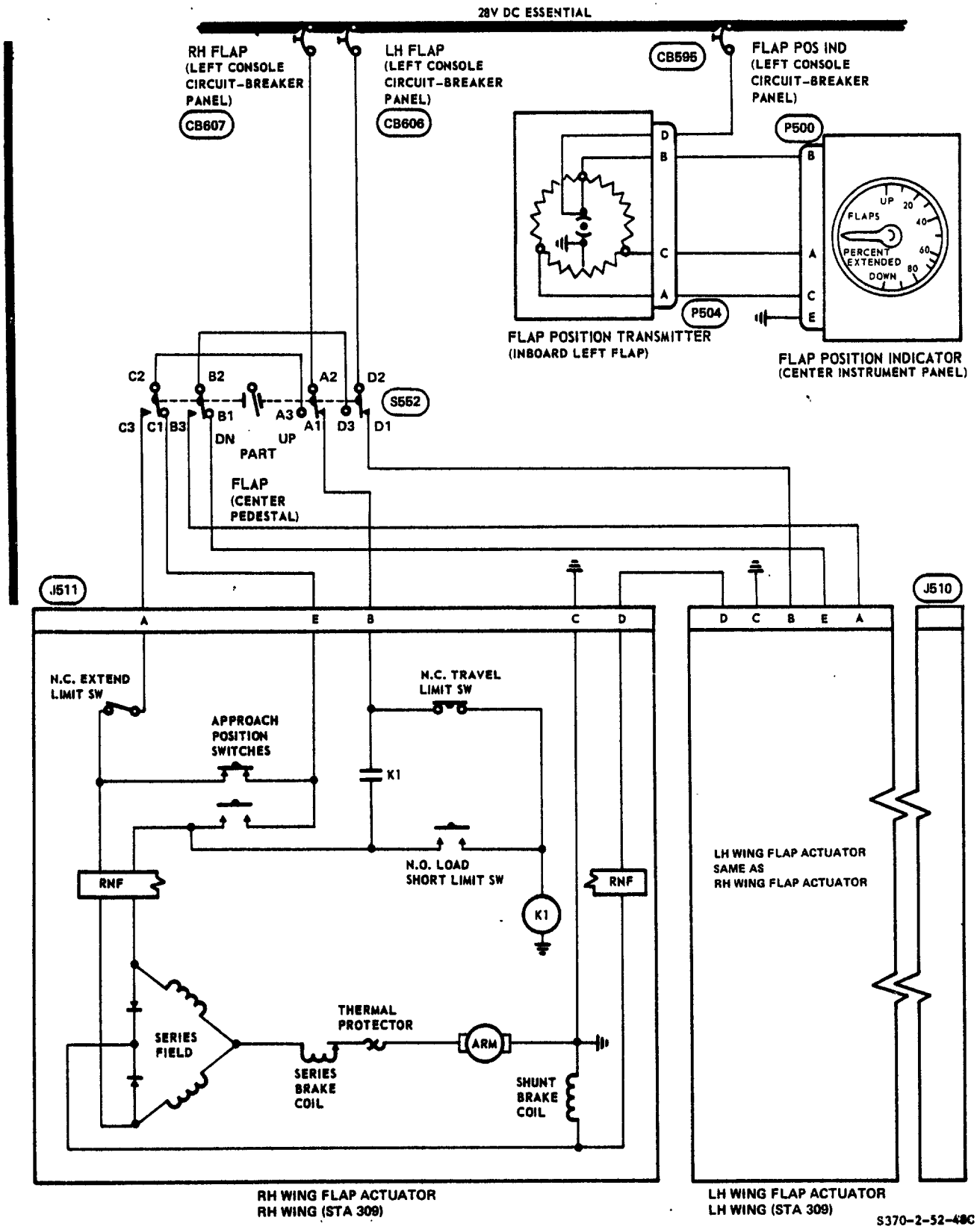
5. Wing Flap Switch

The wing flap switch, recessed in the center pedestal to prevent inadvertent operation, is powered by the dc essential bus and moves in a quadrant marked "UP," "PART," and "DN." To position the flaps full up or full down, the flap switch is moved to the corresponding position and left there. (Power is automatically removed from the actuators when the flaps reach the extreme position.)

The PART position incorporates a "lever-lock" feature which will stop the switch at the PART position when the switch handle is being moved from UP. To move to DN, pull out on switch handle to clear one-way gate. The switch can be freely returned from the DN to the PART or the UP position.

6. Flap Travel Monitor Switch

The flap travel monitor switch is installed in the right wheel well and connected to the inboard flap rib. The switch is adjusted to actuate when the flaps are 14 (±1) degrees. down. The switch is connected into the landing gear position indicating and warning system. For further information, removal, installation, and rigging procedures for this switch, refer to Chapter 34.



Wing Flap System Schematic - Three-position Flap
Figure 27-50-1

FLAPS - TROUBLE SHOOTING

1. Trouble Shooting Wing Flaps

NOTE: The following trouble shooting information is based on the assumption that electrical power is available to the system. It is suggested that frequent reference be made to the wing flap system schematic during trouble shooting of the system. This will aid in isolating a specific trouble.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. FLAPS NOT SYNCHRONIZED.		
(1) Flexible shaft improperly connected, not synchronized, or broken.	Disconnect flexible shaft in wheel well. Check for broken shaft and proper meshing of splined connections.	Replace broken shaft. Retract both flaps to full up and preloaded position; then reconnect flexible shaft.
B. FROZEN SYSTEM.		
(1) Mechanical failure in one actuator, causing other actuator to be held immobile through synchronizing shaft.	Disconnect flexible shaft in wheel well. Pull one circuit breaker; operate opposite actuator. Reverse procedure for other side.	Replace defective actuator. Repair or replace faulty wiring.
C. FLAPS DO NOT REACH LIMIT OF UP OR DOWN TRAVEL.		
(1) Flap improperly installed.	Check installation of flap. (*See figure 27-50-401.*)	
(2) Actuator not fully extended or retracted.	Disconnect flap interconnect shaft in wheel well and operate each flap independently of the other to isolate defective actuator.	Replace defective actuator.
D. FLAPS DO NOT OPERATE TOGETHER WITH ONE ACTUATOR.		
(1) Defective flexible shaft.	Check flexible shaft for proper meshing of splined connections in wheel well and at actuator, and for excessive binding or friction in the flexible shafts.	Replace defective shafts. Retract both actuators to preloaded position, and reconnect flexible shafts at coupling.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
(2) Defective actuator.	Disconnect flexible shaft and operate actuator.	Replace defective actuator.
E. FLAP POSITION INDICATOR INOPERATIVE.		
(1) Electrical system faulty.	Perform continuity check.	Repair electrical system.
(2) Faulty transmitter.	Check transmitter. If transmitter does not meet check-out requirements, a faulty transmitter is indicated.	Replace transmitter.
(3) Faulty flap position indicator.	If there is continuity in electrical system and transmitter meets check-out requirements, a faulty flap position indicator is indicated.	Replace flap position indicator.
F. FLAPS DO NOT MEET TIMING REQUIREMENTS.		
(1) Improper timing of flaps due to binding.	Refer to trouble D.	Correct binding.
(2) Open circuit breaker, open circuit to actuator, or open circuit in actuator.	Disconnect electrical plug at actuator and check for 28-volt power to actuator. If power is available, check actuator receptacle and actuator.	Replace circuit breaker, repair open circuit, or replace actuator.
(3) Improper timing due to faulty switches or improperly functioning actuators.	Check for binding and that all mechanical parts are properly adjusted.	Remove and replace actuator.
G. FLAPS DO NOT STOP AUTOMATICALLY AT PART POSITION.		
(1) Internal limit switch failure in actuator.	Disconnect flap interconnect shaft in wheel well and operate each flap independently to isolate defective actuator.	Replace defective actuator.
H. FLAP PRELOAD CANNOT BE OBTAINED.		
(1) Flap actuator load limit switch inoperative.	Recheck flap system to ensure that flaps are properly rigged.	If flaps are rigged properly, replace defective actuator.

FLAPS - MAINTENANCE PRACTICES

1. Servicing Wing Flaps

The only servicing required on the wing flaps is lubrication of the wing flap actuators. There are two grease fittings, one at the rear spar actuator-to-wing attach point, and the other at the actuator-to-flap attach point. The flap track rollers may be lubricated in accordance with figure 27-50-404.

1. Removing Wing Flaps

A. The removal procedures for the left and right flaps are the same except as noted.

NOTE: If the same flap is to be reinstalled, do not disturb settings of up-stop bolts, roller trucks, or actuator end fitting. If these adjustments are disturbed, the complete installation and adjustment procedure must be checked when the flap is reinstalled.

(1) Connect external dc power supply to airplane, and check that "LH - FLAP" and "RH - FLAP" circuit breakers are engaged; then move wing flap switch to DN. When flaps reach their full down position, turn off power supply.

(2) Disconnect flap bonding braids from wing structure.

(3) Disconnect flap position transmitter rod from inboard end of left flap. (The right flap does not have a transmitter.)

(4) Remove stop screws from end of tracks.

(5) With flap supported, disconnect actuator end fitting from flap by removing the two screws from top of flap to release flap clevis. Then guide flap off tracks. If actuator or flap is being replaced, disconnect flap clevis from actuator end fitting.

2. Installing Wing Flaps

A. The installation procedures for the left and right flaps are the same except as noted.

NOTE: This procedure is based on the assumption that the flap to be installed has been previously adjusted, and that adjustments to up-stop bolts, flap roller trucks, and the actuator end fitting have not been disturbed. If these adjustments have been disturbed, readjust as necessary as shown in figure 27-50-401.

(1) Check that actuator is properly installed.

(2) Check that flap clevis is properly installed on actuator end fitting and that actuator is fully extended.

(3) Guide flap on tracks and connect actuator to flap by installing the two screws through flap and flap clevis.

(4) Install stop screws at aft end of flap tracks.

(5) Connect flap bonding braids and, on left flap only, connect flap position transmitter linkage.

(6) Check flap operation.

CAUTION: IF THE FLAP INTERCONNECT SHAFT HAS BEEN DISCONNECTED, THE FLAPS MUST BE RESYNCHRONIZED. DISCONNECT SHAFT IN WHEEL WELL BEFORE RETRACTING FLAPS; THEN, WHEN BOTH FLAPS ARE FULLY RETRACTED AGAINST THEIR UP-STOP BOLTS, RECONNECT SHAFT.

3. Removing Wing Flap Actuators

A. The removal procedures for the left and right flap actuators are the same.

(1) Remove wing flaps.

(2) Disconnect actuator electrical plug.

(3) Disconnect flexible shaft from actuator.

(4) Remove bolt from forward end of actuator and remove actuator.

4. Installing Wing Flap Actuators

A. The installation procedures for the left and right flap actuators are the same.

(1) Position actuator with rod end aft; install attaching bolt at rear spar, and safety with cotter pin.

(2) Connect actuator electrical plug, and safety.

(3) With external dc power supply connected to airplane and "LH - FLAP" and "RH - FLAP" circuit breakers engaged, position wing flap switch at DN, to fully extend actuators.

(4) Connect flap interconnect shaft to upper fitting on actuator. Disconnect interconnect shaft in wheel well. (The shaft is reconnected only after flaps are installed and both fully retracted against their up-stop bolts.)

CAUTION: WHEN CONNECTING FLAP INTERCONNECT SHAFT, MAKE CERTAIN THAT SPLINED COUPLINGS ARE ALIGNED BEFORE TIGHTENING SLEEVE NUT. IT IS POSSIBLE TO ENGAGE A THREAD OR TWO OF THE COUPLING BEFORE THE SPLINES ENGAGE. IF THE SPLINES ARE MISALIGNED AND THE NUT IS THEN TIGHTENED, THE COUPLING MAY BE RUINED.

(5) Lubricate actuator.

(6) Install wing flap.

NOTE: If installing an actuator other than the one previously removed, perform applicable steps of figure 27-50-401 to ensure proper preload on flap.

5. Removing Wing Flap Position Transmitter

A. The wing flap position transmitter is on the left wing, and is connected by a rod to the inboard forward edge of the left flap.

(1) Through left main gear strut well, disconnect electrical plug.

(2) Disconnect arm from shaft.

(3) Remove the four screws from bracket, and remove transmitter.

6. Installing Wing Flap Position Transmitter

A. To install flap position transmitter, proceed as follows:

(1) Install transmitter in left main gear strut well with the four screws.

(2) Connect arm to shaft with 1/8-inch clearance between arm and transmitter.

(3) Connect electrical plug, and safety.

(4) Check transmitter.

7. Adjusting Wing Flaps

For adjustment of wing flaps, see figure 27-50-401.

8. Checking Wing Flap Actuator Jackshaft End Play

A. The following is a suggested method of determining the flap actuator jackshaft end play.

(1) Remove wing flap(s). (Refer to Removing Wing Flaps.)

(2) Attach a cylinder gauge (Starrett Cylinder Gauge, Model 452B or equivalent) to the jackscrew housing. (See figure 27-50-402.)

(3) Position measuring rod against rear face of flap actuator.

(4) Attach push-pull scale (Chatillion Model PPH100 or equivalent) to actuator support adapter (P/N 265-180089-3). Apply a 5-pound load (push on adapter toward actuator housing and set gauge at zero).

(5) Apply a 25-pound load (pull on adapter away from actuator) on jackscrew. Monitor reading on gauge while applying 25-pound load.

(6) Jackscrew end play shall not exceed 0.032 inch.

(7) Install serviceable flap actuator in airplane if required.

(8) Install wing flaps.

(9) Perform operational checkout of wing flap system.

9. Removing Outboard Wing Flap Track

A. The removal procedure for right hand and left hand outboard flap tracks (see figure 27-50-403), are as follows:

(1) Remove wing flap.

(2) Remove NAS1105-8 bolt, 2W18-516 washer and NAS1291-5 nut that attaches flap track to spar lug.

(3) Remove two screws, washers and nuts that attach flap track to lower wing skin.

(4) Remove the flap track and record shims if installed.

10. Installing Outboard Wing Flap Track

B. The installation procedure for right hand and left hand outboard flap tracks are as follows:

(1) Install the two screws, washers and nuts attaching the flap track to the lower wing skin. (See figure 27-50-403 for screw installation). Torque nuts 140 to 203 inch pounds (use Torq-Set screwdriver, EX-170-5/16 and SC-314 adapter).

(2) Check alignment of hole in flap track with hole in wing spar lug. If holes do not align (slip fit), remove two screws installed in step (1); then insert shim between flap track and lower wing skin as necessary to obtain slip fit for NAS1105-8 bolt with flap track attach screws installed and nuts torqued.

NOTE: Make tapered shim from 2024-T42 bare aluminum sheet, and not over .080 inch thickness. Paint shim with jet-skin white primer.

(3) Install NAS1105-8 bolt, washer and torque nut to 140 to 203 inch pounds.

(4) Install wing flap.

10A. Removing Inboard Wing Flap Track

A. The removal procedure for right hand and left hand inboard flap tracks (see figure 27-50-403), are as follows:

(1) Remove wing flap.

(2) Remove NAS1108-13 bolt, 2W18-816 washer, and MS20364-820 nut that attaches flap track to spar lug.

(3) Remove four screws, washers and nuts that attach flap track to lower wing skin.

(4) Remove the flap track and record shims if installed.

10B. Installing Inboard Wing Flap Track

A. The installation procedure for RH and LH inbd flap tracks are as follows:

(1) Install the four screws, washers, and nuts attaching the flap track to the lower wing skin. (See figure 27-50-403 for screw installation.) Torque nuts to 140 to 203 in.-lbs (use Torq-Set screwdriver, EXT-170-5/16 and SC-314 adapter).

NOTE: Make tapered shim from 2024-T42 bare aluminum sheet, and not over .080 thickness. Paint shim with jet-skin white primer.

(2) Check alignment of hole in flap track with hole in wing spar lug. If holes do not align (slip fit), remove four screws installed in step (1); then insert shim between flap track and lower wing skin as necessary to obtain slip fit for NAS1108-13 bolt with flap track attach screws installed and nuts torqued.

(3) Install NAS1108-13 bolt, washer and torque nut to 140 to 203 in.-lbs.

(4) Install wing flap.

11. Replacement of Flap Interconnect Coupling

A. To remove flap interconnect coupling, turn on electrical power and raise flaps.

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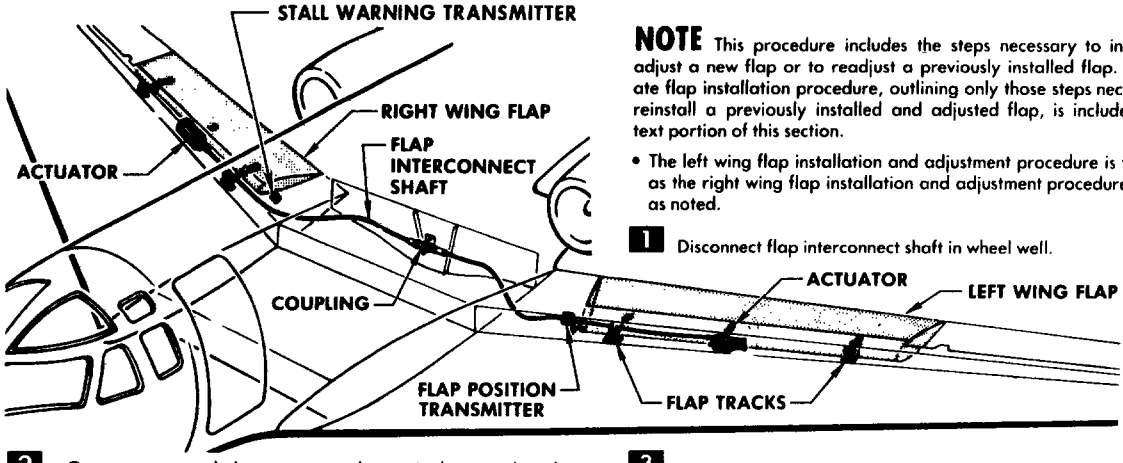


NOTE This procedure includes the steps necessary to install and adjust a new flap or to readjust a previously installed flap. A separate flap installation procedure, outlining only those steps necessary to reinstall a previously installed and adjusted flap, is included in the text portion of this section.

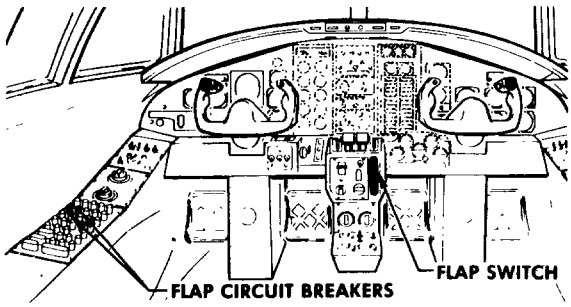
- The left wing flap installation and adjustment procedure is the same as the right wing flap installation and adjustment procedure, except as noted.

1 Disconnect flap interconnect shaft in wheel well.

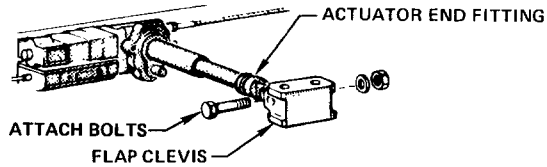
2 Connect external dc power supply to airplane and push in circuit breaker controlling flap to be installed; then pull circuit breaker controlling opposite flap, to prevent unnecessary operation of opposite flap. (Refer to "Wing Flap System Schematic" illustration.)

3 Check that actuator is properly installed. (Refer to "Installing Wing Flap Actuators.")

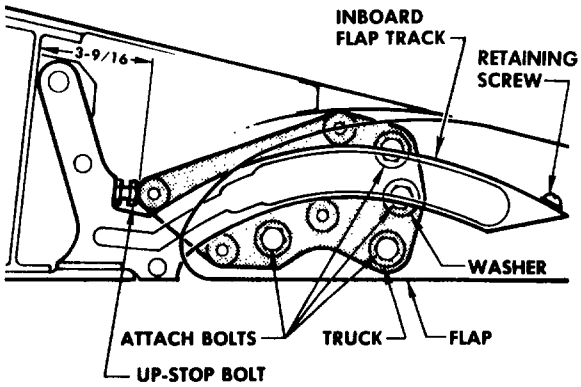
4 Move "FLAP" switch in cockpit to UP, to fully retract actuator.



5 Install flap clevis on actuator end fitting. Install and tighten attach bolts.



6 Remove retaining screw from aft end of each flap track.

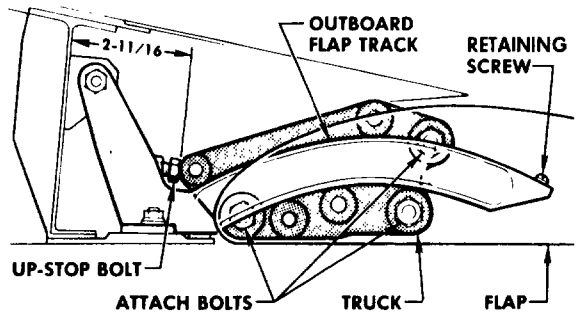


7 Install a short rig pin in the aileron control sector located in the area of the outboard flap track.

Caution The rig pin must be inserted all the way. If the rig pin extends too far aft of the sector, the flap leading edge will be damaged when the flap is retracted.

8 Secure flap bonding braids and, on the left flap only, the transmitter link, in positions such that the bonding braids and the transmitter link will not interfere with flap operation.

9 Check that flap up-stop bolts are properly adjusted, as illustrated.



10 Remove flap adjustment access covers on flap lower surface; then adjust flap roller truck-to-flap attach bolts so that they are loose enough to permit truck to be shifted about within the limits of oversize mounting holes provided in flap ribs.

NOTE The washers under the truck-to-flap attach bolts must have a fresh surface for the serrations on the bushings to "bite into." If the washers have been used, they may be either turned, or replaced with new ones made from 0.125-inch thick 2024-T4 bare aluminum alloy material.

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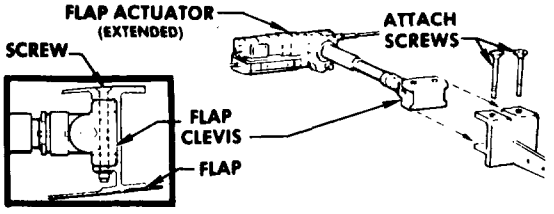
11 Extend flap actuator, then guide flap on tracks and install retaining screws and washers in ends of flap tracks. Then connect actuator to flap by installing the two long screws through the flap clevis, as shown.

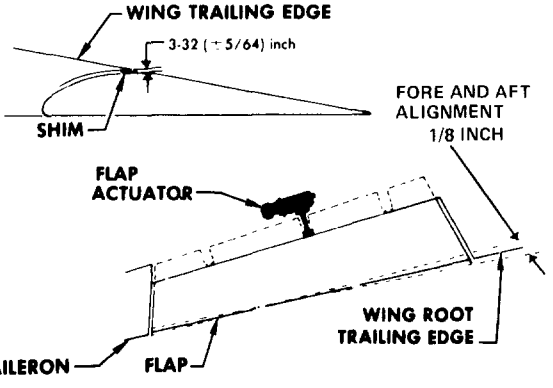


12 Move "FLAP" switch to UP position. Allow flap to retract and shut off automatically.

NOTE Actuator overheating may be prevented by operating actuator through one complete, continuous extension or retraction cycle followed by a 2-minute off period.

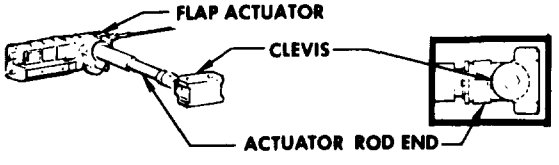
13 Adjust length of actuator as follows:

a. Tape 3 32-inch thick shims to top surface of flap, in the area under the wing trailing edge.

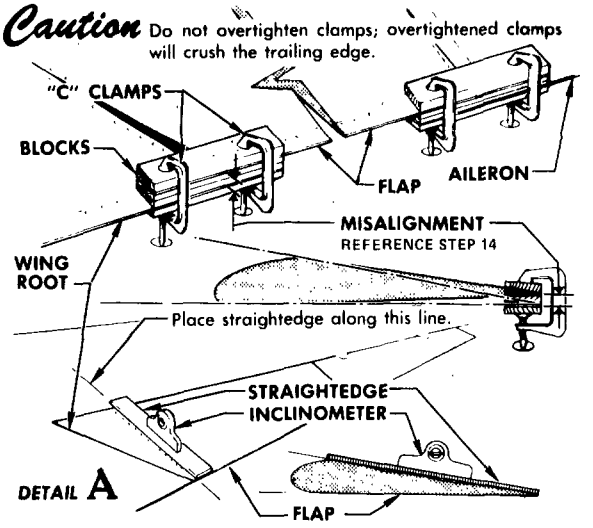


b. Hold flap leading edge up against shims; then, after ensuring that flap trailing edge is aligned within 1/8 inch with aileron and wing root trailing edges, check fore-and-aft alignment of flap trailing edge. Shift flap, to equalize inboard and outboard fore-and-aft misalignment, and note the misalignment dimension.

c. Remove small access cover on lower surface of flap near actuator attach point. Rotate actuator end fitting as necessary to align flap trailing edge to within 1/8 inch with wing root and aileron trailing edges.

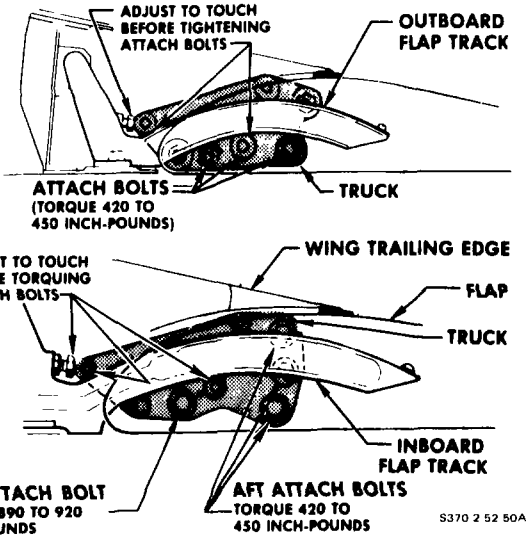


Caution Do not overtighten clamps; overtightened clamps will crush the trailing edge.



14 Adjust flap inboard and outboard roller trucks as follows:

a. Retract flap, allowing actuator to shut off automatically.
b. Locate wing reference plane, and record angle found.
c. Place inclinometer on straight edge, at detail A, of flap, between flap trailing edge and heel along first row of rivets outboard of inboard edge.
d. Measure angle of straight edge. This angle must be 3-1/4(± 1/4) degrees more than wing reference plane angle for required 0 degree flap-up trailing edge rig position.
e. Measure misalignment between inboard flap trailing edge and outboard edge of wing root trailing edge. Record distance.
f. Using wood blocks and "C" clamps as shown, secure flap trailing edge to the dimension noted in step e. (This will hold the flap trailing edge in rig position while the flap roller trucks are being adjusted and secured.)
g. Hold flap leading edge up against shims. Push roller trucks forward as far as possible and push up on aft end of truck; then torque truck-to-flap attach bolts to values shown.
h. Remove blocks and "C" clamps.



Installing and Adjusting Wing Flaps
Figure 27-50-401 (Sheet 2 of 4)

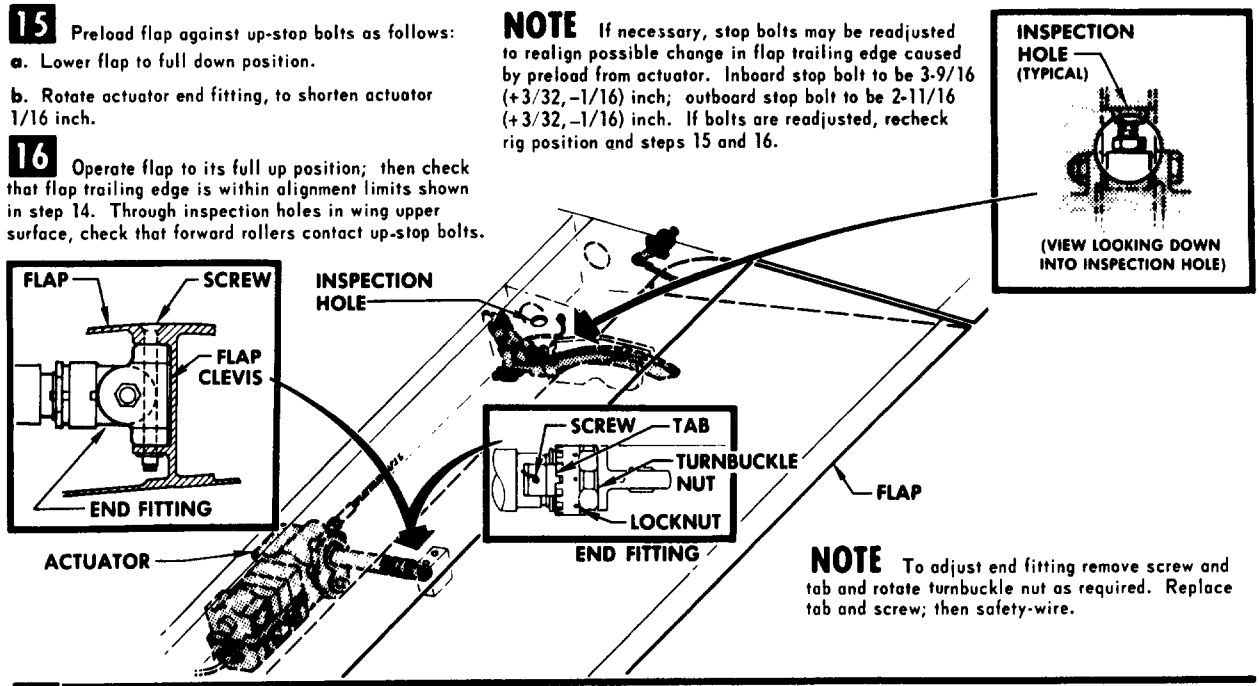
15 Preload flap against up-stop bolts as follows:

a. Lower flap to full down position.

b. Rotate actuator end fitting, to shorten actuator 1/16 inch.

16 Operate flap to its full up position; then check that flap trailing edge is within alignment limits shown in step 14. Through inspection holes in wing upper surface, check that forward rollers contact up-stop bolts.

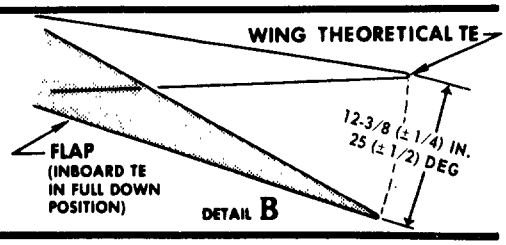
NOTE If necessary, stop bolts may be readjusted to realign possible change in flap trailing edge caused by preload from actuator. Inboard stop bolt to be 3-9/16 (+3/32, -1/16) inch; outboard stop bolt to be 2-11/16 (+3/32, -1/16) inch. If bolts are readjusted, recheck rig position and steps 15 and 16.



NOTE To adjust end fitting remove screw and tab and rotate turnbuckle nut as required. Replace tab and screw; then safety-wire.

17 Operate flap to full down position. Check that distance between flap trailing edge and theoretical wing trailing edge is as shown in detail B.

Caution Do not use limit switches to obtain travel positions; if the flaps have been properly adjusted, travel positions will be within tolerance.



18 If right flap is being adjusted, proceed to step 19; if left flap is being adjusted, rig flap position indicator system as follows:

a. Check that "FLAPS POS IND" circuit breaker is engaged.

b. Adjust length of transmitter actuating rod, until distance between center of rod-end bolt hole and aft face of rear spar is as noted.

c. Loosen clamping nut on transmitter arm; then rotate transmitter shaft until cockpit indicator reads 100 (±3) percent flaps down.

d. Slide transmitter arm laterally on shaft to provide 1/8-inch clearance between arm and transmitter; then tighten clamping nut.

e. Operate flaps to 66-percent position mark. Check that flap position is as shown in detail C.

NOTE The transmitter actuating arm has a adjustment disc. Rotating the disc. will provide an increase or decrease adjustment range.

NOTE Flap position to be within tolerance when operating from up to approach position and from down to approach position.

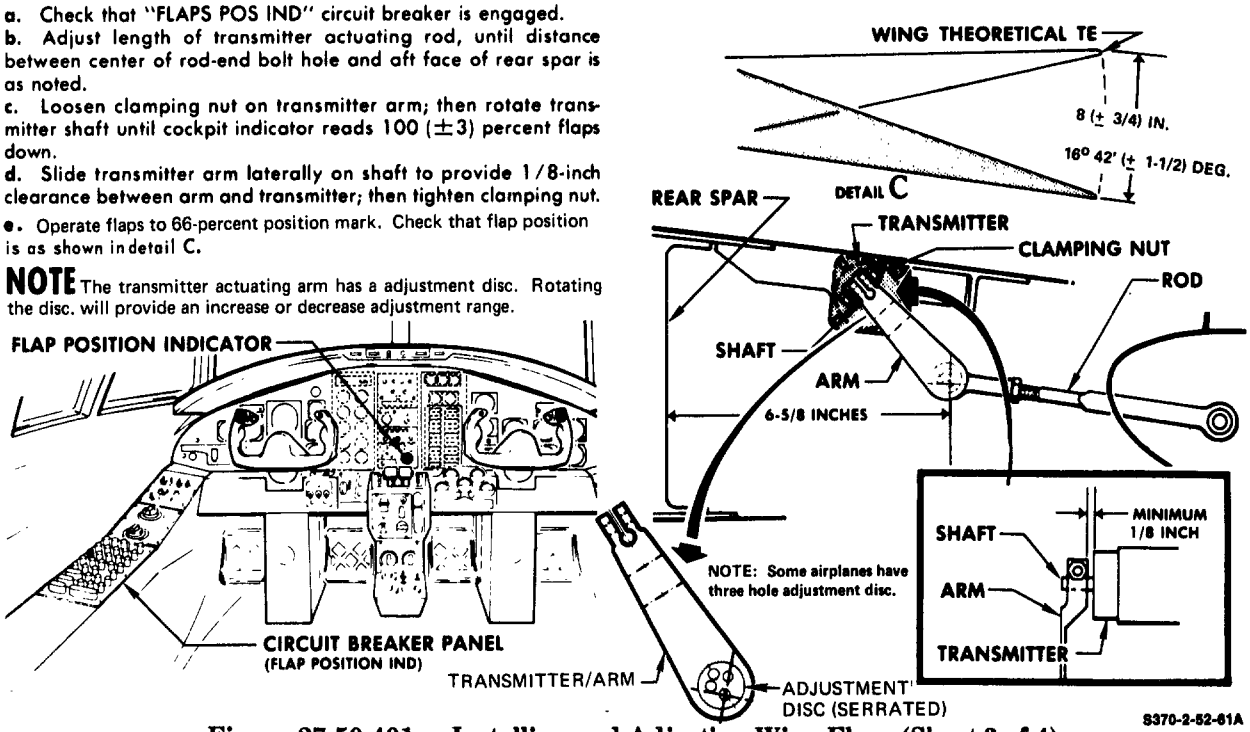


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Figure 27-50-401 — Installing and Adjusting Wing Flaps (Sheet 3 of 4)

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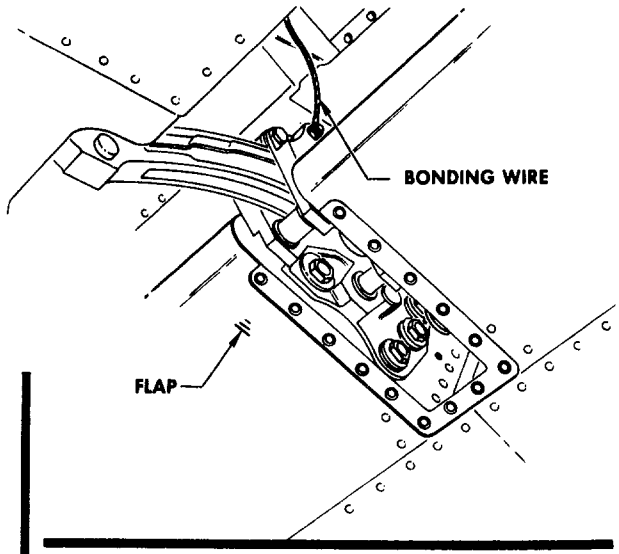


NOTE When R.H. Flap is adjusted the stall warning system switch linkage should be checked. (refer to 34-10-4)

19 Connect flap bonding braids at inboard and outboard tracks. Install flap adjustment access covers.

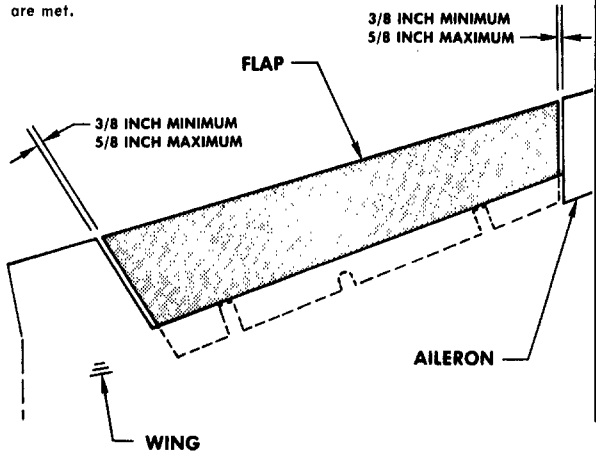
20 Cycle flap full down then to full up position: No binding should be evident throughout its full range of travel. Ensure that flap is contacting both stops, the trailing edge is within tolerance, and actuator is not fully bottomed.

21 Using a push-pull scale, apply a down load at a position 2 (± 1/2) inches forward from flap trailing edge in line with flap actuator. Flap should not rock on rollers with applied load of less than 55 pounds, but should rock on rollers before a maximum of 175 pounds. Record load at which flap rocks on rollers.

NOTE A block of wood should be used to protect flap when load is applied.

- Assure that the inboard track aft lower roller is not free to rotate, and that forward lower roller will rotate freely. On outboard track, either forward or aft roller may rotate freely.
- If flap preload cannot be obtained, refer to trouble shooting wing flaps.

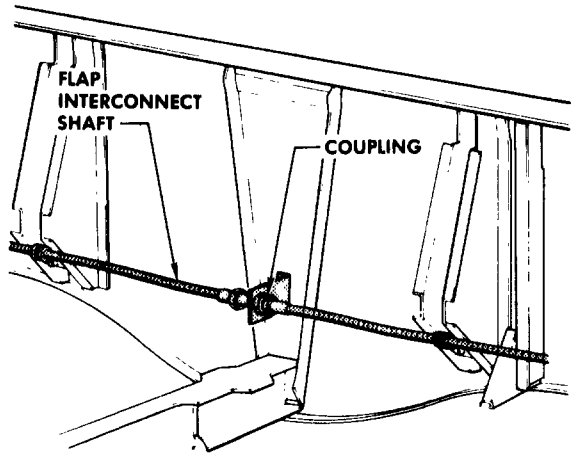
22 Check clearances between flap trailing edge and aileron trailing edge, and between flap trailing edge and wing root trailing edge as shown. Trim excess material from flap until these clearances are met.



23 Engage circuit breaker controlling opposite flap. Partially extend, then retract flaps, to ensure that both flaps are full up, properly preloaded, and synchronized.

24 Connect flap interconnect shaft in wheel well.

25 Partially extend, then fully retract flaps. Verify procedures in step 21 for both flaps. Difference between loads on left and right flaps must be within 10 percent of lowest load recorded in step 21. If load is not within tolerance, the flap with the lowest load must be rerigged.



26 Lubricate actuator.

27 Check flap operation according to procedure in "Checking Wing Flap System."

28 Remove aileron sector rig pin installed in step 7.

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Figure 27-50-401 — Installing and Adjusting Wing Flaps (Sheet 4 of 4)

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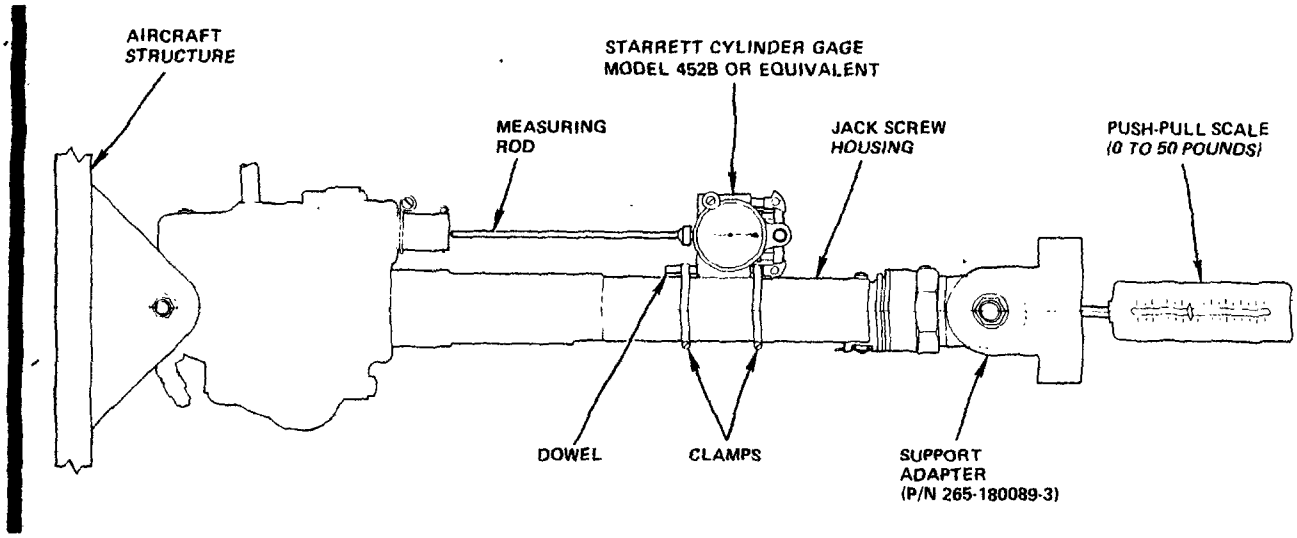


Test Jig for Checking Actuator Jackshaft End Play
Figure 27-50-402

(2) Shut off all electrical power.

(3) Remove jam nut from one side of coupling and remove coupling from mounting bracket. (See figure 27-50-401.)

B. To install flap interconnect coupling, proceed as follows:

(1) Install new coupling on bracket and secure with jam nut.

(2) Engage rotary flex drive shafts with interconnect coupling and tighten coupling nuts. Safety with lockwire.

(3) Perform operational check-out of wing flaps.

12. Replacement of Flap Rotary Flex Drive

A. To remove flap rotary flex drive, proceed as follows:

(1) Turn on electrical power and lower flaps for access to wing rear spar.

(2) Turn all electrical power off. Remove wheel well covers.

(3) Disconnect flex shaft coupling at each flap actuator.

(4) Remove all accessible clamps from flex drive.

(5) Working in main gear wheel well, disconnect both flex shafts from interconnect coupling.

(6) Remove clamps on flex shaft in wheel and strut wells.

(7) Carefully withdraw flex shaft through wheel and strut well and remove from airplane.

B. To install flap rotary flex drive, proceed as follows:

(1) Route new flex drive from flap well through bulkhead into strut well and through wing trailing edge structure into main wheel well.

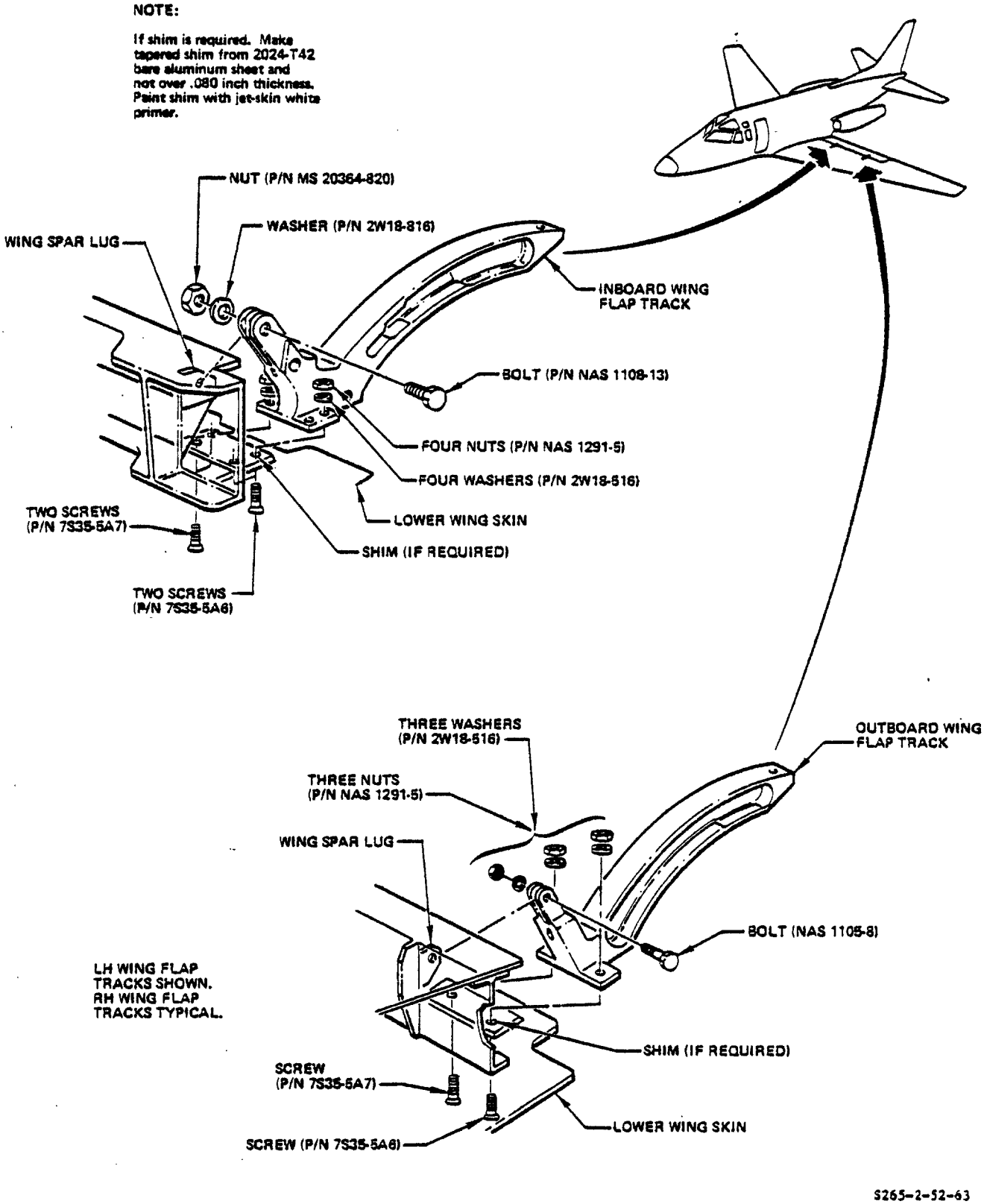
(2) Connect flex shaft coupling to flap actuator in flap well. Safety with lockwire.

(3) Turn on electrical power and fully retract both flaps.

(4) Connect flex drive couplings to interconnect couplings. Safety with lockwire.

(5) Perform operational check out of wing flap system.

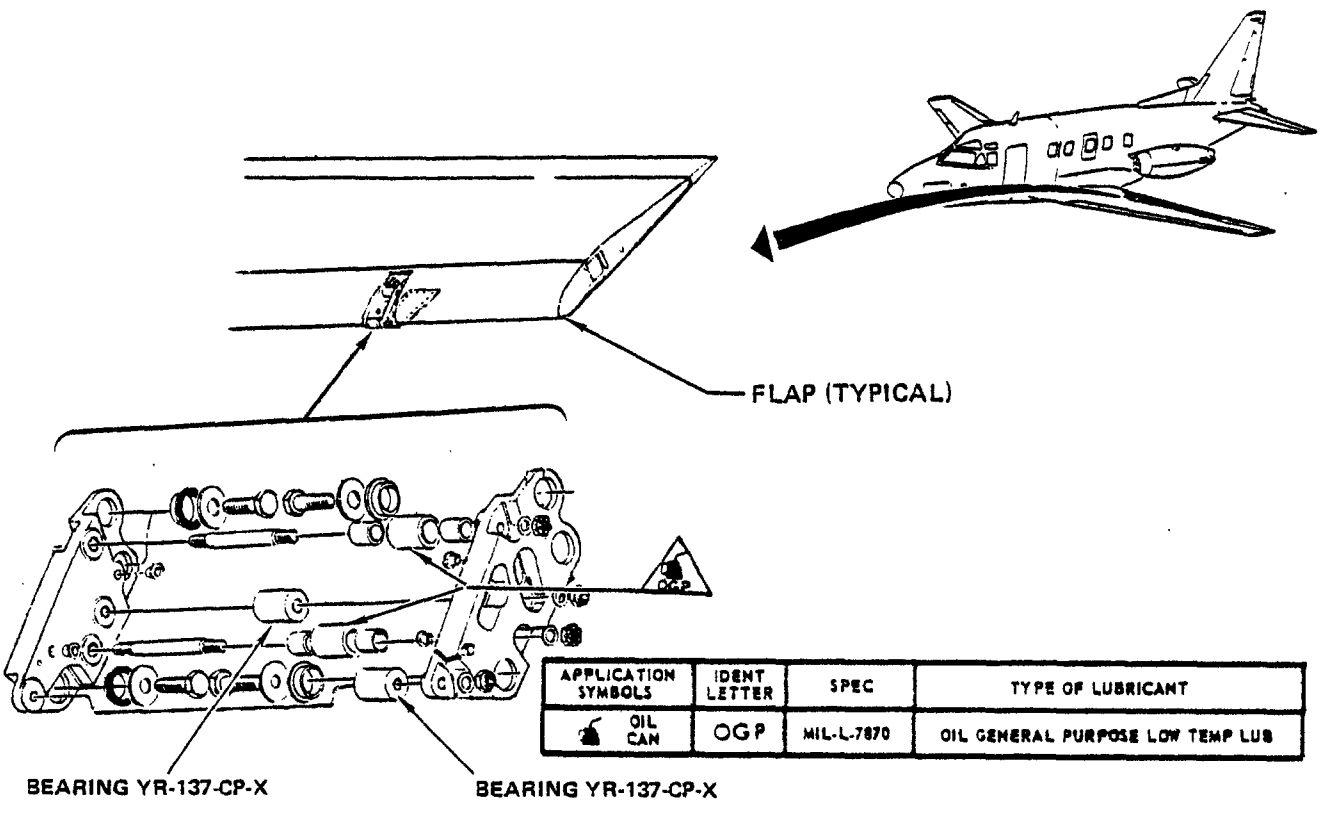
(6) Install wheel well covers.


NOTE:
If shim is required. Make tapered shim from 2024-T42 bare aluminum sheet and not over .080 inch thickness. Paint shim with jet-skin white primer.
NUT (P/N MS 20364-820)
WASHER (P/N 2W18-816)
WING SPAR LUG
INBOARD WING FLAP TRACK
BOLT (P/N NAS 1108-13)
FOUR NUTS (P/N NAS 1291-5)
FOUR WASHERS (P/N 2W18-516)
LOWER WING SKIN
SHIM (IF REQUIRED)
TWO SCREWS (P/N 7S35-5A7)
TWO SCREWS (P/N 7S35-5A6)
THREE WASHERS (P/N 2W18-516)
OUTBOARD WING FLAP TRACK
THREE NUTS (P/N NAS 1291-5)
WING SPAR LUG
BOLT (NAS 1105-8)
LH WING FLAP TRACKS SHOWN. RH WING FLAP TRACKS TYPICAL.
SHIM (IF REQUIRED)
SCREW (P/N 7S35-5A7)
LOWER WING SKIN
SCREW (P/N 7S35-5A6)
S265-2-52-63

Removing and Installing Wing Flap Tracks
Figure 27-50-403

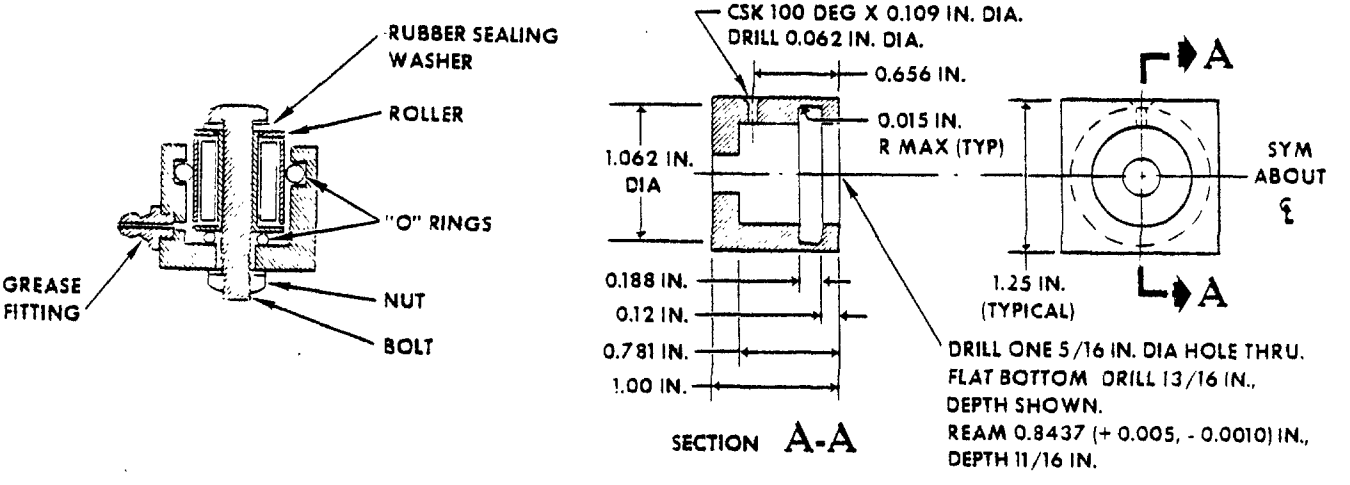
13. Flap Track Roller Lubrication

Lubricate flap track rollers as per figure 27-50-404.



APPLICATION SYMBOLS	IDENT LETTER	SPEC	TYPE OF LUBRICANT
OIL CAN	OGP	MIL-L-7870	OIL GENERAL PURPOSE LOW TEMP LUB

LOCALLY MANUFACTURED TOOL FOR McGILL BEARING YR-137-CP-X



PARTS LIST
2W18-516M WASHER (2 REQD)
MS28775-011 "O" RING (1 REQD)
MS28775-211 "O" RING (1 REQD)
MS21042L5 NUT (1 REQD)
AN5-15A BOLT (1 REQD)
MS15001-1 FITTING (1 REQD)
NAS1515D5 WASHER (1 REQD)

Make from 2024-T42 Al extr bar or Al extr shape, Fed. Spec QQ-A-200/3A, or 2024-T4 Al rolled bar, Fed. Spec QQ-A-225/6A. No finish is required.

Install roller in tool and lubricate with any one of the following greases:

- Supermil Grease No. 72832, American Oil Co
- Cosmolube 678, E. F. Houghton and Co
- Royco 27, Royal Lubricants Co
- Aeroshell Grease 7A, Shell Oil Co
- Mobilgrease 27, Socony Mobil Co, Inc
- Uni Temp EP, Texaco, Inc
- Sinclair 3827 Grease, Sinclair Refining Co
- Ped-3527, Standard Oil of Calif.
- Castrolease A1, Castrol Oils, Inc

Flap Track Roller Lubrication
Figure 27-50-404

1. Checking Wing Flap System

A. Before check-out, the wing flap system must be properly installed and adjusted. (Refer to procedure for installing wing flaps, and see figure 27-50-401.)

(1) Connect 28-volt dc utility power source to airplane.

(2) Push in the following circuit breakers on left console circuit-breaker panel:

- "LH - FLAP"
- "RH - FLAP"
- "POS IND - FLAP"

(3) Move wing flap switch, on center pedestal, to DN. The flaps should fully extend, and the actuator should shut off automatically. The wing flap position indicator should read flaps extended 100 (±3) percent. The trailing edges of the flaps should be 25-1/2 (±1/2) degrees down from the inboard trailing edge of the wing, a distance of 12-3/8 (±1/4) inches.

NOTE: To prevent overheating actuators during checking procedure, the actuators may be operated for one complete cycle followed by 2 minutes off.

(4) Insert 5/16-inch diameter rig pins in both aileron outboard cable sectors to make check in step (7). Use short rig pins to prevent damage to flap leading edges.

(5) Place a small piece of masking tape on each of the four up-stop bolts to make check in step (11).

(6) Place wing flaps switch at UP and allow both flaps to retract until actuators stop running. Note that, as actuators top running, they shut off smoothly with no twisting action or sharp bottoming sound of jackscrews. This indicates that the actuators are being shut off by their load limit switches.

(7) Check position of flaps. The wing flap position indicator should read 0 (±3) percent.

NOTE: The flaps should be rigged at 0 degrees; and the trailing edges of the wing, aileron, and flap should be aligned within 1/8 inch. (See figure 27-50-401.)

(8) Check for 3/32 (±5/64) inch clearance between top of flap and trailing edge of wing over flaps.

(9) Place wing flap switch at DN and observe actuators. When flaps are fully extended, note that actuators shut off smoothly with no twisting action or bottoming sound of their jackscrews.

(10) Remove rig pins from aileron outboard cable sectors.

(11) Check the four pieces of tape placed on flap up-stop bolts in step (5). Each should bear a mark indicating that the pressure applied by the flaps in their retracted position was about even. This indicates that the flaps are evenly preloaded against up-stop bolts. Remove tape.

(12) Check approach position of flaps as follows:

(a) Place flap switch at PART. Flaps should rise to 60% (±9%) position and hold.

(b) Place flap switch at UP; when flaps are fully retracted, place switch at PART. Flaps should lower to 60% (±9) position and hold.

(c) Pull out on flap switch and move to DN. Flaps should extend fully. Place flap switch at UP.

(13) Check left flap actuator as follows:

(a) Pull out "RH - FLAP" circuit breaker.

(b) Move wing flap switch to DN. Both flaps should move down.

(14) Check right flap actuator as follows:

(a) Push in "RH - FLAP" circuit breaker.

(b) Pull out "LH - FLAP" circuit breaker.

(c) Move wing flap switch to UP. Both flaps should move up.

(15) Check timing of actuators with both circuit breakers pushed in. The wing flaps should go down in no more than 8 seconds. The flaps should go up in no more than 10 seconds.

(16) Disconnect 28-volt dc utility external power source from airplane.

INSPECTION/CHECK (Continued)

INSPECT — WING FLAP SONIC DOUBLERS

Equipment/Tools

- None

Consumables

- None

Inspection

See Figure 27-50-601 for visual inspection procedure.

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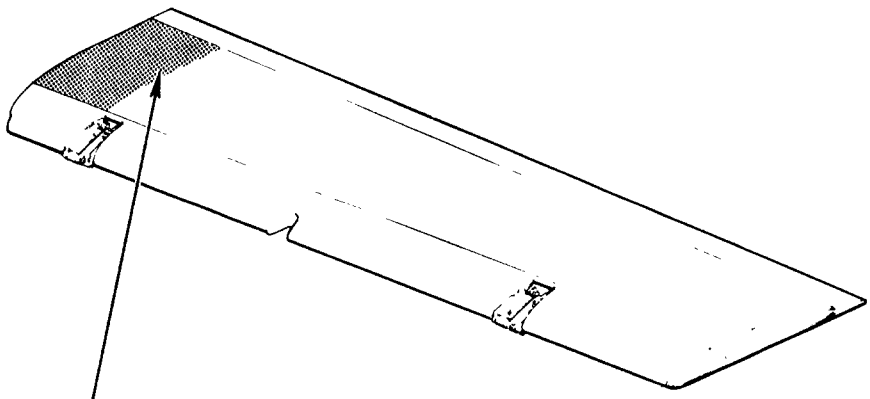

SONIC DOUBLER IS SANDWICHED BETWEEN SKIN AND FRAMES. VISUALLY INSPECT TOP AND BOTTOM FOR BULGES BETWEEN FASTENERS. BUBBLES AND FLAWS IN THE SKIN AND PAINT SYSTEM. CHECK FOR LOOSE, MISSING AND CORRODED FASTENERS.

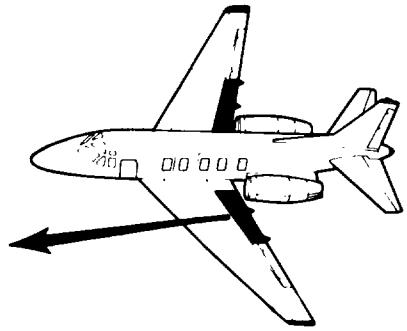

Figure 27-50-601 — Inspect — Sonic Doublers — Flaps

NEW

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SPEED BRAKE SYSTEM - DESCRIPTION AND OPERATION

1. General

The speed brake system increases the maneuverability of the airplane and reduces the landing speed by extending the speed brake panel into the air stream. The speed brake is hinged at its leading edge and, when open, extends down and forward to increase drag. (See figure 27-60-1.)

This rectangular panel is under the fuselage, forward of the main gears and aft of the nose wheel well. Two hydraulic actuating cylinders, controlled by a solenoid-operated, three-position (extend, off, and retract) valve, open and close the speed brake. Hydraulic pressure at 3000 psi is provided by the normal hydraulic power system. In case of failure of the normal hydraulic power system, 3000 psi hydraulic pressure is available from the auxiliary hydraulic system for a limited number of operations.

Direction of speed brake operation is controlled by a serrated speed brake selector switch on the left throttle. This three-position switch has an extend or open position, a retract or closed position, and an off position. The speed brake is opened by actuating the switch aft, closed by actuating the switch forward, and locked in any position by placing the switch in the center position.

When the speed brake selector switch is actuated, hydraulic power is routed from the hydraulic power system to the speed brake control valve. With the speed brake selector switch in the center or off position, hydraulic pressure is blocked at the control valve. When the selector switch is moved from off, the solenoid-operated speed brake control valve is actuated. Hydraulic pressure is then admitted through the control valve to the speed brake actuating cylinders to open or close the speed brake panel. At off, the solenoids are de-energized, and hydraulic fluid is locked in the cylinders and the speed brake is held in position. This hydraulic lock will hold the speed brake open, closed, or in any intermediate position selected.

Restrictors on the actuating cylinder lines control the opening and closing rates. A speed brake limit switch, above the left aft corner of the speed brake, is actuated when the speed brake starts to open. When the speed brake is not closed, and if either throttle is at a setting of about 95%*(80%†) rpm or greater, the "SPEED BRAKE OPEN" caution light and master caution lights are illuminated.

Electrical power for the speed brake system is supplied from the 28-volt dc essential bus.

2. Speed Brake Actuating Cylinders

The speed brake is operated by two actuating cylinders in the speed brake well. The pivoting end of each cylinder is attached to the structure at the aft end of the speed brake well. An adjustable rod end on each cylinder is attached to the speed brake panel.

*NA-370-2 through -6, -8 and -9
†NA-370-1, -7, NA-380-1 and Later

When the speed brake selector switch is positioned to extend, pressure in the speed brake open lines actuates the cylinders to the extend or open position. The speed brake closed line is the hydraulic return line. When the speed brake selector switch is positioned to retract, pressure in the speed brake closed lines actuates the cylinders to the retracted or closed position. The speed brake open line is then the hydraulic return.

3. Speed Brake Indicating System

The speed brake indicating system gives the pilot and/or copilot a visual indication that the speed brake panel is not closed when the throttle is in the 95%*(80%†) rpm position. The system consists of an amber "SPEED BRAKE OPEN" caution light, a speed brake caution relay, two microswitches (one for each throttle assembly), a speed brake limit switch, and a "SPEED BRAKE - IND" circuit breaker powered by the 28-volt dc essential bus. To check or dim the caution light, refer to description or operation of caution-warning and indicator lights dim and test system in Chapter 33.

When the speed brake panel starts to open, the speed brake limit switch is immediately actuated. However, the "SPEED BRAKE OPEN" caution light does not illuminate until either or both throttles are advanced to about 95%*(80%†) rpm.

4. Speed Brake Control Valve

The speed brake control valve is on the bottom of the fuselage, left side, at fuselage station 196. (See figure 27-60-1.) The valve is a four-port, electrically controlled, solenoid-operated valve. Internally, this valve consists of electrically controlled pilot valve poppets and a pressure-operated slide valve. The pilot valve poppets are spring-loaded to the neutral position. The slide valve is spring-loaded to the neutral position by springs at each end of the slide. When hydraulic pressure is available and the speed brake selector switch is at off, the valve locks the hydraulic fluid within and holds the actuating cylinders in fixed positions. When the speed brake selector switch is placed at extend position, the open solenoid is energized. This causes the spring-loaded pilot valve poppet to be displaced. Necessary ports are then connected to allow entry of hydraulic fluid under pressure from the hydraulic system. Fluid is then directed to the end of the slide, repositioning it within the sleeve assembly. This connects the necessary ports to direct hydraulic pressure to extend the speed brake. Return fluid is routed through the properly positioned slide in the valve to a system return line. When the speed brake switch is returned to off, the solenoid is de-energized. This allows the spring-loaded pilot valve poppets and slide to return to neutral position, blocking hydraulic pressure at the entry to the valve. Movement of the speed brake switch to the closed position energizes the opposite solenoid, causing the internal mechanism of the valve to be shifted in the opposite direction. Hydraulic pressure is then routed to close the speed brake.

5. Speed Brake Dump Valve

The speed brake system dump valve is aft of the speed brake panel and can be reached through an access panel aft and to the left of the speed brake. The dump valve is used, in case of electrical or hydraulic power system failure, to remove hydraulic pressure trapped in the speed brake actuating

*NA-370-2 through -6, -8 and -9
†NA-370-1, -7, NA-380-1 and Later

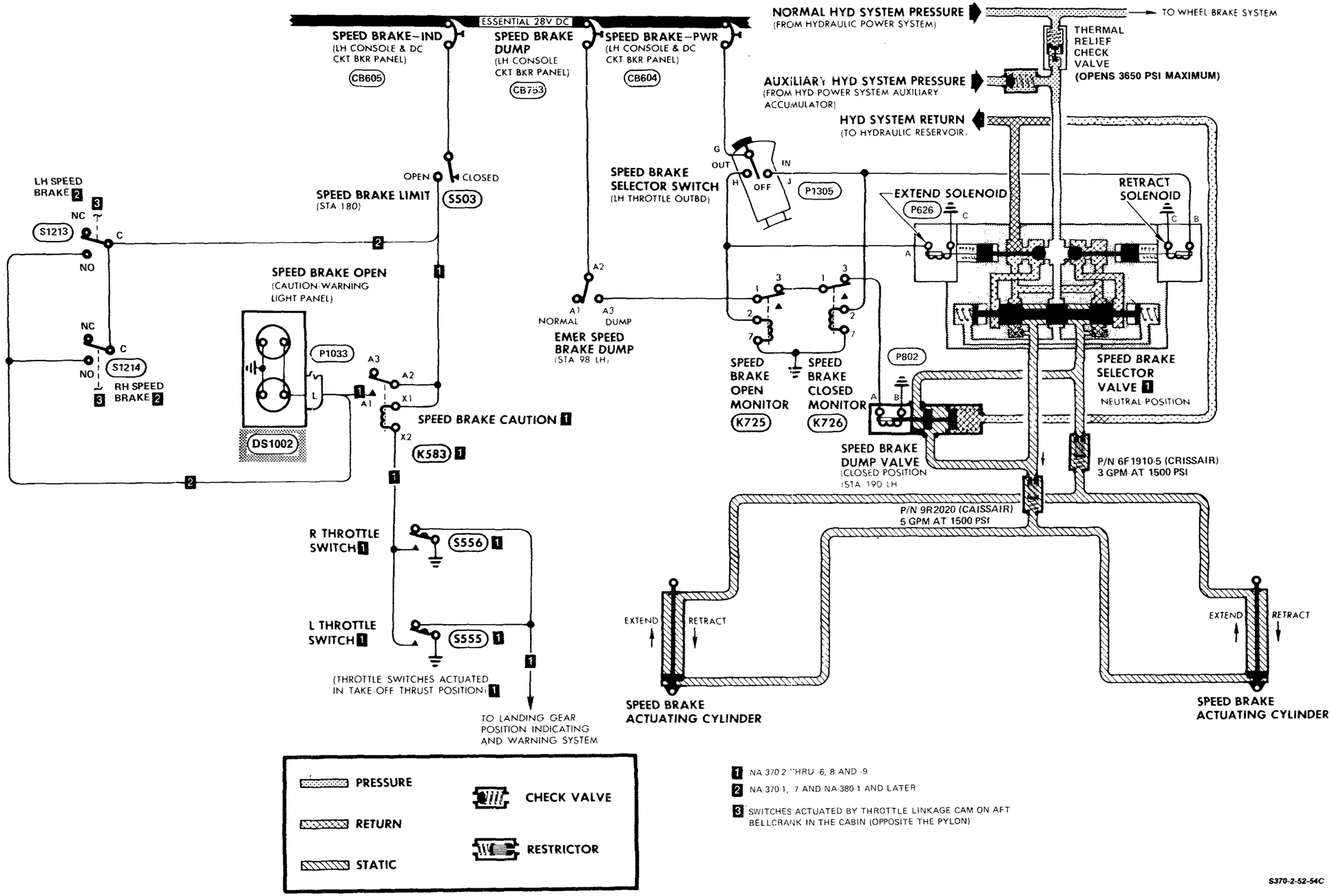
cylinders. The dump valve is a three-port, two-position, electrically operated valve. The dump valve is normally closed and is electrically actuated to the open position. When the valve is actuated to the open position, hydraulic fluid from the speed brake open and the speed brake closed lines is allowed to dump into the hydraulic system return line. This condition allows the speed brake panel to return to a "trail" or semiclosed position by in-flight air loads on the panel. The dump valve is also used during maintenance work when a change in speed brake position is desired and hydraulic pressure is not available.

6. Speed Brake Monitor Relays

Two speed brake monitor relays (speed brake closed monitor relay and speed brake open monitor relay) are incorporated in the speed brake dump valve system. (See figure 27-60-1.) The relays are used to prevent operation of the dump valve when the speed brake selector switch is in either extend or retract position. A set of normally closed contacts of each relay are connected in series in the circuit of the speed brake dump valve. Positioning the emergency speed brake dump switch to the IN (retract) or OUT (extend) position will actuate one of the speed brake monitor relays, thereby opening the circuit to the speed brake dump valve When the speed brake selector switch at off (centered) position (speed brake monitor relays de-energized) and the emergency speed brake dump switch at DUMP, electrical power passes through the normally closed set of contacts of the monitor relays, energizing the dump valve.

7. Restrictors

A single one-way restrictor in the common return lines of the two actuators (figure 27-60-1) reduces the possibility of damage to the speed brake from differential extension or retraction of the two speed brake actuators, which could occur in case of failure, incorrect installation, or obstructions of a single restrictor.



1 NA-370-2 THRU -6, 8 AND -9

2 NA-370-1, 7 AND NA-380-1 AND LATER

3 SWITCHES ACTUATED BY THROTTLE LINKAGE CAM ON AFT BELLCRANK IN THE CABIN (OPPOSITE THE PYLON)

S370-2-52-54C

Speed Brake System Schematic
Figure 27-60-1

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SPEED BRAKE SYSTEM - TROUBLE SHOOTING

1. Trouble Shooting Speed Brake System

NOTE: If a system unit is replaced, repaired, or adjusted, an operational check-out of the system is required.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. SPEED BRAKE DOES NOT OPERATE.		
(1) "SPEED BRAKE - PWR" circuit breaker not engaged.	Check "SPEED BRAKE - PWR" circuit breaker, on pilot's left console circuit-breaker panel.	Engage "SPEED BRAKE - PWR" circuit breaker
(2) Faulty "SPEED BRAKE - PWR" circuit breaker.	Close circuit breaker and check for 28-volt dc across contacts.	Replace circuit breaker.
(3) Speed brake selector switch inoperative.	Check switch for mechanical operation and electrical continuity.	Replace switch.
(4) Circuit open, shorted, or grounded between circuit breaker and speed brake control valve.	Check for continuity.	Repair as necessary. (Refer to Wiring Diagram Manual.)
(5) Speed brake control valve in-operative.	Cut off hydraulic pressure and actuate speed brake selector switch to extend and retract. Listen for spool being actuated in control valve.	Replace control valve.
B. SPEED BRAKE OPERATION SLOW, EXCESSIVE DROP IN HYDRAULIC PRESSURE, OR HYDRAULIC PRESSURE SLOW TO BUILD UP FOR OPERATION.		
(1) Speed brake control valve stick-ing or has internal leak.	Operate speed brake to an intermediate position and return switch to off. Speed brake control valve is faulty in any of the following conditions: (a) Speed brake does not stop operation immediately	Replace valve.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
	after switch is returned to off, and simulated air load is momentarily applied to lower side of speed brake. (b) Panel does not remain firm with switch at off position. (c) Valve body has hot spot, and hissing sound is heard from valve. (d) Hydraulic pressure does not remain stable after speed brake operation.	

SPEED BRAKE SYSTEM - MAINTENANCE PRACTICES

1. Servicing Speed Brake System

The upper end of each actuating cylinder is equipped with two grease fittings. (This allows interchangeability of actuating cylinders.) Lubricate only the accessible grease fittings. There are also grease fittings on the adjustable rod ends of the hydraulic actuating cylinders and the speed brake hinge points.

2. Adjusting Speed Brake Throttle Actuated Switches

Refer to Chapter 76.

1. Removing Speed Brake

A. To remove speed brake, proceed as follows:

(1) Connect external 28-volt dc electrical power. Apply hydraulic pressure to airplane.

WARNING: CLEAR WHEEL WELL AREA OF EQUIPMENT AND PERSONNEL. DO NOT ENTER SPEED BRAKE AREA WHEN ELECTRICAL AND HYDRAULIC POWER IS APPLIED. ANYONE CAUGHT BY A MOVING SPEED BRAKE PANEL MAY BE SERIOUSLY INJURED OR KILLED.

(2) Engage "SPEED BRAKE - PWR" circuit breaker. Actuate speed brake selector switch to extend. The speed brake panel should open.

(3) Shut down hydraulic pressure. Disconnect external dc electrical power.

(4) Remove access panels on the outside and in between speed brake hinge point. Disconnect electrical bond braid from speed brake hinge arm.

(5) Disconnect actuating cylinders at speed brake panel attach points. The panel and cylinders must be supported when the final attach bolt is removed.

NOTE: Do not allow panel to swing free, because it can damage the fuselage skin at the panel hinge point.

(6) Support panel and remove left and right hinge attaching bolts, cotter pins, nuts, washers, and bushings. Remove panel.

2. Installing Speed Brake

A. To install speed brake, see figure 27-60-401.

3. Removing Speed Brake Actuating Cylinders

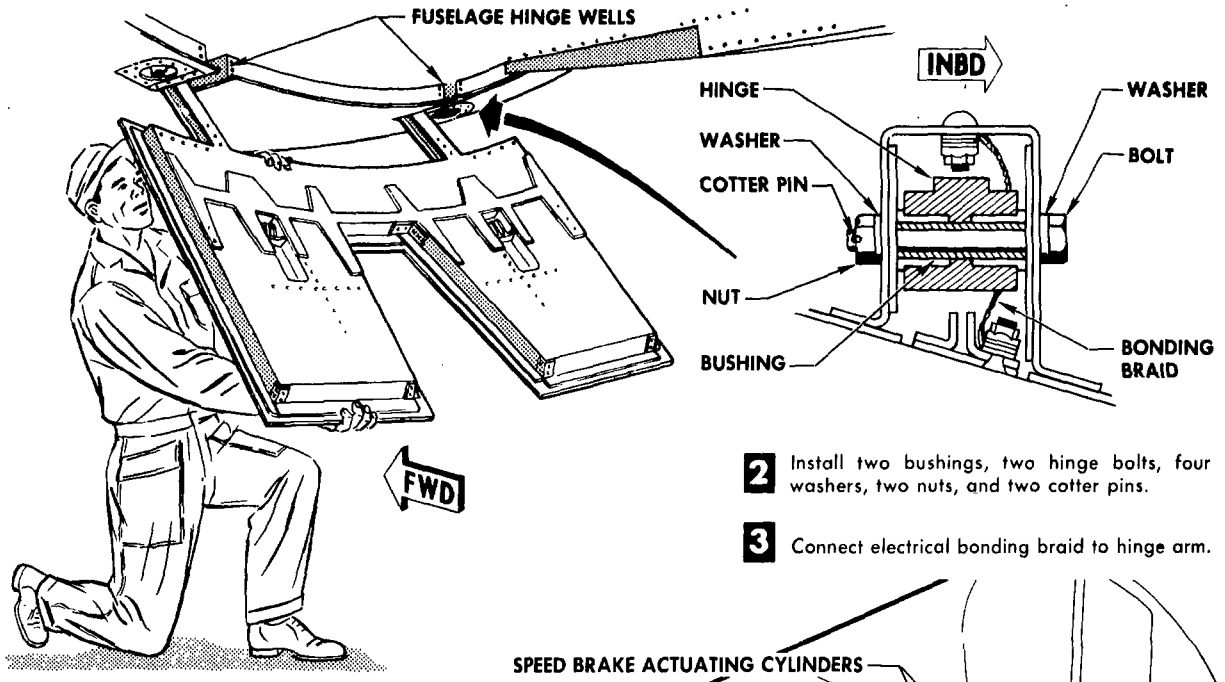
A. To remove speed brake actuating cylinders, proceed as follows:

(1) Connect external 28-volt dc electrical power. Apply hydraulic pressure to airplane.

WARNING: CLEAR WHEEL WELL AREA OF EQUIPMENT AND PERSONNEL. DO NOT ENTER SPEED BRAKE AREA WHEN ELECTRICAL AND HYDRAULIC POWER IS APPLIED. ANYONE CAUGHT BY A MOVING SPEED BRAKE PANEL MAY BE SERIOUSLY INJURED OR KILLED.

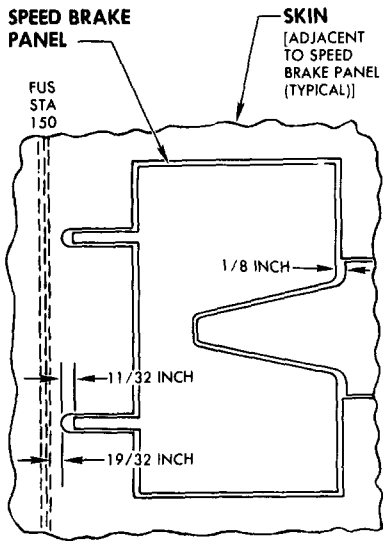
(2) Engage "SPEED BRAKE - PWR" circuit breaker. Actuate speed brake selector switch to extend. The speed brake panel should open.

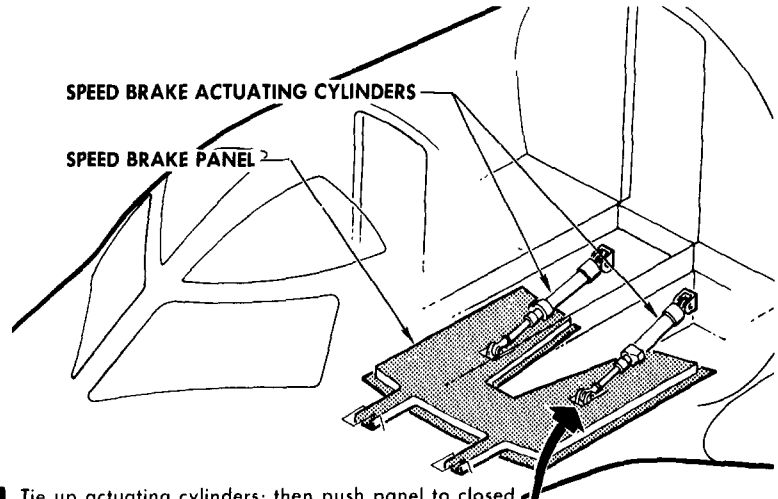
1 Have helper hold panel in position (with its hinge fittings positioned in fuselage hinge wells). Skin on hinge arms may need trimming on replacement panels. (Refer to step 5.)



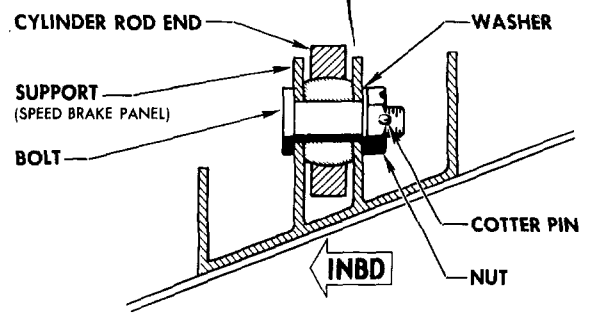
2 Install two bushings, two hinge bolts, four washers, two nuts, and two cotter pins.

3 Connect electrical bonding braid to hinge arm.





4 Tie up actuating cylinders; then push panel to closed position. Check whether panel operates smoothly, and visually check contour of panel, to determine whether it conforms to fuselage mold line.



5 If necessary, trim panel to specified tolerance, as shown.

NOTE Skin gap between panel and adjacent skin is 1/16 inch, except as noted. The tolerance for all dimensions is ±1/32 inch.

6 Adjust speed brake panel actuating cylinders. (Refer to "Adjusting Speed Brake Panel and Actuating Cylinders.") After actuating clylinders are adjusted, make sure rod ends are connected and safetied properly.

5370-2-52-55

Installing Speed Brake
Figure 27-60-401

(3) Shut down hydraulic pressure. Disconnect external dc electrical power.

(4) Disconnect actuating cylinders at speed brake panel attach points. The panel must be supported when the final attach bolt is removed from the actuating cylinders.

NOTE: Do not allow panel to swing free, because it can damage the fuselage skin at the panel hinge point.

(5) Disconnect hydraulic hoses from actuating cylinders.

(6) Plug all hose "B" nuts and cap all fittings.

(7) Disconnect aft end of speed brake actuating cylinder, and remove cylinder.

4. Installing Speed Brake Actuating Cylinders

A. To install speed brake actuating cylinders, proceed as follows:

(1) Attach cylinder to fuselage fitting with a bolt (265-390020), washer (2W1-17-20-32), nut (AN320-4), and cotter pin (MS24665-86).

(2) Remove plugs from hydraulic hoses and caps from fittings; connect hoses to actuating cylinder.

CAUTION: DO NOT BEND HOSES SHARPLY, AS DAMAGE TO TEFLON HOSE LINER MAY RESULT.

- DO NOT ALLOW PANEL TO SWING FREE, BECAUSE IT CAN DAMAGE THE FUSELAGE SKIN AT THE PANEL HINGE POINT.

(3) If only one cylinder is being replaced, disconnect other cylinder rod end and position cylinders so that rod ends clear adjacent structure.

CAUTION: IT IS NECESSARY TO PURGE THE SPEED BRAKE ACTUATING CYLINDERS OF AIR. OPERATION OF THE SPEED BRAKE WITH AIR IN THE CYLINDERS WHICH COULD TWIST AND PERMANENTLY DAMAGE THE SPEED BRAKE PANELS.

(4) Connect external 28-volt dc electrical power to airplane.

WARNING: CLEAR WHEEL WELL AREA OF EQUIPMENT AND PERSONNEL. DO NOT ENTER SPEED BRAKE AREA WHEN ELECTRICAL AND HYDRAULIC POWER IS APPLIED. ANYONE CAUGHT BY A MOVING SPEED BRAKE PANEL MAY BE SERIOUSLY INJURED OR KILLED.

(5) Dump accumulator pressures before applying hydraulic pressure to airplane if hydraulic pressure stand is not being used.

NOTE: By dumping accumulator pressures and then starting hydraulic pump, you can operate the speed brake cylinders slowly by placing the speed brake switch alternately on (IN or OUT) and off (center) position to maintain a low hydraulic pressure (1200 psi maximum). A hydraulic test stand may be used in the following procedures.

(6) Apply hydraulic pressure to airplane.

(7) Actuate speed brake switch (on left throttle) to OUT and IN several times to purge speed brake cylinders of any trapped air.

(8) After cylinders are purged, actuate speed brake switch to OUT. When cylinders are fully extended place switch at off (center) position.

(9) Attach cylinder rod ends to speed brake panel.

NOTE: Subsequent steps of this procedure will require adjustment of the speed brake actuator rod ends. The easiest way to adjust the rod ends is to rotate the piston rod by using a wrench on the flats provided on the lower end of the piston rod. To shorten over-all length of actuator assembly, rotate piston rod counterclockwise, as viewed from forward (or lower) end of actuator. To lengthen assembly, rotate rod clockwise. The rod end jam nut and lock washer may be left loose and backed off as far as possible until adjustment is complete. Be sure that speed brake switch is in off (center) position, to prevent pressure in cylinder from making rod rotation difficult, when adjustment is being made.

(10) Shorten actuator assemblies as much as possible. (Rotate piston rods counterclockwise until rod ends are screwed all the way into piston rods).

CAUTION: PRESSURE MUST NOT EXCEED 1200 PSI DURING THIS PHASE OF ADJUSTMENT. PRESSURES EXCEEDING 1200 PSI WILL TEND TO PULL THE DOOR INWARD TOO MUCH AND MAY CAUSE PERMANENT DISTORTION OF THE SPEED BRAKE PANEL.

(11) Verify that pressure is 1200 psi maximum (see note following step 5). Stand clear of speed brake area; then actuate speed brake switch to IN.

(12) Measure, and record for future reference, vertical gaps between speed brake panel inner skin and land surface on adjacent fuselage structure. Gaps of less than 1/32 inch may be considered zero. Gaps 1/32 inch and greater should be recorded. (It is suggested that gaps be noted with grease pencil on the fuselage skin adjacent to the measured points).

(13) Actuate speed brake switch to OUT, then off.

WARNING: CLEAR WHEEL WELL AREA OF EQUIPMENT AND PERSONNEL. DO NOT ENTER SPEED BRAKE AREA WHEN ELECTRICAL AND HYDRAULIC POWER IS APPLIED. ANYONE CAUGHT BY A MOVING SPEED BRAKE PANEL MAY BE SERIOUSLY INJURED OR KILLED.

(14) Lengthen each actuator (rotate piston rods clockwise) six full turns.

(15) Stand clear of speed brake; then actuate speed brake switch to IN. (The actuator pistons will bottom out, but the panel will not be fully closed).

(16) Measure and record vertical gap between trailing edge of speed brake panel and adjacent fuselage skin in line with actuator attach points. (See figure 27-60-402).

(17) Actuate speed brake switch to OUT, then off.

(18) Adjust actuator lengths (rotate piston rods as necessary) to get a dimension of 3/4 (±1/8) inch between trailing edge of speed brake panel and adjacent fuselage skin in line with actuator attach points (when speed brake actuators are bottomed in retracted position as in step 15).

WARNING: CLEAR WHEEL WELL AREA OF EQUIPMENT AND PERSONNEL. DO NOT ENTER SPEED BRAKE AREA WHEN ELECTRICAL AND HYDRAULIC POWER IS APPLIED. ANYONE CAUGHT BY A MOVING SPEED BRAKE PANEL MAY BE SERIOUSLY INJURED OR KILLED.

NOTE: One-half turn of the piston rod will cause a 7/32 inch deflection of the speed brake panel trailing edge.

(19) Stand clear of speed brake area; then actuate speed brake switch to IN.

(20) Check vertical gap and continue adjustment until measurement is 3/4 (±1/8) inch at both places.

(21) Actuate speed brake switch to OUT: then off (center position).

(22) Shorten each actuator (rotate piston rod counterclockwise) 1-1/2 turns.

(23) Stand clear of speed brake; then actuate speed brake switch to IN.

(24) Compare gaps now existing with gaps recorded in step (12). The gaps should be the same. If the gaps are now greater than they were in step (12), an additional one-half or full turn of the piston may be required.

(25) Increase hydraulic pressure to 3000 psi and recheck gaps. The gaps should remain the same as they were recorded in step (12). (This ensures that both actuators are bottomed out and are not pulling excessive preload on the speed brake).

CAUTION: THE SPEED BRAKE WILL BE DEFORMED IF THE ACTUATORS ARE NOT BOTTOMED WHEN THE SPEED BRAKE CONTACTS THE FUSELAGE.

(26) Dump hydraulic pressure.

(27) On piston rods not having a safety inspection hole, verify that dimension "A", between rod end bolt hole and end of piston rod, is a maximum of 1.718 inches. (See figure 27-60-402). On piston rods that have a safety inspection hole, check that rod end threads are visible in inspection hole.

(28) On each actuator, align lock washer on piston rod and tighten rod end jam nut. Safety jam nut to lock washer with lockwire.

(29) Adjust hydraulic pressure to 3000 psi.

(30) Stand clear of speed brake; then actuate speed brake switch to IN.

(31) Check and mark places on speed brake panel where panel can be crimped to improve its fairing with fuselage. Open speed brake and crimp edges where necessary. (See figure 27-60-402 for crimp dimensions and limits).

(32) Leak-check speed brake actuating cylinders, fittings, and hydraulic lines. Shut down hydraulic pressure and remove external dc electrical power from airplane.

5. Removing Speed Brake Limit Switch

A. The speed brake limit switch is on the speed brake well structure, just above the aft left corner of the speed brake, at fuselage station 180. The speed brake must be lowered before the switch is accessible. To remove speed brake limit switch, proceed as follows:

(1) Connect external 28-volt dc electrical power. Apply hydraulic pressure to airplane.

WARNING: CLEAR WHEEL WELL AREA OF EQUIPMENT AND PERSONNEL. DO NOT ENTER SPEED BRAKE AREA WHEN ELECTRICAL AND HYDRAULIC POWER IS APPLIED. ANYONE CAUGHT BY A MOVING SPEED BRAKE PANEL MAY BE SERIOUSLY INJURED OR KILLED.

(2) Engage "SPEED BRAKE - PWR" circuit breaker. Actuate speed brake selector switch to extend. The speed brake panel should open.

(3) Shut down hydraulic pressure. Disconnect external dc electrical power, and pull "SPEED BRAKE - IND" circuit breaker.

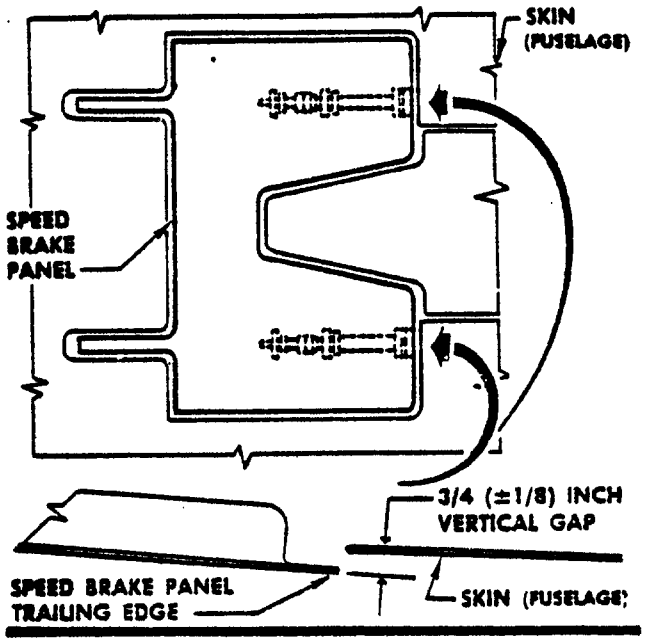
(4) Remove the two mounting screws, two nuts, and two washers from switch.

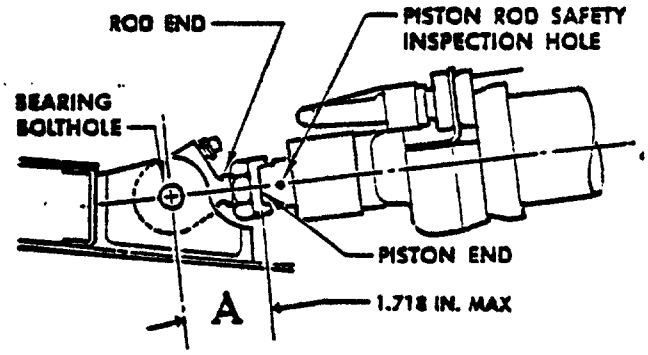
(5) Disconnect and mark wires for proper reinstallation.

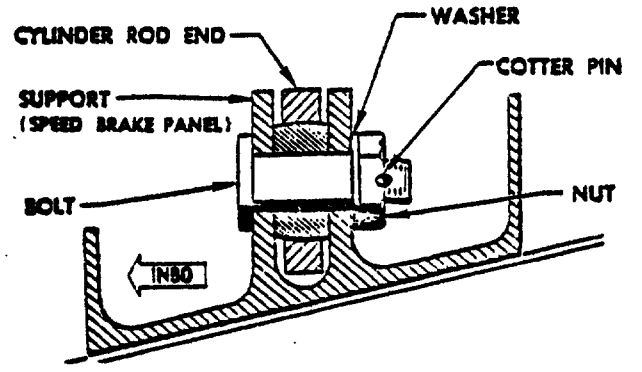
(6) Remove switch.

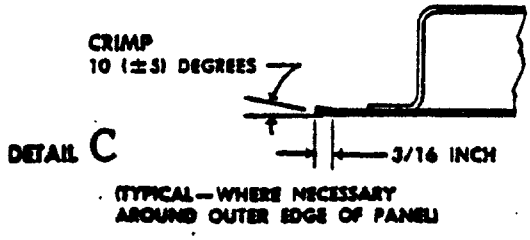
6. Installing Speed Brake Limit Switch

For procedures to install the speed brake limit switch, refer to adjusting procedures for the speed brake limit switch.









ORIGINAL
As Received By
ATP

Adjusting Speed Brake Panel Actuating Cylinders
Figure 27-60-402

1. Adjusting Speed Brake and Actuating Cylinders

A. To adjust speed brake and actuating cylinders, proceed as follows:

(1) Connect external 28-volt dc electrical power. Apply hydraulic pressure to airplane.

WARNING: CLEAR WHEEL WELL AREA OF EQUIPMENT AND PERSONNEL. DO NOT ENTER SPEED BRAKE AREA WHEN ELECTRICAL AND HYDRAULIC POWER IS APPLIED. ANYONE CAUGHT BY A MOVING SPEED BRAKE PANEL MAY BE SERIOUSLY INJURED OR KILLED.

(2) Actuate speed brake selector switch to extend, then off. To fit panel, proceed as follows:

(a) Perform steps (10) through (32) in "Installing Speed Brake Actuating Cylinders" in this chapter.

2. Adjusting Speed Brake Limit Switch

A. To adjust speed brake limit switch, proceed as follows:

(1) Connect wires to proper location on limit switch.

(2) Install switch to structure with two screws, two washers, and two nuts. (Do not tighten screws.)

(3) Place a straightedge across bottom of fuselage skin and measure distance between bottom radius of switch actuating arm roller and top of straightedge. The distance should be 1-1/8 inches.

(4) To obtain proper distance, move switch in elongated slot of bracket. Tighten attach screws when proper setting is obtained.

(5) Connect external 28-volt dc electrical power. Apply hydraulic pressure to airplane.

WARNING: CLEAR WHEEL WELL AREA OF EQUIPMENT AND PERSONNEL. DO NOT ENTER SPEED BRAKE AREA WHEN ELECTRICAL AND HYDRAULIC POWER IS APPLIED. ANYONE CAUGHT BY A MOVING SPEED BRAKE PANEL MAY BE SERIOUSLY INJURED OR KILLED.

(6) Make sure "SPEED BRAKE - PWR" and "SPEED BRAKE - IND" circuit breakers are engaged on the pilot's left console circuit-breaker panel. Make sure that both throttles are in cutoff position.

(7) Place speed brake selector switch at retract (forward). Allow speed brake to completely close; then place selector switch at off (center).

(8) Push left throttle forward to extreme end of travel. Place speed brake selector switch at extend. The "SPEED BRAKE OPEN" caution light and both master caution lights should illuminate as soon as door is free from fuselage.

(9) Place selector switch at retract, and close speed brake door. The lights should go out.

(10) Pull left throttle back to cutoff and push right throttle forward to extreme end of travel. Repeat steps (8) and (9).

(11) Pull right throttle back to cutoff.

(12) Place selector switch at extend, and extend speed brake. When speed brake reaches full open, place selector switch at off. Shut down hydraulic pressure and disconnect external dc electrical power source.

NOTE: An operational check-out of the speed brake system must be made if a unit has been replaced, if hydraulic lines have been disconnected, or if an adjustment has been made on any unit. The check-out or checking procedure must be thorough enough to make sure the system is free from hydraulic leaks and that no trouble could occur because of improper maintenance.

1. Operational Check-out of Speed Brake System

WARNING: MAKE SURE ALL PERSONNEL ARE CLEAR OF AREA BEFORE OPERATING SPEED BRAKE.

A. The following special tools and equipment are needed to make this check:

- External 28-volt dc electrical power source
- Stop watch or equivalent

NOTE: Before this check is made, the speed brake system must be completely rigged and the hydraulic system must be properly serviced and in operating condition. Make sure access doors are installed on airplane before operational check-out of speed brake. If steps do not meet required specifications, refer to trouble shooting procedures.

(1) Remove ground locks from speed brake panel actuating cylinders.

(2) Connect external 28-volt dc electrical power to airplane.

(3) Apply hydraulic pressure to airplane.

(4) Make sure "SPEED BRAKE - PWR" and "SPEED BRAKE - IND" circuit breakers are engaged on the pilot's left console circuit-breaker panel.

(5) Make sure "HYD PRESSURE - AUX" circuit breaker (on left overhead forward circuit-breaker panel) is engaged.

(6) Place "HYDRAULIC - AUX SYSTEM" toggle switch (on hydraulic overhead panel) at OFF.

(7) Place speed brake selector switch at OUT (extend). The speed brake should extend in 2 seconds or less.

(8) Place speed brake selector switch at off (center). The speed brake should maintain its extended position.

(9) Place speed brake selector switch at IN (retract). The speed brake should close in 4 seconds or less.

(10) Place speed brake selector switch at off. The speed brake should maintain its closed position.

(11) Place speed brake selector switch alternately at OUT (extend), then off, before speed brake door has completed its cycle. The speed brake should maintain its position and should not creep.

(12) Place speed brake selector switch alternately at IN, then off, before speed brake door has completed its cycle. The speed brake should maintain its position and should not creep.

(13) Push left throttle forward to extreme end of travel. Place speed brake selector switch at OUT (extend). The "SPEED BRAKES OPEN" caution light and both master caution lights should illuminate as soon as the door is free from the fuselage. If master caution lights do not illuminate, refer to Chapter 33.

(14) Place speed brake selector switch at IN (retract), and close speed speed brake door. The lights should go out.

(15) Pull left throttle back to cutoff and push right throttle forward to extreme end of travel. Repeat steps (13) and (14).

(16) Pull right throttle back to cutoff.

(17) Shut down hydraulic pressure.

(18) With normal accumulator dump valve, dump hydraulic pressure from normal system accumulator.

(19) Place auxiliary power switch at ON.

(20) Place speed brake selector switch at OUT (extend). When speed brake reaches full open, place selector switch at off. If auxiliary system fails to open speed brake refer to Chapter 29.

(21) Place auxiliary power switch at OFF.

(22) Engage "SPEED BRAKE DUMP" circuit breaker.

(23) With speed brake selector switch at OUT (extend) and/or IN (retract), place emergency speed brake dump switch at to DUMP. The speed brake dump valve should not operate in either the extend or the retract position.

(24) Place speed brake selector switch at off.

(25) Place emergency speed brake dump switch at DUMP. The speed brake dump valve should operate. Verify that hydraulic pressure has been dumped by manually moving speed brake panel partially up and down.

(26) Apply hydraulic pressure to airpalne.

INSPECTION/CHECK (Continued)

Operational Check-out of Speed Brake System (Cont'd)

27. Operate speed brake several times to ensure that dump valve is in normal position.
28. Place speed brake selector switch at OUT (extend). When speed brake reaches full open, place selector switch at off. Shut down hydraulic pressure.
29. Disconnect external DC electrical power.
30. Return all switches in cockpit to normal position.
31. Install ground locks on speed brake actuating cylinders.

INSPECT — SPEED BRAKE INNER SKIN AND DOUBLER

Equipment/Tools

- None

Consumables

- As required per Sabreliner Corrosion Control Manual (SR-76-23), chapter 20.

Inspection

WARNING

Clear speed brake area of equipment and personnel. Do not enter speed brake area when electrical and hydraulic power is applied, or ground safety locks not installed. Anyone struck by a moving speed brake panel can be seriously or fatally injured.

1. Lower speed brake and secure, taking the following precautions:
   A. Remove all electrical power.
   B. Deplete hydraulic system pressure.
   C. Install ground safety locks on speed brake.
2. Clean speed brake (shaded area, see Figure 27-60-601) per Sabreliner Corrosion Control Manual (SR-76-23), chapter 20.
3. Visually inspect inner skin and doubler for loose or missing fasteners, cracks and/or corrosion.

NOTE

Any repairs accomplished should be made in accordance with Sabreliner Structural Repair Manual (NA-66-1032) and/or Sabreliner Corrosion Control Manual (SR-76-23)

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WARNING

Clear speed brake area of equipment and personnel. Do not enter speed brake area when electrical and hydraulic power is applied, or ground safety locks not installed. Anyone struck by a moving speed brake panel can be seriously or fatally injured.

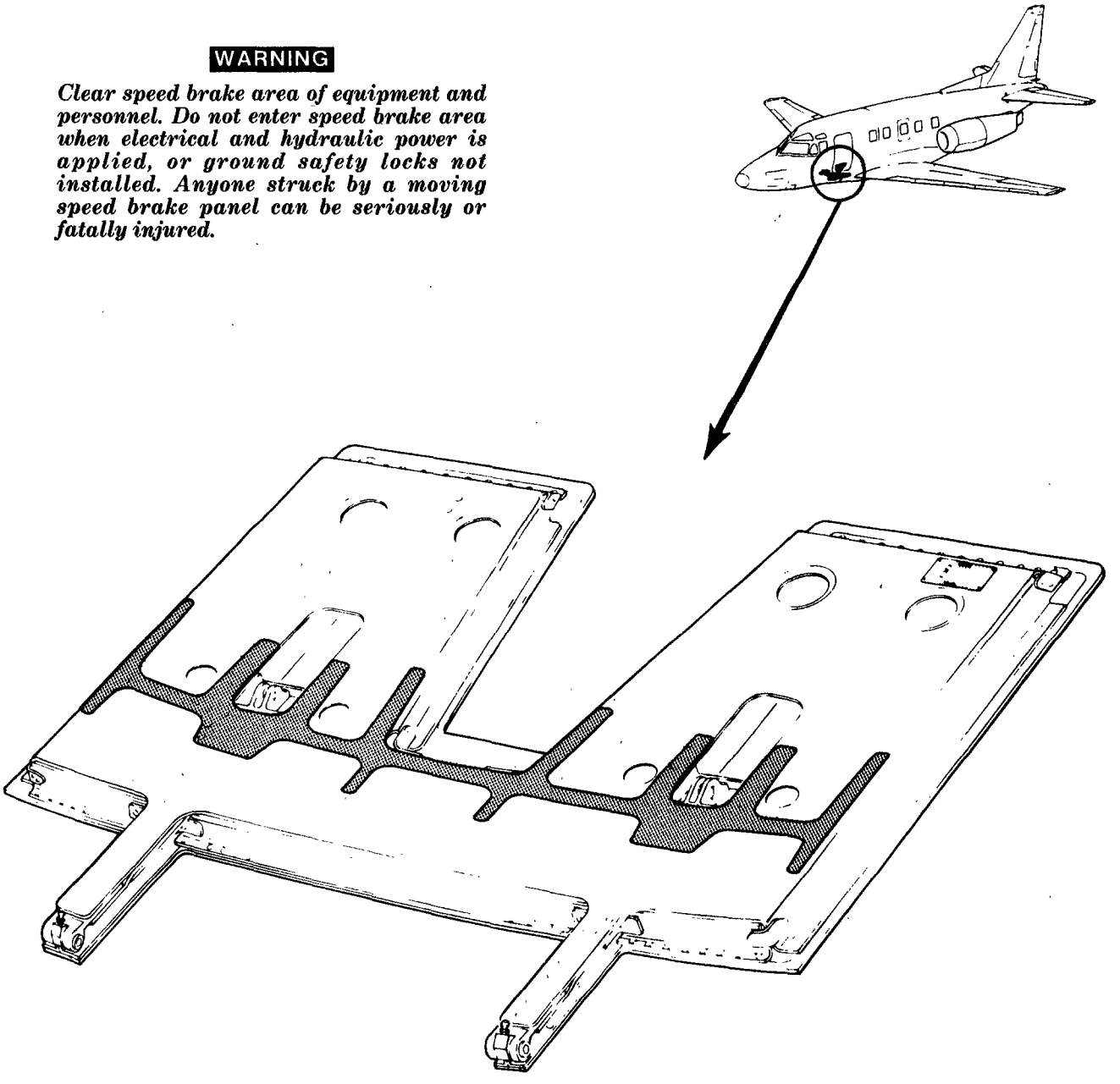

Figure 27-60-601 — Inspect — Speed Brake Inner Skin and Doubler

NEW

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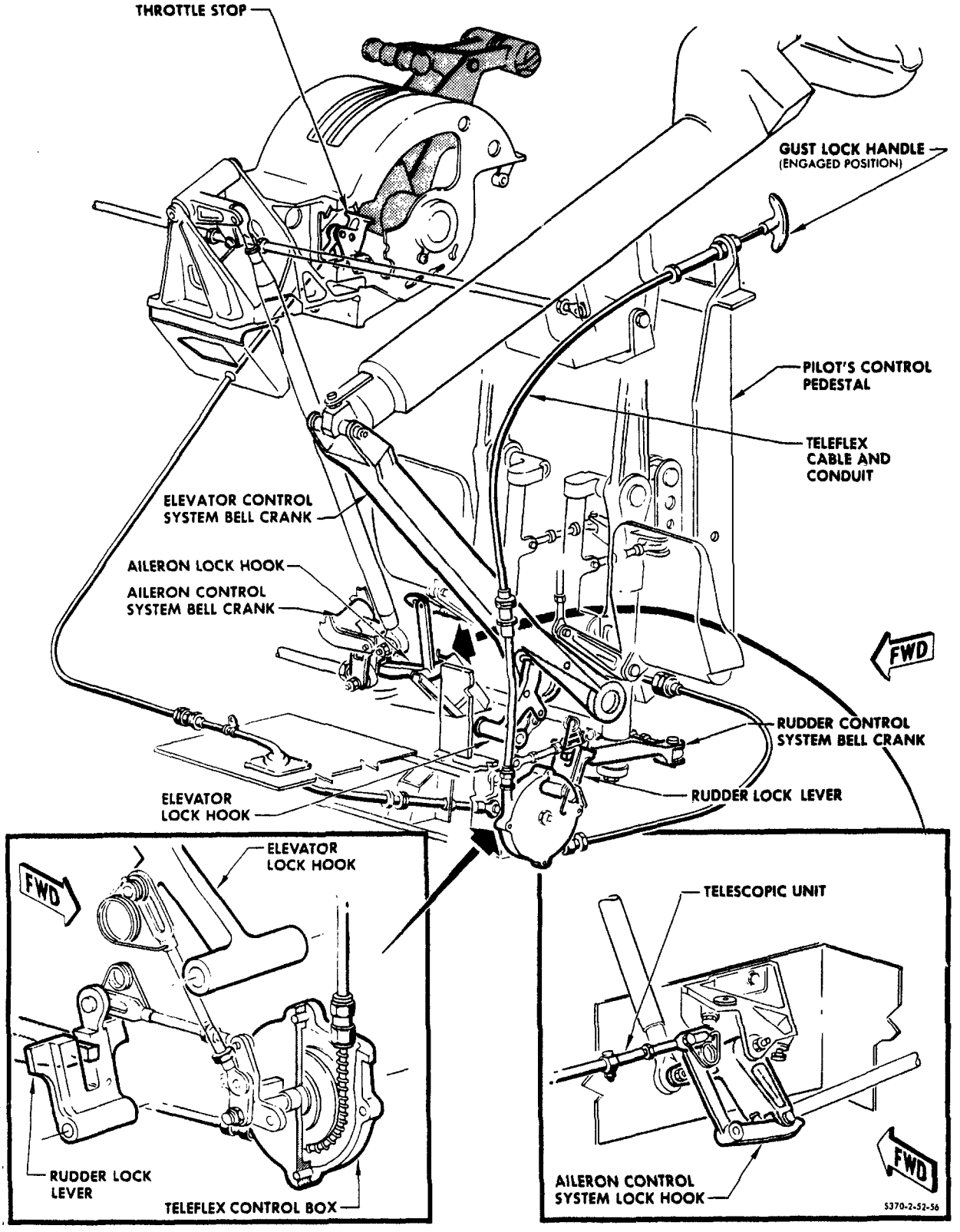
GUST LOCK - DESCRIPTION AND OPERATION

1. Gust Lock System (See figure 27-70-1.)

The gust lock system provides a means of locking the aileron control system, rudder control system, and elevator control system in their near-neutral positions while the airplane is on the ground. The gust lock system consists of three locking devices mechanically actuated, through a push-pull Teleflex cable system, by a handle on the left side of the pilot's control pedestal. The gust lock is engaged by pulling the gust lock handle down (to lock) and aft to the engaged and locked position. Pulling the handle spring-loads the hooks toward their locked positions. Movement of the control systems through neutral engages the locks to the bell cranks of each control system. The handle also operates a throttle stop that prevents the throttle from being advanced while the gust locks are engaged, and prevents the gust locks from being engaged when the throttle is past the idle position. When the handle is pushed down (to unlock) and forward to the disengaged position, the throttle stop and surface control lock hooks are pulled from their engaged positions. The surface controls and throttles are then unlocked and free to move.



Gust Lock System
Figure 27-70-1

GUST LOCK - TROUBLE SHOOTING

1. Trouble Shooting Gust Lock System

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. HANDLE CANNOT BE PULLED TO ENGAGED POSITION.		
(1) Throttles improperly positioned.	Check that throttles are at OFF.	
(2) Mechanical obstruction in linkage.	Check linkage to determine cause of mechanical interference.	
(3) Handle locked in stowed position.		Unlock handle by pushing down, then pulling aft, on handle.
B. HANDLE IN ENGAGED POSITION BUT CONTROLS DO NOT LOCK.		
(1) Telescopic units, rods, hooks, springs, or bell crank disconnected.	Check linkage to see that it is not disconnected at any point.	
(2) Cable broken.	While moving handle back and forth, check movement of linkage. If linkage does not move, cable is disconnected or broken.	Replace or connect cable.
(3) Improperly rigged gust lock system.	While moving controls, check hooks. If hooks do not lock, linkage is probably misrigged.	Rerig gust lock system. (See figure 27-70-201.)
C. HANDLE IN DISENGAGED POSITION BUT CONTROLS DO NOT UNLOCK.		
(1) Improperly rigged gust lock system.	While moving handle back and forth, check movement of linkage. If hooks move but do not unlock, linkage is probably misrigged.	Rerig gust lock system. (See figure 27-70-201.)
(2) Telescopic units, rods, hooks, springs, or bell crank disconnected.	Check linkage to see that it is not disconnected at any point.	
(3) Cable broken or disconnected.	While moving handle back and forth, check movement of linkage. If linkage does not move, cable is disconnected or broken.	Replace or connect cable. CAUTION: BE SURE THAT GUST LOCK HANDLE MOUNTING NUTS AND TELEFLEX "B" NUTS ARE SAFETIED WITH LOCKWIRE.

GUST LOCK - MAINTENANCE PRACTICES

1. Rigging Gust Lock System (See figure 27-70-201.)

2. Checking Gust Lock System

A. Before check-out, the gust lock must be properly rigged. (See figure 27-70-201.) If check-out does not meet requirements, refer to trouble shooting procedures.

(1) Place throttle at OFF.

(2) Pull gust lock handle down (to unlock) and aft to engaged position. The throttles should not advance beyond the gust lock stop (23-1/2 degree travel position). The aileron controls, rudder controls, and elevator controls should lock in rig position.

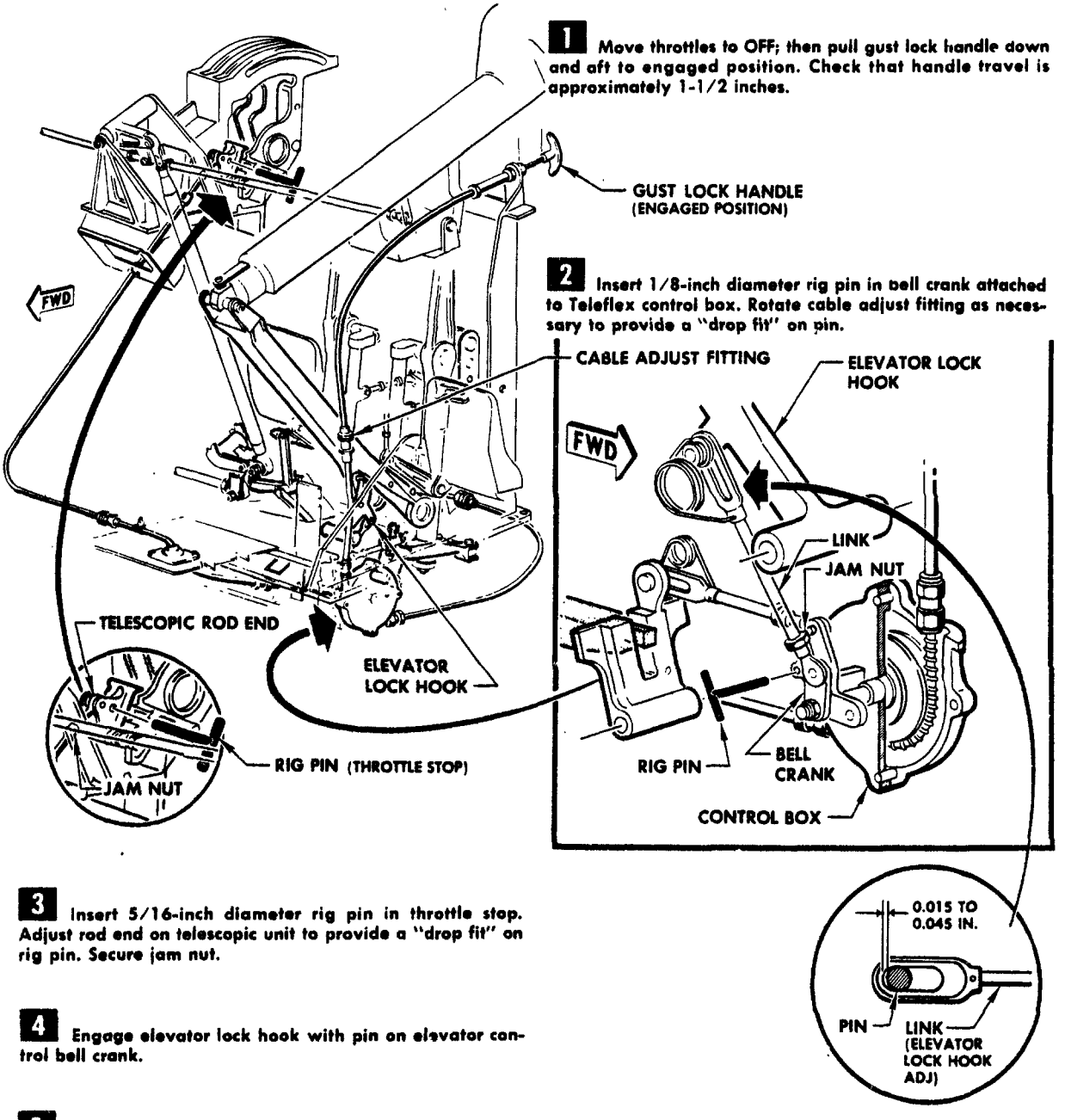
NOTE: The 23-1/2 degree position may be determined by measuring 2-7/8 (±1/16) inches from the aft face of the throttle lever slot to the aft face of the throttle lever along the quadrant upper surface.

(3) Push gust lock handle down (to unlock) and forward to disengaged position: The throttle controls, aileron controls, rudder controls, and elevator controls should operate through complete travel requirements.

(4) Position throttle 5/16 inch forward of 23-1/2 degree travel position. Pull aft on gust lock handle. The gust lock handle should not engage the surface control locks.

(5) Place throttles at OFF.

NOTE Before the gust lock system is rigged, the aileron, rudder, elevator, and engine control systems must be properly rigged.


1 Move throttles to OFF; then pull gust lock handle down and aft to engaged position. Check that handle travel is approximately 1-1/2 inches.
GUST LOCK HANDLE (ENGAGED POSITION)
FWD
2 Insert 1/8-inch diameter rig pin in bell crank attached to Teleflex control box. Rotate cable adjust fitting as necessary to provide a "drop fit" on pin.
CABLE ADJUST FITTING
ELEVATOR LOCK HOOK
FWD
LINK
JAM NUT
TELESCOPIC ROD END
ELEVATOR LOCK HOOK
RIG PIN (THROTTLE STOP)
JAM NUT
RIG PIN
BELL CRANK
CONTROL BOX
0.015 TO 0.045 IN.
PIN
LINK (ELEVATOR LOCK HOOK ADJ)

3 Insert 5/16-inch diameter rig pin in throttle stop. Adjust rod end on telescopic unit to provide a "drop fit" on rig pin. Secure jam nut.

4 Engage elevator lock hook with pin on elevator control bell crank.

5 Adjust elevator lock hook link so that pin in slotted end of link is 0.015 to 0.045 inch from end of slot. Secure jam nut on link.

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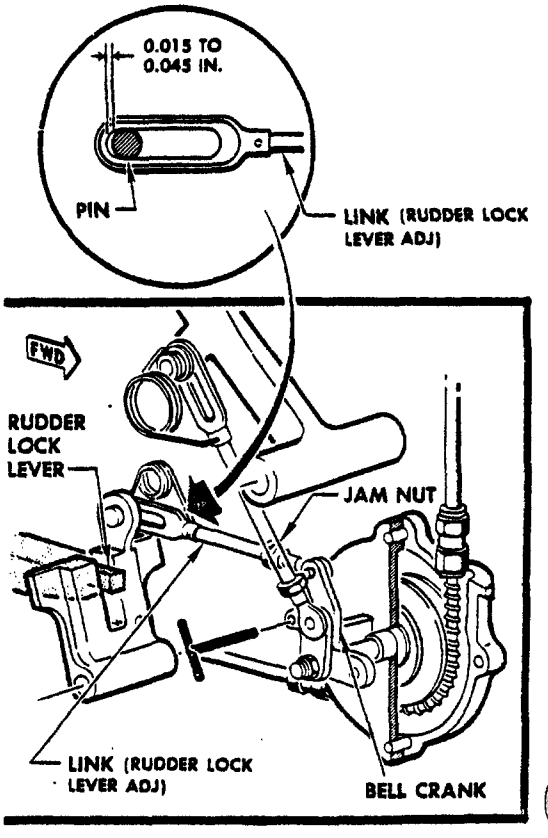
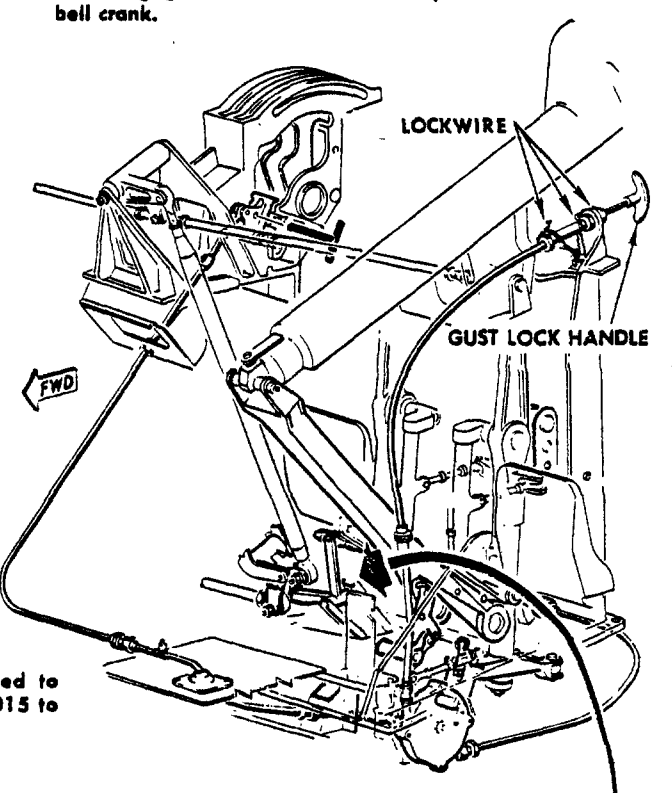
Rigging Gust Lock System
Figure 27-70-201 (Sheet 1 of 2)



6 Engage rudder lock lever with pin on bell crank at lower end of pilot's rudder torque tube.

7 Adjust rudder lock lever link so that pin in slotted end of link is 0.015 to 0.045 inch from end of slot. Secure jam nut on link.

8 Engage aileron lock hook with pin on aileron control bell crank.



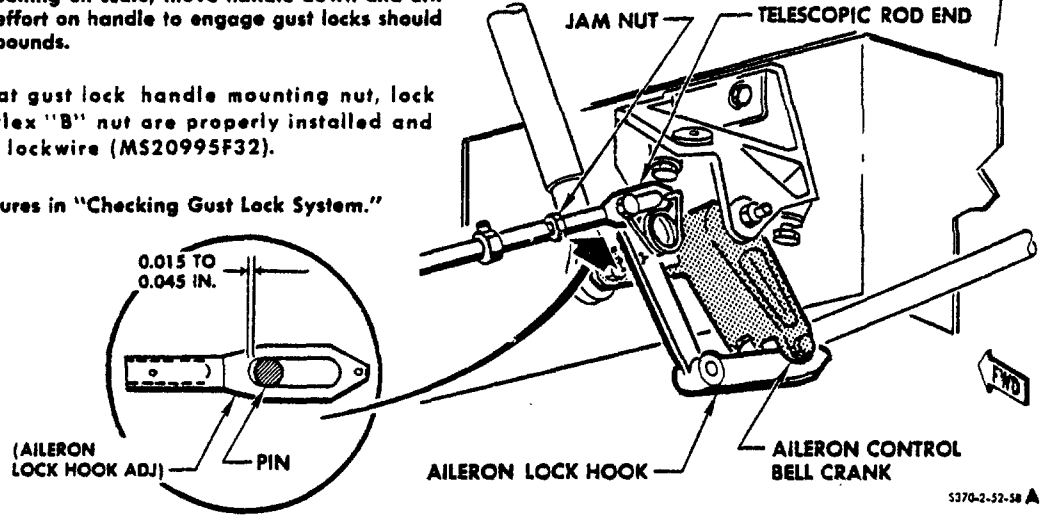
9 Adjust end fitting on telescopic unit attached to aileron lock hook so that pin in slotted fitting is 0.015 to 0.045 inch from end of slot.

10 Remove all rig pins.

11 Move gust lock handle down and forward to disengaged position. Check force required to engage gust locks as follows: By pulling on scale, move handle down and aft. The maximum effort on handle to engage gust locks should not exceed 35 pounds.

12 Check that gust lock handle mounting nut, lock nut, and teleflex "B" nut are properly installed and safetied with lockwire (MS20995F32).

13 Do procedures in "Checking Gust Lock System."



Rigging Gust Lock System
Figure 27-70-201 (Sheet 2 of 2)

CHAPTER

28

FUEL

CHAPTER 28

FUEL SYSTEMS

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FUEL SYSTEM

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FUEL SYSTEM - DESCRIPTION AND OPERATION

1. General

The fuel system consists of integral wing fuel tanks, a fuselage tank, a fuel control system, a fuel flowmeter system, a fuel quantity gaging system, a fuel vent system, a single-point refueling system, a fuel low-pressure warning system, and on NA-370-2 through -6, -8, and -9 airplanes a fuel jettison system. The fuel system is divided into a left and a right system. (See figures 28-00-1, 28-00-2 or 28-00-2A.) Each system supplies fuel boost pressure to one engine from fuel carried in the integral wing tank of each wing half (outer panel and center compartments). Refueling is accomplished by single-point refueling through a receptacle in the right wing inboard leading edge. Fuel is supplied from each fuel system to the engine-driven fuel pumps by two boost pumps, posted on the rear spar in the right and left wing center sections. A fuel cross-feed system provides a means of feeding both engines from a single booster pump pressure source or for balancing the fuel load in the wing tanks. The airplane has two independent fuel quantity indicating systems, one for each wing, with half of the fuselage tank fuel included on each indicator. Two fuel flow indicators show the rate of fuel flow from each engine fuel control unit in pounds per hour. On NA-370-2 through -6, -8, and -9 airplanes, the fuel jettison system provides a means of reducing the airplane gross weight to the design landing weight in a minimum amount of time.

2. Fuel System Components and Locations

COMPONENT	NO. PER AIRPLANE	LOCATION
Gravity refueling filler access	3	One at outboard end of each wing, and one at right fuselage above right engine pylon
Single-point refueling receptacle	1	Bottom inboard leading edge of right wing
Fuel transfer ejectors	2	One in each forward inboard corner of right and left wing center section
Booster pump	2	One attached to rear spar of left wing center section, and one attached to rear spar of right wing center section
Fuel shutoff valves	2	Forward of main wheel wells, on top of center wing section
Tank cross-feed valve	1	Mounted on rib that separates right and left boost pump sump areas
Boost pump cross-feed valve	1	Forward of main wheel wells, on top of center wing section

ORIGINAL
As Received By
ATP

COMPONENT	NO. PER AIRPLANE	LOCATION
Fuel jettison valves*	2	Left valve mounted forward of bulkhead at fuselage station 316; right valve mounted aft of same bulkhead
Jettison fuel level float switches*	2	One in bottom of left and one in bottom of right center wing section
Fuselage tank refueling level control valve	1	Inside of fuselage tank, mounted on aft wall
Refueling test and control valve	1	Flush-mounted inside right wing inboard leading edge, on lower surface
Wing refueling level control pilot and vent valve	2	Right and left wing tip ribs
Wing refueling shutoff valves	2	Right and left wing center sections
Main vent outlet	1	Protrudes from tail cone
Boost pump pressure switches	2	Right and left pressure switches downstream of right and left fuel shutoff valves
Differential pressure switch	2	One aft of each engine fire seal
Fuel flow indicators	2	Pilot's inboard instrument panel
Fuel flow transmitters	2	Forward left side of both engines
Fuel quantity indicators	2	Pilot's inboard instrument panel
Fuel quantity probe	15	Seven in each wing, and one in fuselage tank
Fuel Heaters†	2	Mounted in pylon

*On NA-370-2 thru -6, -8 and -9
†On NA-370-1 and -7 and NA-380-1 and later airplanes

ORIGINAL
As Received By
ATP

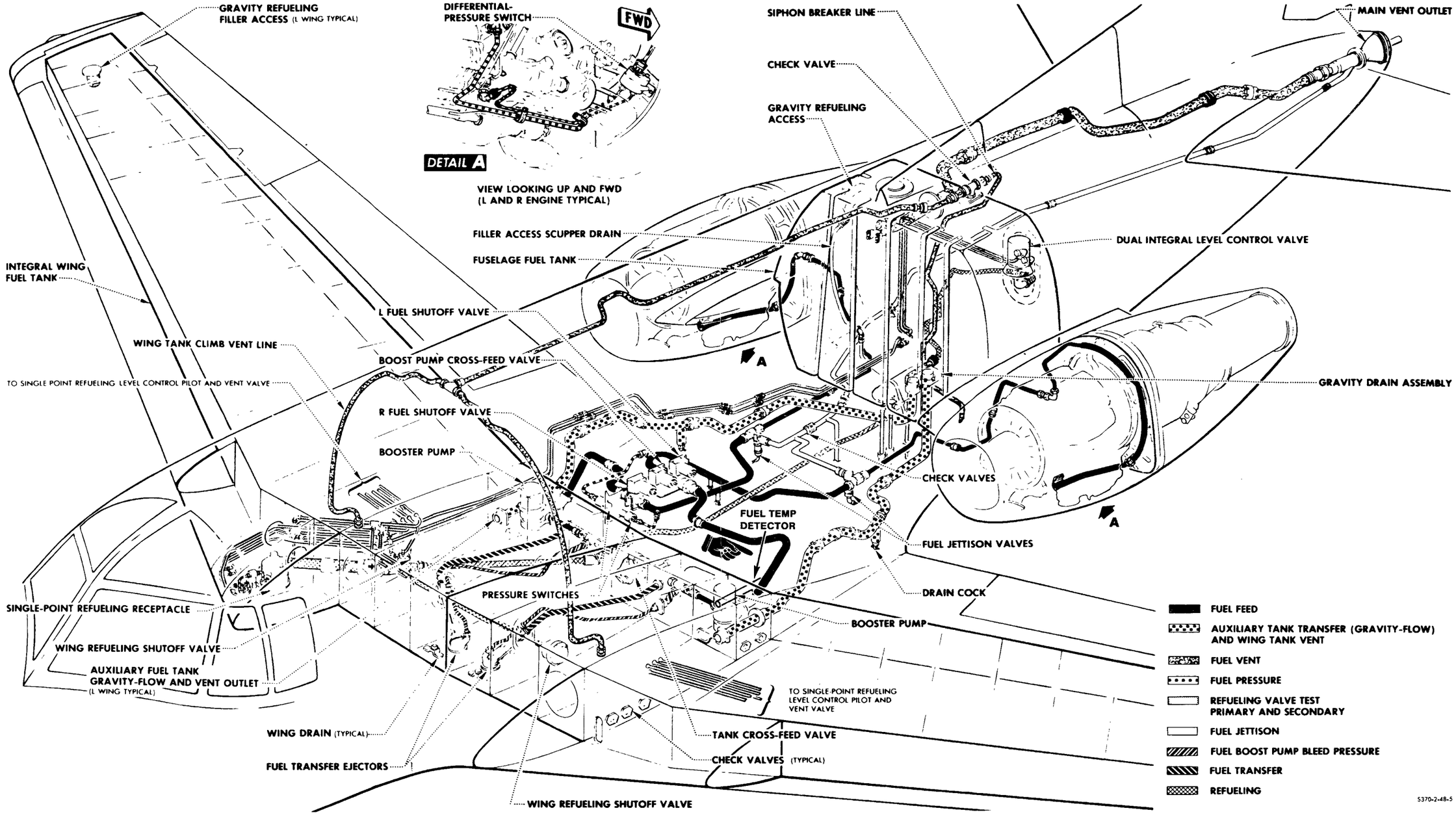

Figure 28-00-1 — Fuel Supply System

S/N 370-2 thru -6, -8 and -9

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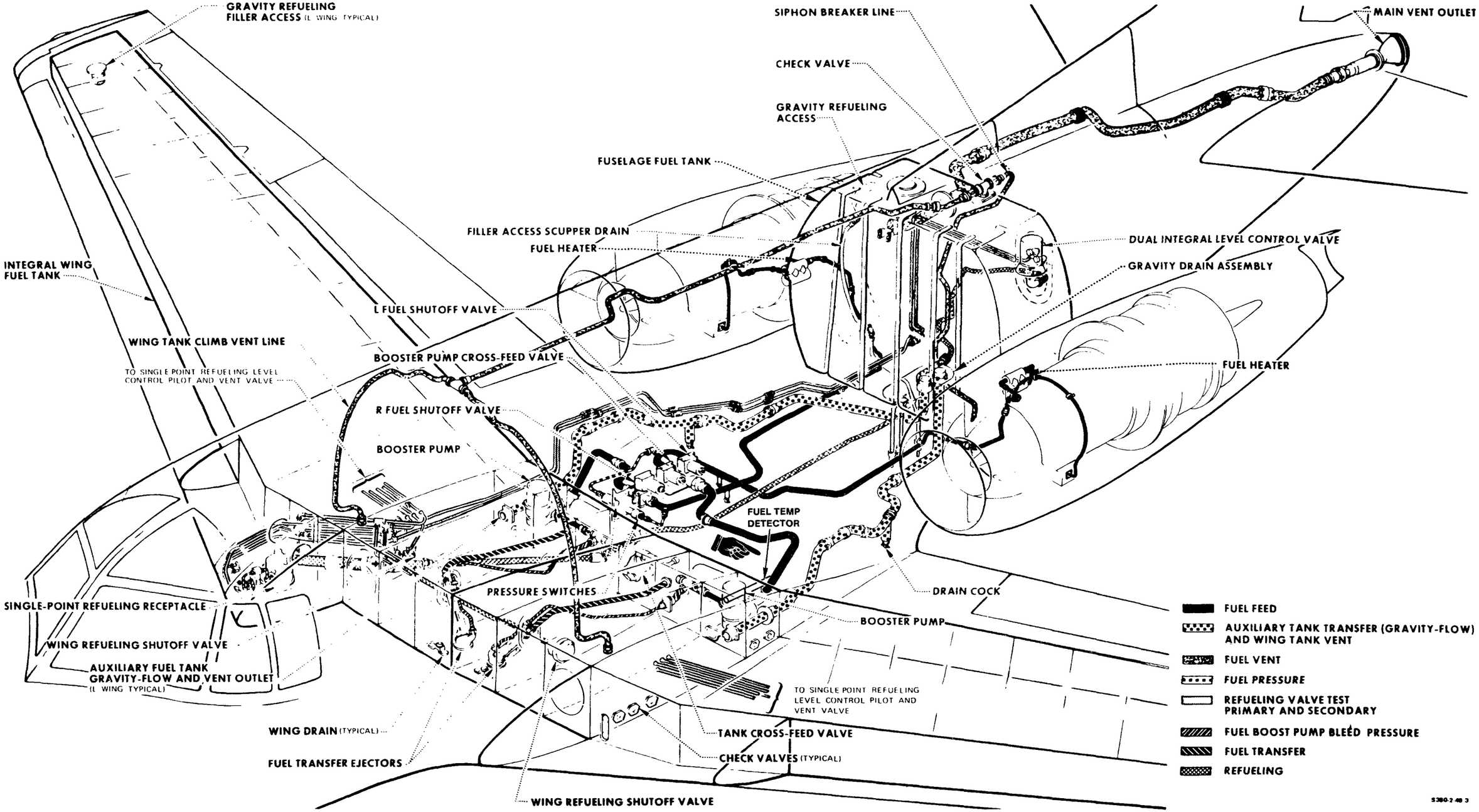

Figure 28-00-1A — Fuel Supply System

S/N 370-1 and -7, S/N 380-1 and Subs

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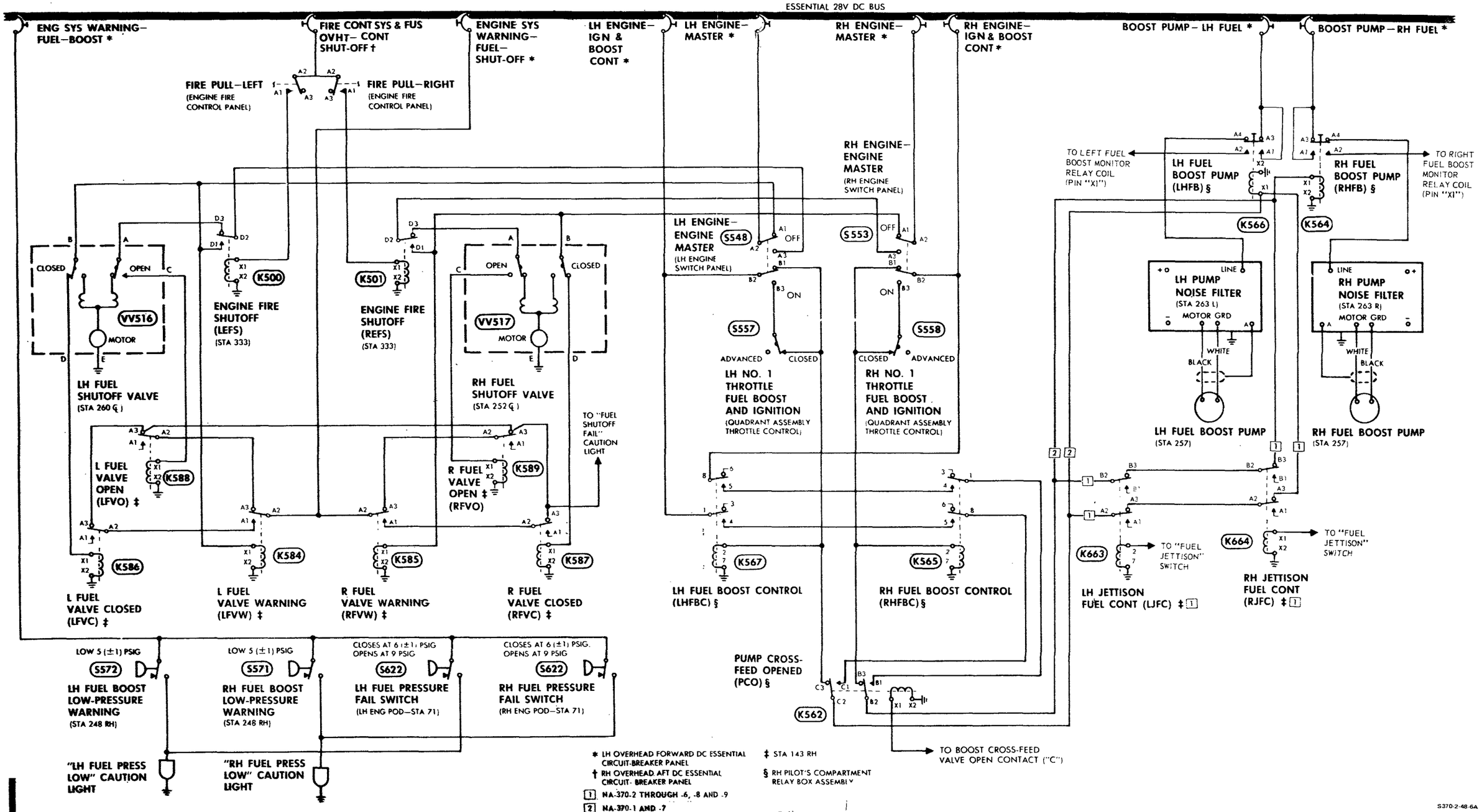
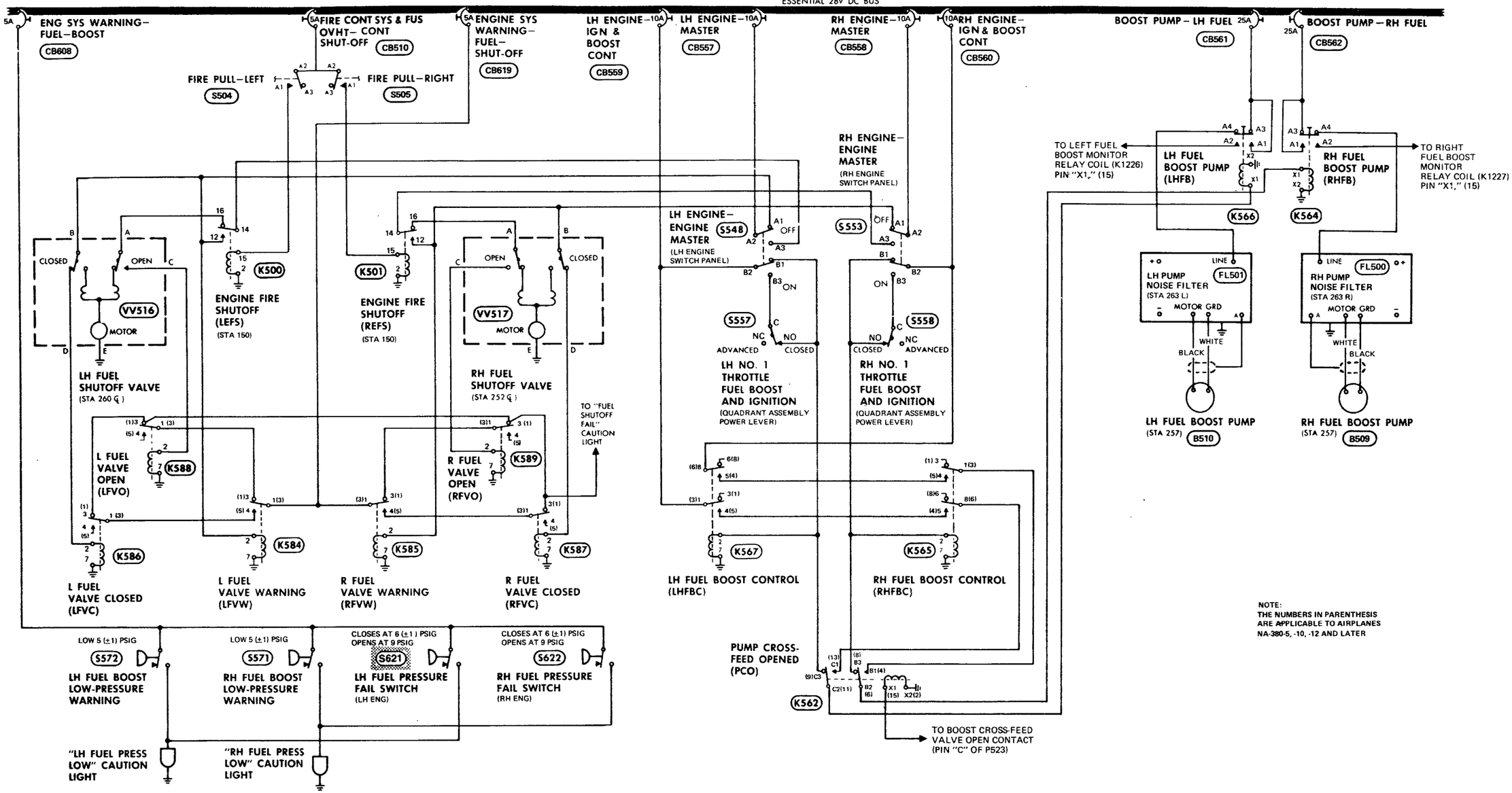

ESSENTIAL 28V DC BUS
ENG SYS WARNING–FUEL–BOOST *
FIRE CONT SYS & FUS OVHT– CONT SHUT-OFF †
ENGINE SYS WARNING–FUEL–SHUT-OFF *
LH ENGINE–IGN & BOOST CONT *
LH ENGINE–MASTER *
RH ENGINE–MASTER *
RH ENGINE–IGN & BOOST CONT *
BOOST PUMP–LH FUEL *
BOOST PUMP–RH FUEL *
FIRE PULL–LEFT (ENGINE FIRE CONTROL PANEL)
FIRE PULL–RIGHT (ENGINE FIRE CONTROL PANEL)
RH ENGINE–ENGINE MASTER (RH ENGINE SWITCH PANEL)
LH ENGINE–ENGINE MASTER (LH ENGINE SWITCH PANEL)
S548
S553
OFF
ON
S557
S558
ADVANCED
CLOSED
LH NO. 1 THROTTLE FUEL BOOST AND IGNITION (QUADRANT ASSEMBLY THROTTLE CONTROL)
RH NO. 1 THROTTLE FUEL BOOST AND IGNITION (QUADRANT ASSEMBLY THROTTLE CONTROL)
TO LEFT FUEL BOOST MONITOR RELAY COIL (PIN "X1")
TO RIGHT FUEL BOOST MONITOR RELAY COIL (PIN "X1")
LH FUEL BOOST PUMP (LHFB) §
RH FUEL BOOST PUMP (RHFB) §
K566
K564
LH PUMP NOISE FILTER (STA 263 L)
RH PUMP NOISE FILTER (STA 263 R)
LINE
MOTOR GRD
WHITE
BLACK
LH FUEL BOOST PUMP (STA 257)
RH FUEL BOOST PUMP (STA 257)
K500
ENGINE FIRE SHUTOFF (LEFS) (STA 333)
K501
ENGINE FIRE SHUTOFF (REFS) (STA 333)
VV516
LH FUEL SHUTOFF VALVE (STA 260 ℄)
VV517
RH FUEL SHUTOFF VALVE (STA 252 ℄)
TO "FUEL SHUTOFF FAIL" CAUTION LIGHT
K588
L FUEL VALVE OPEN (LFVO) ‡
K589
R FUEL VALVE OPEN ‡ (RFVO)
K586
L FUEL VALVE CLOSED (LFVC) ‡
K584
L FUEL VALVE WARNING (LFVW) ‡
K585
R FUEL VALVE WARNING (RFVW) ‡
K587
R FUEL VALVE CLOSED (RFVC) ‡
K567
LH FUEL BOOST CONTROL (LHFBC) §
K565
RH FUEL BOOST CONTROL (RHFBC) §
K663
LH JETTISON FUEL CONT (LJFC) ‡ 1
TO "FUEL JETTISON" SWITCH
K664
RH JETTISON FUEL CONT (RJFC) ‡ 1
TO "FUEL JETTISON" SWITCH
K562
PUMP CROSS-FEED OPENED (PCO) §
TO BOOST CROSS-FEED VALVE OPEN CONTACT ("C")
LOW 5 (±1) PSIG
S572
LH FUEL BOOST LOW-PRESSURE WARNING (STA 248 RH)
S571
RH FUEL BOOST LOW-PRESSURE WARNING (STA 248 RH)
CLOSES AT 6 (±1) PSIG. OPENS AT 9 PSIG
S622
LH FUEL PRESSURE FAIL SWITCH (LH ENG POD–STA 71)
RH FUEL PRESSURE FAIL SWITCH (RH ENG POD–STA 71)
"LH FUEL PRESS LOW" CAUTION LIGHT
"RH FUEL PRESS LOW" CAUTION LIGHT
* LH OVERHEAD FORWARD DC ESSENTIAL CIRCUIT-BREAKER PANEL
† RH OVERHEAD AFT DC ESSENTIAL CIRCUIT- BREAKER PANEL
1 NA-370-2 THROUGH -6, -8 AND -9
2 NA-370-1 AND -7
‡ STA 143 RH
§ RH PILOT'S COMPARTMENT RELAY BOX ASSEMBLY
S370-2-48-6A

Fuel Supply System Electrical Schematic - NA-370 Airplanes
Figure 28-00-2



NOTE:
THE NUMBERS IN PARENTHESIS ARE APPLICABLE TO AIRPLANES NA-380-5, -10, -12 AND LATER

S380-2-48-17A

Fuel Supply System Electrical Schematic
S/N 380 Airplanes
Figure 28-00-2A

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FUEL SYSTEM

MAINTENANCE PRACTICES

FUEL SYSTEM CHECK-OUT PROCEDURES

Fuel System Pressure Check

The following procedure applies to the complete fuel system and is used to determine that the system is free of leaks.

1. The special equipment and/or tools needed for this procedure are as follows:
   A. Air pressure source with air hose and shutoff valve.
   B. Low-pressure regulator (VQ2025, Victor Equipment Co., 3821 Santa Fe Ave., Los Angeles, Calif., or equivalent)
   C. Water manometer (Model A203, Meriam Instrument Co., Cleveland, Ohio) or pressure gauge capable of indicating a drop of 0.050 in Hg or 0.680 inch water when 3 psi pressure is indicated.
   D. Fuel vent pressurization adapter, 370-810078
   E. Hose to connect manometer or pressure gauge to fuel vent pressurization adapter
   F. Siphon breaker line plug, 265-810082
   G. Necessary caps and plugs to seal designated outlets

NOTE

Pressure check should be accomplished in an area where the ambient temperature remains comparatively constant. A 1°F change within the pressurized area of the airplane can cause a change of about one inch of water or 0.075 inch of mercury.

2. Remove fuel vent flame arrester assembly. (Refer to 28-10-3.)
3. Install fuel vent pressurization vent adapter on flame arrester mounting flange.
4. Make sure that right and left fuel shutoff valves, and on S/N 370-2 thru -6, -8 and -9) the right and left fuel jettison valves are in open position.
5. Check that fuel gravity refueling receptacle covers and all access doors and panels are installed and sealed properly. Check that all fuel tank sump drain valves and line drain valves are closed.
6. Plug outlet of fuel siphon breaker line at fuselage station 355. On S/N 370-2 thru -6, -8 and -9, plug fuel jettison line outlet. Plug right and left wing vent outlets. Cap fuel feed lines to right and left engines. Disconnect lines in engine pod.
7. Connect air pressure source to manometer or pressure gauge; then connect pressure recording unit to fuel vent pressurization adapter.
8. Pressurize fuel system to 3 psi. Allow 5 minutes for air pressure to stabilize in fuel tanks.

WARNING

Do not allow air pressure to exceed 3.0 psi maximum during this check. Excessive pressure can damage the fuselage fuel tank.

9. Close air pressure valve.
10. Conduct checks in steps A. and B. below. If, after 15 minutes, the results of the checks are as specified, no leakage is indicated in the fuel system.
    A. Check that water manometer (if this equipment is used) does not indicate a drop of more than 0.680 inch of water.
    B. Check that pressure gauge (if this equipment is used) does not indicate a drop of 0.050 inch of mercury or 0.680 inch of water.
11. Actuate relief valve on fuel vent pressurization adapter and exhaust air from fuel system.
12. Disconnect pressure recording unit from fuel vent pressurization adapter. Remove adapter from flame arrester mounting flange.
13. Remove all plugs and caps installed for this check.

WARNING

Failure to remove all plugs and caps can damage the fuselage tank when the fuel system is pressure-refueled.

14. Install fuel vent flame arrester assembly. (Refer to 28-10-3.)

Fuel System Obstruction Check

1. The special equipment and/or tools needed for this procedure are as follows:
   A. Air pressure source with air hose and shutoff valve
   B. Low-pressure regulator (VQ2025, Victor Equipment Co., 3821 Santa Fe Ave., Los Angeles, Calif., or equivalent)
   C. Fuel vent pressurization adapter, 370-810078

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MAINTENANCE PRACTICES (Continued)

Fuel System Obstruction Check (Cont'd)

D. Necessary caps to seal designated outlets

2. Remove fuel vent flame arrester assembly. (Refer to 28-10-3.)
3. Install fuel vent pressurization adapter to flame arrester mouting flange.
4. Cap fuel feed lines leading to right and left engines. Disconnect lines in engine pod.
5. Connect air pressure source to fuel vent pressurization adapter. Apply 3 psi pressure.

WARNING

Do not allow air pressure to exceed 3 psi. Damage to the fuselage cell can be caused by excessive pressure.

6. Check for airflow at any fuel sump drain in left wing and then for airflow at any fuel sump drain in right wing. Airflow indicates fuel vent and transfer lines are unobstructed.
7. Actuate right fuel shutoff valve and, on S/N 370-2 thru -6, -8, and -9 the right jettison valve to open position.
8. On S/N 370-2 thru -6, -8, and -9, check for airflow at fuel jettison line outlet. Airflow indicates right fuel jettison line is unobstructed.
9. Remove cap from fuel feed line leading to right engine. Airflow indicates right fuel feed line is unobstructed. Recap line.
10. Actuate right fuel shutoff valve to close position and left fuel shutoff valve to open position. On S/N 370-2 thru -6, -8, and -9, actuate right jettison valve to closed position, and left jettison valve to open position.
11. On S/N 370-2 thru -6, -8, and -9 check for airflow at fuel jettison line outlet. Airflow indicates left fuel jettison line is unobstructed.
12. Remove cap from fuel feed line leading to left engine. Airflow indicates left fuel feed line is unobstructed.
13. Remove air pressure from main fuel vent outlet. Actuate left fuel shutoff valve and, on S/N 370-2 thru -6, -8, and -9 the left jettison valve to closed position. Connect fuel feed lines at right and left engine pods.
14. Install fuel vent flame arrester. (Refer to 28-10-3.)

MICROBIOLOGICAL FUEL DECONTAMINATION

General

Hydrocarbon fuels, such as aircraft turbine fuel, are subject to bacterial and fungal attack during storage, transportation, and use. The microorganisms that feed on these hydrocarbons not only grow in small quantities of water present in the fuel system, but also grow in the hydrocarbon fuel itself. Microbes can live for months in essentially anhydrous fuels. Subsequent presence of free water may produce rapid population increase.

NOTE

Draining any water collected in the fuel sump areas on a DAILY basis is the most critical maintenance procedure to prevent water accumulation that may, if left unchecked, result in microorganisms contaminating the fuel system. This is especially important if the airplane is inactive for any period of time.

Microorganisms produce a slime which suspends water and particulate matter in fuel. Suspended contaminants lead to fouled airplane fuel quantity probes, inaccurate gauge readings, and fuel filter plugging. Persistant or multiple fuel probe failures and/or presence of fouling in the fuel filters or sump drains may be indications of microbe contamination.

Microorganisms also attack and perforate fuel tank sealants and coatings, permitting corrosive attack on the exposed metal. On unprotected areas, the microorganisms promote rusting of steel and corrosion of aluminum.

CAUTION

When initially using a biocide agent, check the fuel filters at increased intervals. Dying microorganisms will release specks of dirt and rust that have been caught in the fungal mats. Also, the fungus itself may come off walls and structures as it is killed. Once dead, the slimy nature of the fungus will disappear, and it should become more filterable.

Draining any water collected in the fuel sump areas on a daily basis helps prevent a water accumulation that may, if left unchecked, result in microorganisms damaging the fuel system.

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MAINTENANCE PRACTICES (Continued)

Treatment of Fuel with Biocide Agent

1. Fuel Without PRIST™

NOTE

Fuel without PRIST™ must be treated on a one month interval. Treat fuel system with a biocide agent at the 135 ppm level. If microbe contamination is suspected or detected, treat fuel system with a biocide agent at the 270 ppm level. Continue to treat the fuel system at the 270 ppm level until all traces of microbe contamination is eliminated (refer to FUEL TREATMENT PROCEDURE).

2. Fuel With PRIST™

NOTES

- ***Airplanes fueling with PRIST™ additive a minimum of ten consecutive days each month may extend the one month interval to six months.***
- ***Approved fuels must contain a concentration of 0.10 to 0.15% by volume of PRIST™ fuel additive. Treat fuel system with a biocide agent at the 135 ppm level. If microbe contamination is suspected or detected, treat fuel system with a biocide agent at the 270 ppm level. Continue to treat fuel system at the 270 ppm level until all traces of microbe contamination are eliminated.***

Fuel Test Procedures

1. Drain a small amount of fuel from tank sumps and test using Boron Microb Monitor Test Kit. Kits may be obtained from:

   BP Oil Company
   200 Public Square
   Cleveland, OH 44114-2375

2. Treat fuel system with biocide agent if microbes are detected per treatment instructions on Table 601 or 602. Maintain a 135 ppm biocide agent concentration if contamination persists.

Fuel Treatment Procedures

1. Fuel supply tank(s) and systems

NOTE

Metered bulk treatment of the fuel supply tank and system is the most effective method of controlling microbe contamination and growth.

CAUTION

Do not exceed the 270 ppm concentration of biocide agent in the fuel tank. Do not add biocide agent to fuel with large concentrations of water. Biocide agents can form crystals and solidify. All water must be drained from the sumps prior to the addition of biocide agent.

Recommended treatment levels are as follows:

A. One gallon of biocide agent per 5000 gallons of fuel (270 ppm concentration) for treatment of contaminated fuel storage systems.

B. One-half gallon of biocide agent per 5000 gallons of fuel (135 ppm concentration) for maintenance of a clean fuel storage system.

NOTE

The fluid ounces of biocide agent required may be calculated by multiplying lbs. of fuel x 0.004 to obtain a concentration of 270 ppm, or lbs. of fuel x 0.002 for a concentration of 135 pp.

C. Fill fuel tank approximately half full; slowly add biocide agent during the addition of the remaining fuel.

NOTE

This is an alternative method of adding biocide agent when metering is not possible during gravity refueling. This will disperse the agent and avoid areas of high concentration.

2. Airplane fuel system

A. Treat fuel per Fuel Treatment Table 601 or 602, and allow the biocide treated fuel to remain in the fuel tank for 72 hours if the airplane is to remain inactive for several days.

B. Retreat fuel at each refueling stop to maintain biocide treated fuel in the fuel tanks for a 72 hour period if the airplane is scheduled to fly during biocide application. Caution should be exercised not to exceed the 270 ppm maximum concentration of biocide agent in the airplane fuel.

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MAINTENANCE PRACTICES (Continued)

Table 601. Biocide Fuel Treatment — NA 265-70 and -80

TANK	POUNDS OF FUEL (Gravity/refueling @ 6.7 lbs/gallon)	OUNCES OF BIOCIDE Concentration	
		A 270 ppm	B 135 ppm
Aft Fuselage	1369	5.5	2.7
RH Wing	3067	12.3	6.1
LH Wing	3067	12.3	6.1
TOTAL	7503	30.1	14.9

NOTES
A For treatment of contaminated fuel.
B For maintenance of clean fuel.

Table 602. Biocide Fuel Treatment — for NA 265-80, S/N 380-70 and -71 only

TANK	POUNDS OF FUEL (Gravity/refueling @ 6.7 lbs/gallon)	OUNCES OF BIOCIDE Concentration	
		A 270 ppm	B 135 ppm
Aft Fuselage	1369	5.5	2.7
RH Wing w/o Fwd Tank	3067	12.3	6.1
RH Wing with Fwd Tank	3460	13.8	6.9
LH Wing	3067	12.3	6.1

NOTES
A For treatment of contaminated fuel.
B For maintenance of clean fuel.

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MAINTENANCE PRACTICES (Continued)

PROCEDURE	NORMAL INDICATION
1. Circuit breakers:	
BOOST PUMP — RH FUEL ..... in	
FWD FUEL TRANSFER ..... in	
RH ENG IGN BOOST CONT ..... out	
2. Apply external 28 VDC to airplane.	
3. Switches:	
INV ..... NO. 1	
FUEL QTY ..... RH IND	
FWD FUEL TANK TRANSFER ..... MAN	Monitor RH FUEL QTY gauge until approximately 100 lbs. fuel remains in fwd fuel tank.
4. FWD FUEL TANK TRANSFER ..... AUTO	Fuel transfer should stop.
5. Circuit breakers:	
STALL WARN ..... out	
BATT VENT ..... out	
LDG POS ..... out	Fuel transfer should start. (Check decreasing level on RH FUEL QTY gauge.)
6. Allow all fuel to transfer from fwd tank.	
7. Check terminal L4 of TB593 for 28 VDC.	No voltage should be present.
8. FWD FUEL TANK TRANSFER ..... MAN	
9. Check for 28 VDC at terminal L4 of TB593	Voltage should be present.
10. Circuit breakers:	
LDG POS ..... in	
RH ENG — IGN BOOST CONT ..... in	
11. FWD FUEL TANK FILL switch ..... ON	Fuel will transfer from wing tanks to fwd fuel tank.
12. Allow approximately 100 lbs. of fuel to transfer to fwd tank.	
13. FWD FUEL TANK TRANSFER ..... AUTO	
FWD FUEL TANK FILL ..... OFF	Check that RH boost pump stops.
14. Finish refueling fwd tank.	Fwd fuel tank float valve is operating correctly when boost pump stops.
15. Safetywire FWD FUEL TANK FILL switch with 0.020 soft copper wire.	
16. Remove electrical power from airplane.	

NEW

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FUEL TANKS - DESCRIPTION

1. Fuselage Fuel Tank

The fuselage tank is a nonsealing, rubber and nylon bladder-type cell, suitable for use with aromatic fuel. This tank is supported and held securely in place by the various cell fittings and hangers. It is installed in the fuselage rear section, just aft of fuselage station 333. This tank will hold 203 US gallons plus a 1.2 percent expansion space for the fuel in the wing and fuselage tanks. The fuselage tank is installed in a sealed, fluid- and vapor-tight bay. An overboard drain from the bottom of the sealed bay drains any fuel leakage or condensation that accumulates in the bay. Any area in the bay that might be abrasive to the rubber surface of the cell is covered with pressure-sensitive tape. The fuselage tank is interconnected to the wing tanks by two vent and transfer lines, a pressure refueling line, and two fuel pressure lines (primary test and secondary test) for testing the dual level control valve. A vent line, running from the top of the fuselage tank to an outlet in the tail cone, vents the entire fuel system. A siphon-breaking vent line, with a check valve allowing air to enter the tank only, prevents any possible siphoning of fuel from the fuselage tank through the vent line to the tail-cone outlet. This tank is normally filled by the pressure refueling method; however, the fuselage tank can be refueled through a filler access on the right side of the fuselage, at fuselage station 351, just above the right engine pod.

2. Integral Wing Fuel Tanks

Each integral wing tank contains 451.5 gallons of fuel. This fuel is carried in the sealed main structural box, between the front and rear spars and extending from the inboard or centerline rib to the tip panel. Each wing is divided into four compartments by three wing ribs, one at wing station 37, one at wing station 106, and the other at wing station 171.6. These ribs prevent excessive outboard fuel flow during airplane maneuvers. These compartment dividing ribs are not sealed, but the fit at each end and along the bottom is snug and tight. However, the ribs at wing stations 37 and 106 have cutouts between the top of the rib attach points and the upper wing skin, and the rib at wing station 171.6 has a gap between the top of the rib and the upper wing skin to permit venting between compartments.

Flapper-type check valves and interconnect tubes allow transfer of fuel inboard to the wing center section. The check valves prevent fuel flow outboard; the interconnect tubes are positioned to prevent fuel flow outboard except during refueling. The pressure refueling outlet for each wing is in the right and left center sections and when refueling is being done, the inboard compartment fills up first; then the fuel spills over the compartment bulkhead and flows through the interconnect tubes into the next outboard compartment. This procedure continues until the fuel reaches the wing tip, actuates the level control valve, and shuts the refueling flow off. Wing fuel caps are in the outer wing panel, near the wing tip and provide a means for gravity refueling. The fuel cap consists of a body, an inner cap, a spring, an outer cap, and a nozzle guard. The function of the spring-loaded inner cap is to prevent fuel from gushing out when the outer cap is removed and the fuel is under pressure from fuel in the fuselage tank.

Basic fuel sealing of the integral wing tank structure has been done by the groove injection sealing method. Grooves 0.050 inch deep and 0.188 inch wide are machined into one structural member at mating surfaces. These grooves run along the centerlines of rows of closely spaced screws or bolts which hold the mating parts together. A spot face in the groove at each fastener provides a continuous mating surface sealant path which seals the fastener while continuing the mating surface seal. A nonhardening sealant is injected into the grooves through special holes a few inches apart. After injection, these holes are closed by inserting self-tapping screws. Injection is done with a special gun, similar to an ordinary lubrication gun. The sealing gun is operated with air pressure. The pressure ratio between the incoming regulated air pressure and the injection nozzle is 70 to 1. This ratio allows injection pressure at the groove injection holes to reach as high as 6000 psi, although the pressure is usually much lower. This method of sealing the tank envelope has certain advantages. One is that injection can be done at all times from the exterior of the wing. Another is that periodic repair of the sealed article can be done by reinjecting the grooves with more nonhardening sealant. Where mating-surface sealing is not required (as at the intermediate rib spars), screws have been individually sealed during installation.

The second sealing method used in the wing involves the use of rubber. A few bolts in isolated but accessible locations, as well as the screws attaching the access doors, have been sealed with Stat-O-Seals, which are "O" rings bonded inside metal washers. The access doors have a synthetic rubber sealing ring bonded directly into the doors. Some access doors and fittings are sealed with a replaceable synthetic rubber O-ring seal. With few exceptions, synthetic rubber seals are used where removal of parts may be frequent.

3. Fuel Vent System

The vent system maintains a proper balance of pressure within the fuel system during all operating conditions: climbing, diving, in-flight and ground refueling, and draining. The vent system (1) enables air or vapor replacement of the fuel in each tank as the fuel is transferred, drained, or used; (2) maintains higher pressure inside the fuselage tank cell than outside, preventing collapse and possible rupture; (3) keeps internal pressure at a safe level, determined by the structural limits of the airplane; and (4) maintains balanced pressure between tanks, enabling fuel transfer. Obstructions in the vent system may cause structural failure, failure of the fuel system to completely fill, failure of fuel to transfer from tank to tank, and collapse of the fuselage fuel cell and integral wing tanks accompanied by bending of the fuel gage tank units.

The fuselage tank provides the necessary expansion volume and an overboard venting outlet for the fuel system. Additional venting for single-point pressure refueling is provided by a pressure-actuated wing tip vent valve in each wing tip.

The wing tanks are vented to the fuselage tank through interconnecting vent and transfer lines and two spring-loaded check valves. The 1-1/4 (±1/4) psi setting of the check valves prevents fuel transfer from the wing tanks to the fuselage tank during ground refueling or normal flight attitudes.

The wing tanks are climb-vented to the fuselage tank. A check valve in the line (attached to the "T" fitting on top of the fuselage tank) allows airflow from the wing tanks to the auxiliary tank only. This venting assists fuel transfer during climbing attitudes and expedites completion of fuel transfer when level flight attitude is attained.

The inboard leading edge sections of the wing tanks are vented to the inboard compartments of the integral wing tanks. The right and left booster pump sump areas are vented to the right and left wing center section areas. The wing compartments are vented through vent openings in the ribs separating each compartment. The fuselage tank main fuel vent line runs aft along the fuselage upper centerline to the tail cone vent outlet.

As the airplane climbs, the outside air pressure decreases. Whenever the interconnecting vent and transfer lines between the wing fuel tanks and the fuselage cell are below the fuel level in the wing tanks and the pressure within the wing tanks exceeds the check valve pressure of 1-1/4 (±1/4) psi, fuel will be transferred to the fuselage tank to provide expansion room in the wing tanks. When the fuel is below the vent and transfer lines, the wing tanks will vent directly to the fuselage tank and the overboard vent outlet. Any fuel transferred to the fuselage tank by expansion in the wing tank will return to the wing tank as use of fuel in the wing tanks and cooling of the wing tanks in flight results in pressure reduction in those tanks. As the airplane dives or loses altitude, the outside air pressure increases. Air pressure enters the vent line and keeps the pressure within the wing tanks and the fuselage tank equal to the pressure outside. If, through vent malfunction, the wing tank pressure should exceed a negative of 4 to 5 psi, the wing tip vent valve will open and admit air to prevent exceeding that pressure differential.

FUEL TANKS - TROUBLE SHOOTING

1. Trouble Shooting Fuel Tanks

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. FUEL LEAKAGE AROUND FUSELAGE TANK ACCESS DOORS.		
(1) Attaching bolts undertorqued.	Check for proper torque setting.	Torque properly.
(2) "O" ring improperly installed in tank fitting.	Remove and visually inspect "O" ring.	Install "O" ring properly.
(3) "O" ring not installed in tank fitting.	Disconnect fitting and inspect for presence of "O" ring.	Install "O" ring properly.
(4) Faulty fuel cell fitting.	Visually inspect fitting for defect of sealing flanges.	Replace cell.
B. TOO MUCH FUEL REMAINS IN RIGHT OR LEFT CENTER WING SECTION AFTER FUEL SYSTEM HAS BEEN DEFUELED.		
(1) Improper equipment used to defuel airplane.	Check that defueling hose has a restrictor line (3/8 inch OD tubing with 0.035-inch wall thickness, 8 feet long) to keep boost pump from transferring more fuel than it receives.	If restrictor line is not available, jack up nose of airplane 8 to 9 degrees to provide gravity flow to boost pump sump.
(2) Check valve in fuel transfer ejector line reversed.	Remove access plate on rear spar and inspect check valve for proper installation.	Install check valve properly.
(3) Boost pump fuel bleed flow line not installed properly.	Remove access plate on rear spar and inspect line for proper installation at slip joint or "B" nut.	Install line properly.
(4) Ejector outlet to boost pump sump area improperly installed.	Remove access plate on rear wing spar and inspect for proper installation of ejector outlet.	Install ejector to boost pump sump line properly.
(5) Ejector inlet screen covered.	Remove sump drain valve beneath ejector and check for foreign object.	Remove foreign object and check operation of ejector.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
C. SUCTION - FEED FUEL SUPPLY INSUFFICIENT.		
(1) Boost pump check valve does not open.	Through boost pump access door, check boost pump flapper valve for sticking or binding.	Clean flapper-valve seat. Replace pump.
(2) Boost pump compartment flapper valves do not allow proper fuel flow to compartment.	Through boost pump access door, check eight flapper valves for sticking or binding.	Clean flapper-valve seats. Replace flapper valve or valves.

FUEL TANKS - MAINTENANCE PRACTICES

1. Defueling Fuel Tanks

A. Fuel Servicing Precautions

(1) Defueling should be done out of doors, with the airplane at least 100 feet from hangars or other airplanes.

(2) Place "HAZARDOUS AREA" signs around airplane.

(3) Ground airplane with two ground wires, from different points on airplane. Ground all workstands and power units to same ground as that used for airplane.

(4) Stop all other maintenance on airplane. All other persons not engaged in defueling or inerting airplane should remain clear of the airplane.

B. Defueling - On Airplanes Serial No. NA370-2 through -6, -8 and -9

CAUTION: DO NOT RUN ENGINE TO DEFUEL AIRPLANE. THE ENGINE-DRIVEN FUEL PUMP MAY BE DAMAGED AT TIME OF FLAME-OUT IF THE ENGINE IS RUN UNTIL THE FUEL SUPPLY IS EXHAUSTED.

NOTE: The fuel system may be defueled through fuel feed line at right engine, if desired, by following these procedures and substituting RH in place of LH in steps 1, 3, 5, 6, 8 and 11, and step 7, LH for RH.

(1) On left overhead forward dc essential circuit-breaker panel, engaged the following circuit breakers:

(a) "LH ENGINE - IGN & BOOST"

(b) "LH ENGINE - MASTER"

(c) "BOOST PUMP - LH FUEL"

(2) Make sure "EMERGENCY - FUEL JETTISON" switch on master and emergency overhead switch panel is at OFF.

CAUTION: A RESTRICTOR TUBE IS INSTALLED ON END OF DEFUELING HOSE TO HOLD THE BOOST PUMP OUTPUT FLOW TO A VALUE THAT PREVENTS EMPTYING THE BOOST PUMP SUMP CENTER WING COMPARTMENTS AHEAD OF THE CENTER WING COMPARTMENT CONTAINING THE EJECTOR. THIS WILL PREVENT FUEL BOOST PUMP CAVITATION DAMAGE.

(3) Disconnect fuel feed line at left engine; then connect a hose to line. Attach a restrictor tube (3/8 inch O.D. (.035 inch wall thickness) by 8 feet) to end of defueling hose.

(4) Connect 28-volt dc external power source to airplane.

(5) Actuate "LH ENGINE - ENGINE MASTER" switch to ON, opening left fuel shutoff valve.

(6) Actuate fuel cross-feed and tank selector switch to X-FEED. Advance left engine throttle to IDLE, opening both boost cross-feed valve and tank cross-feed valve and energizing left boost pump.

(7) Engage "BOOST PUMP - RH FUEL" circuit breaker, energizing right boost pump.

(8) When fuel ceases to flow from hose to tank truck, actuate left engine throttle to CLOSE, fuel cross-feed and tank selector switch to NORM, and "LH ENGINE - ENGINE MASTER" switch to OFF.

(9) Disconnect external power source and position circuit breakers and switches to normal positions.

(10) Drain residual fuel into closed containers. (Refer to Chapter 12.)

(11) Connect left fuel feed line to left engine.

C. Defueling - On Airplanes Serial No. NA370-1 and -7 and NA380-1 through -37 not changed by Service Bulletin 74-23.

CAUTION: DO NOT RUN ENGINE TO DEFUEL AIRPLANE. THE ENGINE-DRIVEN FUEL PUMP MAY BE DAMAGED AT TIME OF FLAME-OUT IF THE ENGINE IS RUN UNTIL THE FUEL SUPPLY IS EXHAUSTED.

NOTE: The fuel system may be defueled through RH engine fuel filter, if desired, by following these procedures and substituting RH in place of LH in steps 1, 2, 4, 5 and 7 and in step 6, LH for RH.

(1) On left overhead forward dc essential circuit-breaker panel, engage the following circuit breakers:

(a) "LH ENGINE - IGN & BOOST"

(b) "LH ENGINE - MASTER"

(c) "BOOST PUMP - LH FUEL"

CAUTION: A RESTRICTOR TUBE IS INSTALLED ON END OF DEFUELING HOSE TO HOLD THE BOOST PUMP OUTPUT FLOW TO A VALUE THAT PREVENTS EMPTYING THE BOOST PUMP SUMP CENTER WING COMPARTMENTS AHEAD OF THE CENTER WING COMPARTMENT CONTAINING THE EJECTOR. THIS WILL PREVENT FUEL BOOST PUMP CAVITATION DAMAGE.

(2) Remove the plate from the bottom of LH engine fuel filter and remove filter and attach defueling adapter. (See figure 28-10-1.) Connect a defueling hose to adapter. Attach a restrictor tube (3/8 inch O.D. (.035 inch wall thickness) by 8 feet long) to end of defueling hose. Extend hose and line to tank truck.

NOTE: To reduce refueling time, the restrictor tube is not required until 335 pounds of fuel is remaining in the tanks.

(3) Connect 28-volt dc external power source to airplane.

(4) Actuate "LH ENGINE - ENGINE MASTER" switch to ON, opening left fuel shutoff valve.

(5) Actuate fuel cross-feed and tank selector switch to X-FEED. Advance left engine throttle to IDLE, opening both boost cross-feed valve and tank cross-feed valve and energizing left boost pump.

(6) Engage "BOOST PUMP - RH FUEL" circuit breaker, energizing right boost pump.

(7) When fuel ceases to flow from hose to tank truck, actuate left engine throttle to CLOSE, fuel cross-feed and tank selector switch to NORM, and "LH ENGINE - ENGINE MASTER" switch to OFF.

(8) Disconnect external power source and position circuit breakers and switches to normal positions.

(9) Drain residual fuel into closed containers. (Refer to Chapter 12.)

(10) Disconnect defueling hose from defueling adapter and remove adapter and reinstall filter and plate on fuel filter.

D. Defueling - On Airplanes NA370-1 and -7 and NA380-1 through -37 changed by Service Bulletin 74-23 and NA380-38 and Later.

CAUTION: DO NOT RUN ENGINE TO DEFUEL AIRPLANE. THE ENGINE-DRIVEN FUEL PUMP MAY BE DAMAGED AT TIME OF FLAME-OUT IF THE ENGINE IS RUN UNTIL THE FUEL SUPPLY IS EXHAUSTED.

NOTE: The fuel system may be defueled through the RH fuel feed line, if desired, by following these procedures and substituting RH in place of LH in steps 1 through 3 and 5, 6 and 8 and in step 7, LH for RH.

(1) On left overhead forward dc essential circuit-breaker panel, engage the following circuit breakers:

(a) "LH ENGINE - IGN & BOOST"

(b) "LH ENGINE - MASTER"

(c) "BOOST PUMP - LH FUEL"

(2) Remove lower access door from bottom of rear fuselage to gain access to LH engine fuel feed line drain cock.

CAUTION: A RESTRICTOR TUBE IS INSTALLED ON END OF DEFUELING HOSE TO HOLD THE BOOST PUMP OUTPUT FLOW TO A VALUE THAT PREVENTS EMPTYING THE BOOST PUMP SUMP CENTER WING COMPART-

MENTS AHEAD OF THE CENTER WING COMPARTMENT CONTAINING THE EJECTOR. THIS WILL PREVENT FUEL BOOST PUMP CAVITATION DAMAGE.

(3) Remove "B" nut with drain cock from LH fuel feed line. Connect a defueling hose to "B" nut fitting on LH fuel feed line. Attach a restrictor tube (3/8 inch O.D. (.035 inch wall thickness) by 8 feet long) to end of defueling hose. Extend hose and line to tank truck.

(4) Connect 28-volt dc external power source to airplane.

(5) Actuate "LH ENGINE - ENGINE MASTER" switch to ON, opening left fuel shutoff valve.

(6) Actuate fuel cross-feed and tank selector switch to X-Feed. Advance left engine throttle to IDLE, opening both boost cross-feed valve and tank cross-feed valve and energizing left boost pump.

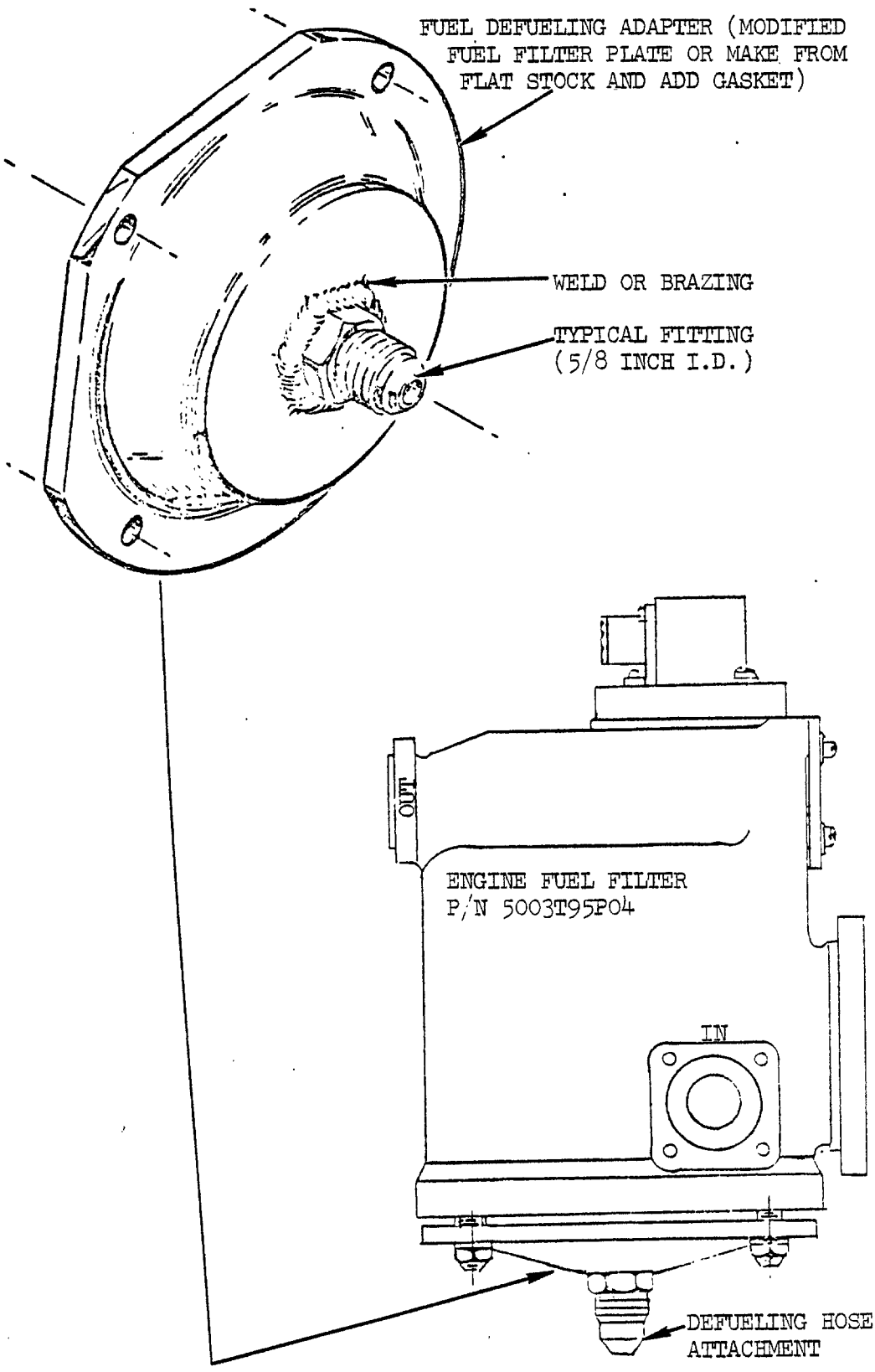
(7) Engage "BOOST PUMP - RH FUEL" circuit breaker, energizing right boost pump.

(8) When fuel ceases to flow from hose to tank truck, actuate left engine throttle to CLOSE, fuel cross-feed and tank selector switch to NORM, and "LH ENGINE - ENGINE MASTER" switch to OFF.

(9) Disconnect external power source and position circuit breakers and switches to normal positions.

(10) Drain residual fuel into closed containers. (Refer to Chapter 12.)

(11) Disconnect defueling hose from "B" nut fitting and reinstall "B" nut with drain cock.


FUEL DEFUELING ADAPTER (MODIFIED FUEL FILTER PLATE OR MAKE FROM FLAT STOCK AND ADD GASKET)
WELD OR BRAZING
TYPICAL FITTING (5/8 INCH I.D.)
OUT
ENGINE FUEL FILTER P/N 5003T95P04
IN
DEFUELING HOSE ATTACHMENT
S380-2-48-5

Fuel Defueling Adapter (Typical)
Figure 28-10-1

1. Handling Fuel Cells

Special attention should be given to the care and handling of the nylon bladder-type fuel cells.

2. Removing Fuselage Fuel Tank

Access to the cell is through the fuselage rear section lower access door and through a large door in the aft face of the tank compartment, fuselage station 357. The fuel fittings to be disconnected are readily accessible from the aft side and through doors in the fuselage rear section below the tank. Removal of the filler assembly provides access to the top of the cell.

A. Remove fuselage tank as follows:

(1) Cover all disconnected fittings, "B" nuts, screen, and all cell openings as soon as possible to protect fuel system.

WARNING: MAKE SURE BATTERY SWITCH IS OFF AND EXTERNAL ELECTRICAL POWER IS DISCONNECTED FROM AIRPLANE; FAILURE TO DO THIS MAY RESULT IN AN EXPLOSION.

(2) Remove all necessary heat and vent lines in front of fuselage tank access panel and tank connections.

(3) In left top aft portion of cell, disconnect lines to check valve and dual integral level control valve.

(4) Disconnect fuel vent line.

(5) Remove nuts from inner panel support studs (four).

(6) Remove outer panel. Place bolts and washers in a bag and mark.

(7) Mark location of four inner panel support studs; then remove inner panel. Bag bolts and washers and mark. Remove "O" ring.

(8) On right top of cell, disconnect fuel quantity coaxial cables. Remove fuel probe assembly.

(9) Remove boss assembly holding vent line in position.

(9A) Remove four bolts and washers from fuel cell support pad. These bolts are on the forward side of the aft cabin bulkhead.

(10) Remove vent line from inside cell.

(11) Remove check valves and T-fitting assembly from aft end of top of cell.

(12) Remove 24 bolts from plate holding dual integral level valve assembly to structure. Bag bolts and washers; then mark.

(13) Remove five bolts attaching plate to valve flange. Bag bolts and washers; then mark.

(14) Remove five countersunk screws holding dual integral level control valve. Remove valve. Bag screws; then mark.

(15) Remove 10 bolts, 10 washers, and 10 Stat-O-Seal seals from gravity drain valve assembly. Bag raw stock; then mark.

(16) Cover gravity drain valve assembly with protective covering.

(17) Remove three bolts and washers from fuel cap mounting ring. Remove gasket.

(18) Remove 16 bolts and washers holding fuel cap mounting ring to phenolic ring. Bag bolts and washers; then mark.

(19) Remove three countersunk screws from phenolic ring to fuel cell opening. Bag screws; then mark. Remove remaining parts of assembly.

(20) Remove two bolts and washers from cell support pad in forward side of cell.

(21) Remove 10 bolts and washers from cell support pads on top of cell. Bag bolts and washers; then mark.

(22) Disconnect cell support springs.

(23) Remove cell from bay.

3. Installing Fuselage Fuel Tank

NOTE: Remove and discard all protective covers and/or coating from fuel cell fittings, and carefully wipe fittings clean before installation of "O" rings or mating parts. Retain "O" rings in groove. Do not smear petroleum over metal surface.

- Torque all bolts.
- The fuselage fuel tank is manufactured from nylon fiber which will shrink over a prolonged storage period. Due to this possible shrinkage the cell should be submerged in water which has been heated to 120°F and soaked for 4 hours. Subsequent to the soaking period, the cell should be installed within 24 hours.

WARNING: MAKE SURE BATTERY SWITCH IS OFF AND EXTERNAL ELECTRICAL POWER IS DISCONNECTED FROM AIRPLANE; FAILURE TO DO THIS MAY RESULT IN AN EXPLOSION.

A. Install fuselage tank as follows:

(1) Make sure all support springs have been positioned in their containers; then dust cell with talcum powder.

(2) Fold tank for installation. Make sure gravity feed valve opening is positioned for installation on bottom and aft cell opening is positioned to face aft.

(3) With one mechanic in fuselage rear section ready to receive cell, carefully feed cell through fuselage rear section access opening.

(4) Slide cell through cell compartment access opening and unfold for spring hookups.

CAUTION: IF LINER TAPE HAS BEEN REPLACED OR IF SEALANT HAS BEEN APPLIED OVER TAPE, ALLOW 4 TO 6 HOURS DRYING TIME BEFORE INSTALLING CELL.

NOTE: As a help in positioning the five support pads on top of cell, to reduce possibility of damage, cut off boltheads and radius ends of five 2-inch full-threaded 1/4-inch bolts. Secure a nut for each locator stud and make sure it screws on and off the stud easily.

(5) Place a locating stud in one side of each cell support pad. There are four pads positioned around the fuel probe access and one on the left aft top area.

(6) Install cell support hanger springs in the following sequence.

NOTE: A helpful method of positioning the springs is as follows: Take a piece of strong nylon; tie cord about 20 inches long; then build an 8-inch loop with a slip bow tie at one end. Hook up one end of spring to retainer cup, put loop around the other end; then pull on string and guide spring end into its retainer cup. Untie slip bow and pull string out.

(a) Starting at left top of cell, install top two springs.

(b) In top of cell, center and forward, install two springs.

(c) In top of cell, center and to right, install two springs.

(7) Have one man on top of fuselage rear section at filler neck access and one man to push top of cell upwards. Position top of cell to top of compartment so that guide studs will go through proper holes. Have man on top of fuselage put stud nuts on studs and draw cell into position.

(8) Install proper bolts and washers beside each stud, remove studs, and install remaining bolts and washers.

(9) Continue to install cell support hangar springs.

(a) On forward side at top, install two springs.

(b) On forward left side, install two upper springs.

(c) On forward right side, install two upper springs.

(10) Have one man go inside fuselage forward section to aft bulkhead. Through aft fuselage bulkhead, install four bolts and washers in cell support pad in forward side of cell. The man can assist by centering pad from inside of cell.

(11) Continue to install cell support hanger springs.

(a) On forward side, on bottom, install three springs.

(b) On left side, along bottom, one forward and two aft, install three springs.

(c) On right side, one forward bottom and two aft, install three springs.

(d) On aft side, install three springs on top and three on bottom.

(12) Install filler neck assembly in right side of aft fuselage.

(a) Place phenolic ring over "O" ring; then place fuel cap mounting ring over phenolic ring. Align holes in two units and cell pad. Install three countersunk screws.

(b) Place 265-484083 gasket inside metal ring. Place ring, gasket down, on rim of access hole. Install 16 bolts and washers.

(c) Place 265-484082 gasket on fuel cap mounting ring and align bolt holes. Insert hand in ring hole, lift upward until gasket is next to metal ring; then install three bolts and washers. Install fuel cap.

(13) Remove protective covering from gravity drain valve assembly. Attach bottom of cell to drain assembly flange with 10 bolts, 10 washers, and 10 Stat-O-Seal seals.

(14) On upper left, aft side, install dual integral level control valve.

(a) Position valve assembly, align holes; then install five 1/4-inch countersunk screws.

(b) Position 265-484073 gasket inside solid plate. Position 265-484074 gasket next to valve flange and align holes; then install solid plate to valve and structure.

(c) Install five 1/4-inch bolts and washers through plate to valve flange, then install 24 bolts and washer through plate to structure.

(15) In left top aft portion of cell, install "T" fitting and check valve assembly.

(a) Make sure "O" ring on "T" fitting is not nicked or damaged. Jam nut should be on upper portion of fitting threads. "O" ring should be positioned in groove between threaded portions.

(b) Install "T" fitting and check valve assembly (P/N F61A0208), with valve flapper hinge positioned at the top. When "O" ring touches surface of pad and check valve points aft, tighten jam nut. Arrows should point toward "T" fitting.

NOTE: Make sure check valve (P/N F61A0208) is installed with flapper hinge positioned at the top; otherwise, valve will not function as intended. The hinge location is marked on the outside hex portion of the valve body with red dye.

(16) Install overboard vent line inside cell.

(a) Install line 265-488415 inside cell with aft end protruding through hole in cell and forward end aligned with holes in mounting pad. Install two 1/4-inch bolts and washers.

(b) Install "O" ring inside boss at flange end. Slide boss over end of line; then bolt to tank with three 1/4-inch countersunk screws.

(c) Position 265-484072 gasket over boss flange and 265-484071 gasket inside mounting plate. Position plate over boss flange and structure; align with holes. Bolt plate to boss with three 1/4-inch bolts and washers, and to structure with 16 bolts and washers.

(17) Install fuel probe in right top of cell.

(a) Position and install fuel probe with two 1/4-inch countersunk screws.

(b) Position 265-484099 gasket on probe flange and 265-484100 gasket on inside of mounting plate.

(c) Install plate to structure with 16 bolts and washers and two 1/4-inch bolts and washers.

(d) Hook up fuel quantity coaxial cables to probe.

(18) Install aft inner close-out door.

(a) Install "O" ring (180-948040-5) in open groove.

(b) Position inner access door and install sixteen 1/4-inch bolts and washers. Leave previously marked holes open.

(c) Install four support studs in marked holes. Slide a piece of lockwire through holes in end of studs (about 2 feet); double and twist ends. Install gasket on outer access panel.

(d) Position outer panel next to airplane structure; then pull the four loops of lockwire through matching holes in panel.

(e) Install panel with 53 bolts and washers.

(f) Pull on lockwire until studs come through holes, place nuts on studs; then tighten nuts. Remove safety wire.

(19) Install fuel vent line.

(20) Install "B" nuts on dual integral level control valve and check valves.

4. Replacement of Fuel Vent Nozzle and Flame Arrester Assembly

A. Removing Fuel Vent Nozzle and Flame Arrester Assembly

(1) Remove fuel vent plastic tail fairing.

(2) Remove six screws through mounting flange of vent nozzle.

(3) Slide nozzle assembly off jettison line*. Use care not to damage vent line.

(4) Remove gasket and check condition for re-use.

(5) Close opening with clean cloth or paper; do not use tape, because removal of tape can generate a spark and cause an explosion.

B. Installing Fuel Vent Nozzle and Flame Arrester Assembly

(1) Position gasket over nozzle mounting flange with one 5/16 hole to top and the other two over existing screw heads.

(2) Slide nozzle assembly into position so attach holes match gasket.

(3) Install 1/4-inch screw and washer in top hole and No. 10 screws with washers in remaining holes. Tighten screws evenly.

(4) Torque 1/4-inch screw to 50-70 inch-pounds. Torque No. 10 screws to 20-25 inch-pounds.

(5) Install fuel vent plastic tail fairing. Use twelve 7S15-10-11 screws. Draw up snugly.

*NA-370-2 through -6, -8 and -9

1. Maintaining Fuselage Fuel Tank

NOTES:
- Goodyear bladder type fuel cells made of BTC-39 material cannot be repaired.
- Nylon sandwich material (5200/5187, 5063, or P.F. 10056) is used for all inside fuel cell repair.
- Repairs to fuel cells with nylon sandwich material (5200/5187) do not require a cover patch.
- Nylon sandwich material (5063 or P.F. 10056) used as repair material on a fuel cell containing a fabric liner require a cover patch of nylon fabric (5188 or equivalent).
- Fuel cells containing a gum rubber inner liner do not require a cover patch.

A. Materials Required for Repair of Bladder-type Fuel Cells

Table 1. Bladder-type Fuel Cells Repair Materials

MFR PART NO.	DESCRIPTION
5200/5187 US Rubber Co.	Sheet - nylon sandwich, cured both sides, inside repair patching material
5188 US Rubber Co. (or equivalent)	Sheet - nylon fabric, cured both sides, outside repair patching material
5063 US Rubber Co. or P.F. 10056 Firestone Tire & Rubber Co. (or equivalent)	Sheet - Buna-N, nylon sandwich, cured both sides (for self-sealing cells), inside repair material
EC-678 Minnesota Mining & Mfg. Co	Cement - aromatic, fuel-resistant (Specification MIL-A-9117) One quart
MEK	Solvent
No. 80	Cloth - abrasive aluminum oxide

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B. Repairing Fuel Cells

CAUTION: No repairs are to be made on the radius of a cell or in the fitting area of a cell. No damaged areas, such as cuts and tears, larger than one inch are to be repaired in the field.

(1) Repair cuts in fuel cells while cell is installed in airplane, if access to damage area can be obtained through access door or by removal of a cell unit.

(2) Drain fuel supply system; then, with a cloth, dry cell surface around damaged area.

(3) Lightly and thoroughly buff a piece of cured US Rubber Co 5200/5187 nylon sandwich material large enough to cover damage at least 2 in. from cut in any direction.

CAUTION: Use methyl ethyl ketone sparingly. Excessive use will damage sandwich material.

(4) Wash buffing dust off with methyl ethyl ketone solution.

(5) Cement buffed side of patch with two coats of EC-678 black rubber cement and allow each coat to dry 10 to 15 minutes.

CAUTION: Use methyl ethyl ketone sparingly. Excessive use will damage sandwich material

(6) Buff cell area to be patched lightly and thoroughly with abrasive cloth No. 60; then wash off buffing dust with methyl ethyl ketone solution.

(7) Coat buffed area with two coats of EC-678 black rubber cement and allow each coat to dry 10 to 15 minutes.

(8) Freshen cemented area of patch and cemented area of cell with methyl ethyl ketone solution.

(9) While area is still tacky (touch with knuckle of hand to check for tackiness), apply edge of patch to edge of cemented area, centering patch over cut in cell.

(10) With a roller or blunt instrument, roll or press patch to cemented area on cell. Hold part of patch off cemented area, and roll or press it down one-half to one in. across at a time, so as not to trap air between patch and cell.

(11) Apply a suitable weight (shot bag) of about 35 lbs to repaired area, and do not disturb it for 6 hours.

(12) Seal patch one-half in. from edge of patch with two coats of EC-678 rubber cement. Allow first coat to dry one hour or more. Allow patch to set 24 hours before fueling.

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2. Maintaining Integral Wing Fuel Tanks

A. General Precautions

(1) Always remember that cleanness is required whenever maintenance work is being performed on the interior of the integral wing fuel tank.

(2) Screws and grooves must be degreased (with trichlorethylene).

(3) Chips or other foreign matter, even though very small may hold apart the sealing elements (including rubber-type seals), causing leaks, and may also clog screens in the fuel lines and interfere with the operation of the flapper check valves.

(4) All available injection holes in an area to be reinjected should be used, and in proper sequence. Attempting to force sealant through too long a portion of a groove can result in side extrusion.

(5) Where there is internal access, resealed areas should be inspected. Extrusion of excess sealant into the tank interior can cause the fuel system to work incorrectly by clogging valves and screens.

(6) Fasteners should not be turned on Stat-O-Seals. A washer should always be used between the bolthead and seal.

(7) Careless insertion of screws through Stat-O-Seals causes threads to serve as a rasp.

(8) Stat-O-Seals used with access door bolts should be replaced after several door removals. Stat-O-Seals should be replaced sooner if damaged.

(9) Sealed fasteners in the integral wing fuel tank area should be retorqued periodically.

(10) Do not attempt to inject sealant into groove-sealed areas when temperature of wing structure is less than about 45°F. Resealing is easiest at skin temperatures above 65°F. This restriction does not apply to individually sealed screws. They my be sealed readily at any temperature.

(11) If organic coating inside integral wing tank area peels or becomes scratched, refer to procedure for applying sealing compound to interior surfaces of integral wing tanks.

(12) When loosening a skin fastener, know which type of nut secures the fastener. Do not begin to back the fastener out until you know how to identify nut. For nut identification, refer to procedure for identifying hidden wing skin nuts.

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B. Identifying Hidden Wing Skin Nuts

Nut plates, AN365 nuts, and bumper nuts are used to secure the skin fasteners. Because the individually sealed fasteners must be backed off for the installation of an O-ring, it is necessary to identify the nuts. The type of nut can usually be determined during the first part of a turn of the fastener. If the fastener is secured with a nut plate, it will start turning with a sudden breakaway, and it will rise from the hole as it is turned. If it does not break and rise, it may be secured with either a bumper nut or an AN365 nut.

NOTE: Do not turn fastener more than two turns if it does not begin to rise.

C. Wing Sealing Procedures (See figure 28-10-801.)

D. Identifying Leaks and Stains (See figure 28-10-803.)

E. Stopping Leaks

Leaks through the structure of the integral wing fuel tanks will appear through the two different types of seals which are used: mechanical-rubber Stat-O-Seals and Gask-O-Seals, and nonhardening sealant. Leakage through the first type of seal occurs around bolts sealed with Stat-O-Seals, or around Gask-O-Seal access doors. These leaks can be stopped by retorquing the fasteners, or by replacing the Stat-O-Seal washers or Gask-O-Seal doors.

Leakage through nonhardening sealants can usually be stopped by tightening the screws or bolts. This is always the first step. If tightening does not stop the leak, reinjection of sealant is necessary. To reinject sealant, first find whether the leak is through a groove-type seal or through an individual screw seal; then refer to applicable reinjection procedure.

F. Replacing Torq-Set Screws

Torq-Set screws are used in many locations in the wing structure. Most of these screws have a standard diameter. However, in some location, screws having either a 0.015-or a 0.031-in. oversize shank are used. These are not easily identified, except by a part number.

(1) The following table shows the thread size and the part numbers for standard diameter and oversize Torq-Set screws. The part number of each screw is stamped on the head.

(2) Oversize screws must be replaced with oversize screws of the same diameter. Using a standard screw in place of an oversize screw will result in weakened structure. In addition, if the screw is used in a sealed area, a poor seal will result.

(3) Because these screws are similar in appearance, the only sure method of identification is the number of the head. Screws should be checked carefully before being replaced.

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PART NUMBERS FOR OVERSIZE DIAMETER
HIGH-STRENGTH FLUSH BOLTS

THREAD	STD DIA	OVERSIZE SHANK	
		0.015 IN.	0.031 IN.
10-32 NF-3A	7S34-3	7S34-31	7S34-32
1/4-28 UNF-3A	7S34-4	7S34-41	7S34-42
5/16-24 UNF-3A	7S34-5	7S34-51	7S34-52
3/8-24 UNF-3A	7S34-6	7S34-61	7S34-62
7/16-20 UNF-3A	7S34-7	7S34-71	7S34-72
1/20 UNF-3A	7S34-8	7S34-81	7S34-82
9/16-18 UNF-3A	7S34-9	7S34-91	7S34-92

Length code: Last dash number indicates nominal grip length in 16ths.

Example of part numbers: 7S34-41-10 indicates a 1/4-28 Torq-Set recess screw with 0.015-inch oversize shank and 5/18-inch nominal grip length.

G. Repetitive Leaks Around Individually Sealed Screws

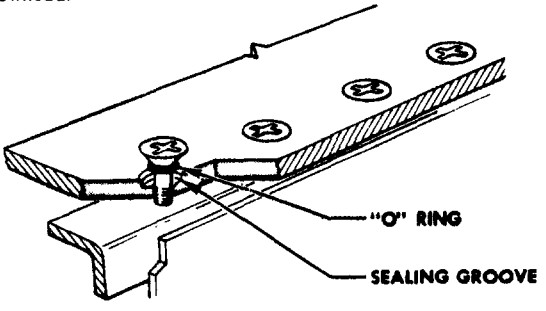
It is possible that stains will reappear at an individually sealed screw, even when torquing has been done properly. If repetitive stains develop, the screw might be too long, the countersink might be elongated, or the screw might be tilted in relation to the countersink. Check for cause of repetitive leaks and repair. First, back out screw and inspect threads for signs of bottoming on nut. If threads show such signs, replace screw with one of proper length. If screw is of proper length, inspect countersink for damage or elongation. With an elongated countersink, a gap might show at one side of screwhead. Such a countersink should be reworked. If countersink is not damaged or elongated, check whether it is aligned with the screw. This condition can be identified by a tilted screwhead. Also, a gap might appear between the head and the countersink until just before screw is fully tightened. A new countersink should be made.

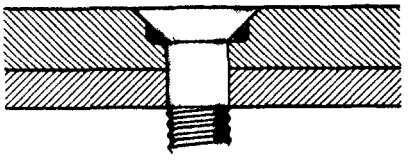
H. Loading Groove-sealant Injection Gun

The groove-sealant injection guns are air-pressure-operated and hand-operated. The guns should be loaded so that no air is trapped in the sealing compound. Precautions must be taken so that no grease or oil will contact the sealant.

INDIVIDUAL SCREW SEALING

1 Partially back out screw. Do not completely disengage screw from nut. Install new "O" ring by stretching it very carefully over screwhead and positioning it on screw shank, adjacent to screwhead.

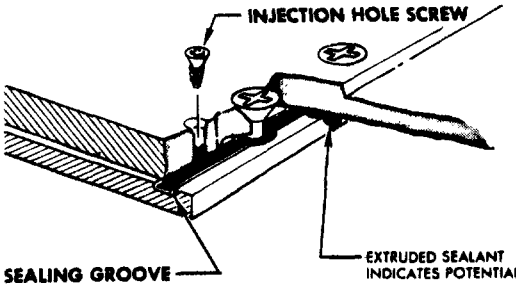


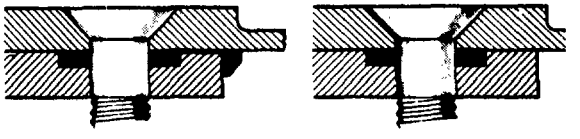


2 Tighten screw, and torque to required torque value.

GROOVE-TYPE SEALING — BAD SEALING — GOOD SEALING

NOTE Sealant squeezing out from between skin and spar or rib flange indicates a poor groove-type seal. To correct this condition, follow steps 3 through 5.





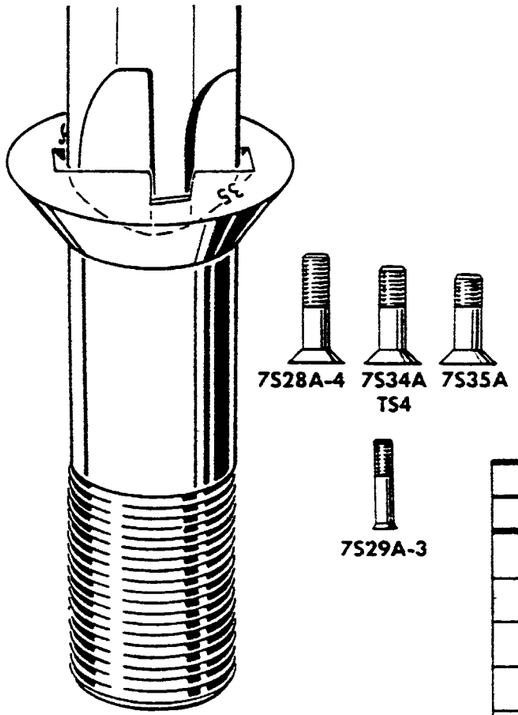
3 Clean off extruded sealant.

4 Torque nearby fasteners.

5 Remove screws from injection holes in that area, inject sealant again, and replace screws in injection holes.

5370-2-48-7

TORQ-SET FASTENERS SOCKET ADAPTER AND DRIVER BIT



7S28A-4 7S34A TS4 7S35A

7S29A-3

NOTE The 7S35 fastener has shorter threaded length to save weight and is used with low-profile nuts.

TORQ-SET FASTENER TORQUE

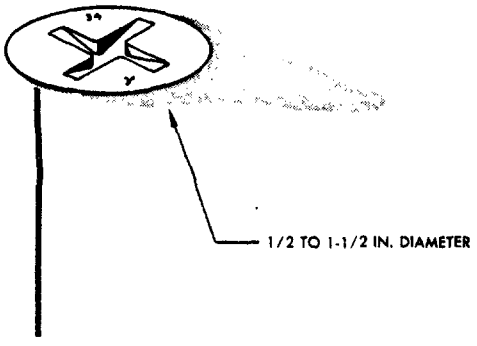
This is a Torq-Set and high-strength bolt torque table for 100-degree flush bolts (heat-treated to 160,000-180,000 psi). Torque values in the table apply to all nuts except the shear type. The higher values should be used when torquing heads of bolts; however, these values should not be exceeded by more than 10 percent.

FASTENER		SIZE	DRIVER BIT	TORQUE IN. LB
COML NO.	NAA STD			
TS3	7S34-3 7S35-3	10-32	EX-170-3/16	25-35
TS4	7S34-4 7S35-4	1/4-28	EX-170-1/4	70-90
TS5	7S34-5 7S35-5	5/16-24	EX-170-5/16	140-203
TS6	7S34-6 7S35-6	3/8-24	EX-170-3/8	190-351
TS7	7S34-7 7S35-7	7/16-20	EX-170-7/16	500-756
TS8	7S34-8 7S35-8	1/2-20	EX-170-1/2	690-990
TS9	7S34-9 7S35-9	9/16-18	EX-170-9/16	1000-1440

S370-2-48-8

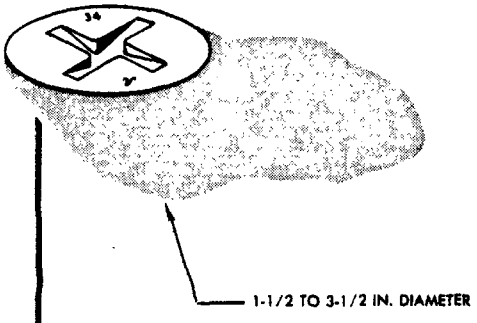
ORIGINAL
As Received By
ATP

CLASS A STAIN



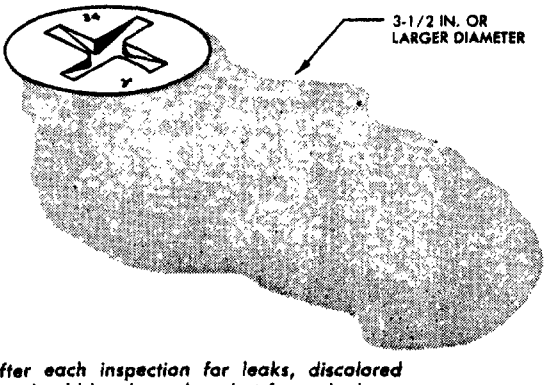
CLASS B STAIN

• Class A or Class B stains should be recorded but need not be stopped until airplane is grounded for other maintenance.



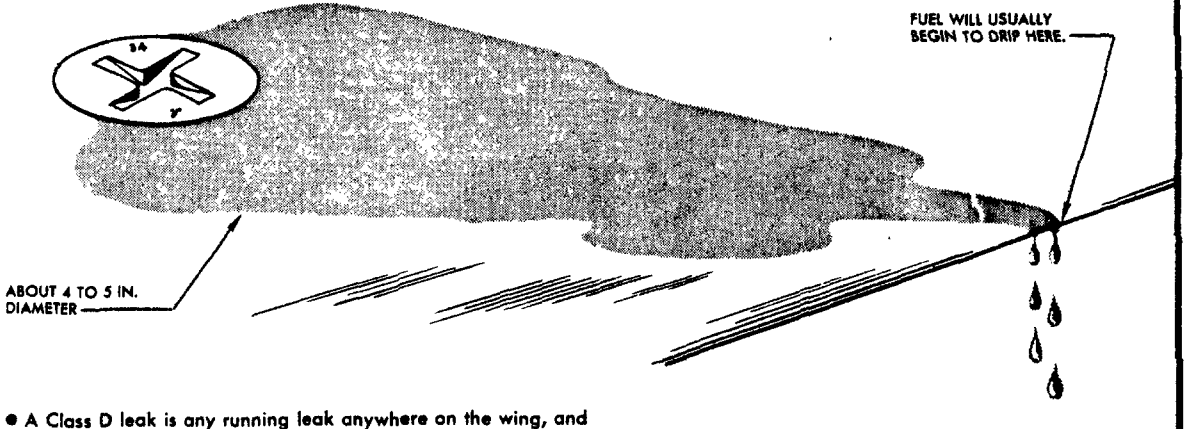
CLASS C HEAVY SEEP

• A Class C seep on the outer wing panels should be recorded and need not be stopped, but a Class C seep within any enclosed area or in an area that will allow seepage into an enclosed area can create a trapped fumes condition. This condition is cause for immediate grounding of the airplane until a satisfactory repair is made.



• *After each inspection for leaks, discolored* areas should be cleaned so that future leakage can be evaluated accurately.

CLASS D LEAK



• A Class D leak is any running leak anywhere on the wing, and is cause for grounding the airplane until a satisfactory repair is made.

5370-2-48-9

Wing Leak Classification
Figure 28-10-803

MAINTENANCE PRACTICES (Continued)

1. If it is necessary to clean gun, wash it in toluol.
2. Remove loading chamber from gun barrel; depress loading piston to bottomed condition.
3. Roll groove-sealant (DC94-011) into a cylindrical plug about 1-3/4 to 2 inches in diameter.

CAUTION

There is no substitute for sealant DC 94-011. The use of curing-type sealants makes repair by the injection method impossible in case of future leakage.

4. Insert cylindrical plug of sealant into loading chamber; tamp compound with a wooden stick. Fill nozzle pump casting with compound, leaving threads exposed for reassembly.
5. Reassemble gun.
6. When the air-pressure-operated injection gun is used, the air pressure should be 100 (+25, -10) psi.

Injecting Sealant Into Grooves

Injection of sealant into groove-sealed areas should not be attempted when the temperature of the wing structure itself is below 40°F. Whenever possible, it is recommended that the temperature of the structure be at least 77°F. Injection is easier at higher temperatures. Ease of injection may further be increased by heating the loaded gun to a temperature not greater than 110°F. However, it is emphasized that a controlled method of heating must be used, because high temperatures are harmful to the sealant.

1. Load groove-sealant injection gun.
2. Remove injection hole screws where necessary.

NOTE

When injecting sealant into the tank grooves, the injection holes to either side of the tip should be unplugged.

3. Retorque all fasteners in area to be sealed.
4. Place nozzle tip of gun into countersink of an injection hole. Hold gun firmly in position and depress trigger for about 30 seconds. Release trigger and allow piston to return. (Piston return will take at least 5 seconds.) Continue sequence until compound flows out of adjacent injection holes for a minimum distance of 1/8 inch above skin surface.
5. Replace injection screw in hole in which injection gun has been inserted.
6. Move gun tip to adjoining hole through which compound has been forced, and repeat steps 4 and 5.
7. Continue injecting along groove until area has been sealed.
8. After final installation of injection hole screws, scrape away most of excess groove sealant with wood or a plastic blade; remove remainder with toluol (Federal Specification TT-T-548) or cleaning solvent. Do not use toluol on any surfaces coated with corrosion resistant finish.

Sealing Wing Forward Spar Splice Plates

The right or left wing forward spar splice plates can be resealed while the wing is attached to the fuselage. This may be necessary if a fuel leak should develop in this area. The airplane should be in a hangar long enough to allow the airplane temperature to be the same as the hangar temperature. The higher the airframe temperature, the easier it is to force out the old sealant and replace it with new sealant.

1. Remove center floorboard access panel.
2. Remove lower access door.
3. Remove a sealant screw from top and bottom skin at outboard edge of wing splice plate.
4. Load groove-sealant gun. (Refer to procedure for loading groove-sealant injection gun.) Keeping the loaded groove-sealant injection gun heated to the maximum temperature will help to inject the sealant.
5. Place nozzle tip of gun into countersink of injection hole in skin at outboard edge of wing splice plate. Hold gun firmly in position and depress trigger for about 30 seconds. Release trigger and allow piston to return to its original position. (Piston return will take at least 5 seconds.) Continue sequence until all of the old compound is forced out of bottom hole.
6. Replace sealant screws removed in step 3. Using a wooden or plastic blade, remove excess groove sealant; then use toluol or cleaning solvent to remove remainder of sealant. Do not use toluol on any surfaces coated with corrosion-resistant finish.
7. Install lower access door.
8. Install center floorboard access panel.

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MAINTENANCE PRACTICES (Continued)

Rework of Individually Sealed Screws

All individually sealed screws on the wing are sealed with "O" rings. The "O" rings may be replaced in the following manner:

1. Partially back out affected screw. Do not completely disengage screw from nut. This is especially true where bumper nuts are used, since the nut may drop into the tank.
2. Remove old or leaking O-ring by cutting it with a knife or by stretching it over screwhead.
3. Make sure countersink and counterbore area in skin is thoroughly clean and absolutely free of all chips and foreign matter.
4. Install a new O-ring seal by stretching it very carefully over screwhead and positioning it on screw shank adjacent to screwhead.
5. Tighten screw and torque to required torque value (see figure 28-10-802.)

CAUTION

Use caution not to cut or damage o-ring seal. Damaged o-ring seals must be replaced with unused o-ring seals.

Substitute Fasteners for Inaccessible Areas of Sealed Wing

The restricted use of Jo-bolts is approved in repair procedures. It must be noted that the areas of use and the number of substitute fasteners in given areas are limited. If a substitution is to be made where channel nuts are installed, special precautions are necessary to ensure a good seal and solid fastening.

Applying Sealing Compound to Interior Surfaces of Integral Wing Tanks

When work is done inside the wing, the coating on the interior surfaces may become peeled or scratched. When this happens, the damaged coating should be touched up.

1. Wipe damaged area with clean cheesecloth dampened with cleaning solvent.
2. Immediately wipe solvent off with another clean, dry cheesecloth.
3. Where bare metal has been exposed by more than just a scratch, apply chemical film treatment to bared surface.
4. Apply sealing compound to area.
5. If Topcoating is to be used, apply two coats, allowing a minimum of 20 minutes between coats. Use PR-1005L for Topcoating.

NOTE

No coating should be applied to any surface sealed with gaskets or O-rings. Allow joining surfaces to air-dry a minimum of 30 minutes before joining.

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FUEL SYSTEM DISTRIBUTION - DESCRIPTION AND OPERATION

1. Fuel Control Systems

The fuel control systems consist of the fuel boost pump system, the fuel cross-feed system, the suction-feed fuel supply, and the fuel transfer system and their components.

A. Fuel Boost Pump System

The fuel supply for the engines is normally delivered to the engine-driven fuel pumps by two fuel boost pumps. These pumps are positioned on the rear spar in the wing center secion, and are single-speed, side-mounted, submerged units, driven by a 28-volt dc motor.

Two jet-type fuel transfer ejectors, one in the forward left wing center compartment and one in the right wing forward center compartment, utilize fuel bleed flow from their respective boost pump as a motivating power to transfer fuel from the low point in their areas to the boost pump sump compartment. The pumps, thereby, maintain a high fuel level in the boost pump sump compartments in airplane nose-down attitudes and when fuel supply is low.

The boost pumps are powered through the "BOOST PUMP - LH FUEL" and the "BOOST PUMP - RH FUEL" circuit breakers on the 28-volt dc essential bus. Two fail-safe relays, left fuel boost pump and right fuel boost pump, control the power input to the boost pumps. The left boost pump normally supplies fuel to the left engine-driven fuel pump, and the right boost pump normally supplies fuel to the right engine-driven fuel pump. However, if a malfunction causes either boost pump to cease operating, the cross-feed system should be placed in operation, allowing either boost pump to supply fuel to both engine-driven fuel pumps.

B. Fuel Cross-feed System

The fuel cross-feed system is used to balance the fuel load in the wing tanks. The cross-feed system has two fuel valves (tank cross-feed valve and boost cross-feed valve) and the necessary relays and switches to actuate the valves. The system is also tied into the boost pump relay circuit, allowing the deactivation of either boost pump when wing fuel balance is desired.

The fuel cross-feed system is powered by the 28-volt dc essential bus controlled by a four-position "FUEL" (mode selector) switch. The mode selector switch controls the four functions of the fuel cross-feed system. The four functions of the fuel cross-feed system are normal fuel supply, cross-feed fuel supply, right-hand tank fuel supply, and left-hand tank fuel supply. The following paragraphs discuss the operation of these functions.

(1) Normal Fuel Supply

Normal operation of the cross-feed system is accomplished by positioning the mode selector switch at NORMAL. With the switch at NORMAL, no control circuits to the cross-feed valve circuitry are energized, both cross-feed valves are closed, and all warning and caution lights are off. In this mode, each engine is supplied with fuel from its respective boost pump and wing tank, independent of the other engine, boost pump, and wing tank.

(2) Cross-feed Fuel Supply

Cross-feed is accomplished by positioning the mode selector switch at X-FEED. This routes 28-volt dc power from the "FUEL-X-FEED - BOOST" circuit breaker to energize the fuel cross-feed relay (FCF). The FCF relay breaks a circuit through the tank and boost crossfeed valves closed limit switches to the tank cross-feed closed (TCC) and pump cross-feed closed (PCC) relays, causing the relays to de-energize. The FCF relay, at the same time, completes two circuits. The first circuit completed, when the FCF relay is energized, routes power from the "FUEL-X-FEED - TANK" circuit breaker to open the tank cross-feed valve and to energize the tank cross-feed warning (TCW) relay. The TCW relay completes a circuit from the "FUEL WARNING - X-FEED" circuit breaker through normally closed contacts in the tank cross-feed opened (TCO) relay to the "FUEL TANK X-FD FAIL" caution light, causing the light to come on. The second circuit completed. when the FCF relay is energized, routes power from the "FUEL-X-FEED - BOOST" circuit breaker through normally open contacts in the left jettison fuel control (LJFC),* left tank selector (LTS), and right tank selector (RTS) relays to the open winding of the boost cross-feed valve motor, causing the valve to open, and to the pump cross-feed warning (PCW) relay. The PCW relay energizes, connecting power from the "FUEL WARNING - X-FEED" circuit breaker through normally closed contacts in the pump cross-feed opened (PCO) relay to the "FUEL PUMP X-FD FAIL" caution light, causing the light to come on.

When the tank cross-feed valve is fully open, limit switches in the valve are actuated. The open limit switch opens the power circuit to the open winding of the valve motor and closes a circuit to energize the TCO relay. The TCO relay then breaks the circuit to the "FUEL TANK X-FD FAIL" caution light, causing the light to go off. The closed limit switch arms the circuit to the closed winding of the valve motor and opens a circuit to de-energize the TCC relay. When the boost cross-feed valve is fully open, limit switches in the valve are actuated. The open limit switch breaks the circuit to the open winding of the valve motor and closes a circuit to energize the pump cross-feed opened (PCO) relay. The PCO relay breaks the circuit to the "FUEL PUMP X-FD FAIL" caution light, causing the light to go off, and closes a circuit from the "FUEL WARNING - JETTISON"

*NA-370-2 through -6, -8, and -9 airplanes

circuit breaker on NA-370-2 through -6, -8, and -9 airplanes "FUEL WARNING SELECT" circuit breaker on NA-380-1 and later airplanes, through the closed contacts in the right fuel boost monitor (RFBM) relay and left fuel boost monitor (LFBM) relay to the "FUEL SEL SW NOT NORMAL" caution light, causing the light to come on. The closed limit switch arms the circuit to the closed winding of the valve motor and breaks the circuit to de-energize the PCC relay.

In the cross-feed mode, both cross-feed valves are open, allowing fuel to balance between the two wing tanks and allowing both boost

pumps to share in the supply of fuel to both engines. The fuel supply system in this mode is, in effect, a single common fuel supply with two boost pumps supplying fuel to both engines.

(3) Right-hand Tank Fuel Supply

Right-hand tank fuel supply to the engines is accomplished by positioning the mode selector switch at RH TANK. This routes power from the "FUEL-X-FEED - BOOST" circuit breaker to energize the right tank selector (RTS) relay. One set of contacts in the RTS relay closes to route power from the "FUEL-X-FEED - BOOST" circuit breaker through normally closed contacts in the LJFC,* RJFC,* and LTS relays to the open windings of the boost cross-feed valve motor, causing the valve to open, and to the PCW relay, causing the relay to energize. The PCW relay routes power from the "FUEL WARNING - X-FEED" circuit breaker through normally closed contacts in the PCO relay to the "FUEL PUMP X-FD FAIL" caution light, causing the light to come on. When the boost cross-feed valve reaches the full open position, limit switches in the valve are actuated. The close limit switch breaks a circuit to the PCC relay, causing the relay to de-energize, and at the same time, arms the circuit to the close winding of the boost cross-feed valve motor. The open limit switch breaks the circuit to the open winding of the valve motor and completes a circuit, routing power from the "FUEL WARNING - JETTISON" circuit breaker through closed contacts in the RTS, LJFC,* and RJFC* relays to energize the left-hand fuel boost (LHFB) relay. The LHFB relay energizes and breaks the boost pump circuit, shutting the left boost pump off, and completes a circuit to energize the left fuel boost monitor (LFBM) relay. The LFBM relay completes a circuit from the "FUEL WARNING - JETTISON" circuit breaker on NA-370-2 through -6, -8, and -9 airplanes, or "FUEL WARNING SELECT" circuit breaker on NA-380-1 and later airplanes, through closed contacts in the PCO, TCC, RFBM, and RTS relays to the "FUEL SEL SW NOT NORMAL" caution light, causing the light to come on. In this mode of operation, the right wing tank and boost pump supply all of the fuel to both engines.

(4) Left-hand Tank Fuel Supply

The left-hand tank fuel supply mode is accomplished by positioning the mode selector switch at LH TANK. This mode of operation is the same as described for the right-hand tank fuel supply, except that the right tank selector (RTS) relay, the left fuel boost monitor (LFBM) relay, and the right-hand fuel boost (RHFB) relay are the controlling relays instead of the LTS, RFBM, and LHFB relays. This mode of operation opens the pump cross-feed valve, shuts off the right boost pump, and allows the left wing tank and boost pump to supply all of the fuel to the engines.

*NA-370-2 through -6, -8, and -9 airplanes

C. Suction-feed Fuel Supply

When both fuel boost pumps fail, fuel is delivered to the engine-driven fuel pumps by suction feed. Fuel transfers from the fuselage fuel tank to the right and left integral wing tanks. The wing fuel transfers by gravity flow through check valves in the wing ribs into the right and left center sections and into the boost pump sump area. One-way check valves in the boost pump, downstream of the boost pump impeller, allow fuel to be fed by suction to the two engine-driven fuel pumps.

D. Fuel Transfer System

There is no manual fuel tank sequencing transfer system; however, the gravity fuel transfer arrangement of the fuel system ensures that all the fuel in the fuselage tank (when the fuselage tank is used) will be used before the fuel in the wing tank is used, because the fuselage tank is higher than the wing tanks and because the vent exit is submerged, preventing venting of the wing tanks until the fuselage tank is emptied. Transfer of fuel from the wing tanks to the fuselage tank is prevented (under normal conditions) by two spring-loaded (1-1/4 psi) check valves. However, during extreme airplane nose-up attitudes, wing tank vent valve failure, or any other situation in which the wing tank pressure exceeds the fuselage tank pressure by 1-1/4 psi, fuel transfer to the fuselage tank can occur to relieve the pressure in the wing tanks.

Wing dihedral creates a gravity flow from the outer wing panels to the right and left center sections and booster pump sump areas in a normal manner up to a 9-degree nose-up attitude. Wing ribs at wing stations 37, 106, and 171.6 prevent excessive outboard fuel flow during airplane maneuvers. Fuel transfers inboard into the right and left center sections through three 1-1/4-inch diameter check valves in each rib bulkhead, and into the right and left boost pump sump areas through two check valves in each bulkhead. Air-load rib lightening holes and wing skin stringer cutouts allow gravity flow within the compartments. Fuel in the center sections transfers by gravity flow, through six check valves in each boost pump sump bulkhead, into the boost pump sump areas.

Two jet-type fuel ejectors, powered by the boost pumps, assist in transferring fuel from the right and left center sections to the right and left boost pump sumps. These ejectors function as scavenge pumps, scavenging fuel from the low points in the right and left center sections.

2. Fuel Boost Pumps

Two 28-volt dc, single-speed, motor-driven, submerged, centrifugal-type, side-mounted pumps supply fuel under pressure to two engine-driven fuel pumps. The boost pumps are positioned on the aft bulkhead of the wing center section. The right boost pump is accessible through an access plate on the rear wing spar in the right wheel well, and the left boost pump is accessible through an access plate on the rear wing spar in the left wheel well. The boost pumps run continuously (power on) as long as the respective master switch (left or right) is on, the respective throttle (left or right) is

moved from the off position, and the "FUEL" switch is at NORMAL or -X-FEED. When the "FUEL" switch is placed at RH TANK or LH TANK, the opposite boost pump is shut off. This allows the pilot to balance the fuel load in the wing tanks. The right and left boost pump circuit breakers are on the left overhead forward dc essential circuit-breaker panel. The pumps may develop 17 to 30 psi static pressure. The allowable leakage from the drive shaft seal drain is 2 cubic centimeters per hour. The pumps operate normally on a 25- to 29-volt dc power source but will operate on a reduced voltage of 14-volt dc.

3. Fuel Transfer Ejectors

Two jet-type fuel ejectors assist the transfer of fuel from the right and left center sections to the right and left boost pump sumps. These tubular-shaped ejectors are in the forward, inboard section of their respective tank areas, right and left center sections. The ejector consists of a 3/4-inch tube, a screened inlet to pick up fuel, and a jet nozzle to discharge a pressurized flow of fuel down the center of the tube. A fuel line, connected to the boost pump, receives fuel bleed flow from the pump. This line routes fuel pressure to the jet nozzles. The fuel bleed flow from the boost pumps is forced through the jet nozzle, propelling the stream of fuel through the tube and connecting lines to the boost pump sump. This pressurized flow pulls fuel under the ejector inlet into the ejector tube, establishing a flow to the boost pump sump. The ejector, then, serves as a scavenge pump. Access to these ejectors is through access panels in the front spar at about wing station 30.

4. Fuel Shutoff Valves

There are two motor-driven, one-inch gate-type main fuel shutoff valves on top of the center wing section, forward of the main wheel wells. These valves are the main fuel shutoff valves. Access to these valves is through a removable portion of the cabin floor. An indicator pin on the valves shows the position of the gate. Internal limit switches within each valve open the circuit to the motor to stop it when the valve reaches a fully open or closed position and energize a caution light indicating system if the valve does not run to the position selected by the "LH ENGINE - ENGINE MASTER" and "RH ENGINE - ENGINE MASTER" switches or does not close when the associated "FIRE PULL" handle is pulled out. Normal operating time is about one second to open or close at 26-volt dc and 70°F; however, the valves will operate on 18- to 30-volt dc.

When power is applied to the essential bus, the "RH ENGINE - MASTER," "LH ENGINE - MASTER," and "FUEL WARNING - SHUT OFF" circuit breakers are engaged, and the "LH ENGINE - ENGINE MASTER" and "RH ENGINE - ENGINE MASTER" switches are positioned at OFF, the right and left fuel valve warning and right and left fuel valve closed relays are energized. Therefore, the fuel shutoff valves remain closed and the "FUEL SHUT OFF FAIL" caution light will not illuminate. (See figure 28-00-2.)

Positioning the "LH ENGINE - ENGINE MASTER" and "RH ENGINE - ENGINE MASTER" switches at ON opens the shutoff valves and de-energizes the right and left

fuel valve warning and right and left fuel valve closed relays. This causes the "FUEL SHUT OFF FAIL" light to light. When the valves are motored fully open, the right and left fuel valve open relays are energized and the "FUEL SHUT OFF FAIL" light goes out. If the light remains lighted, either valve or possibly both valves did not fully open.

When the "LH ENGINE - ENGINE MASTER" and "RH ENGINE - ENGINE MASTER" switches are positioned at OFF, the right and left fuel valve warning relays are energized and the right and left valve closed relays are de-energized. This causes the "FUEL SHUT OFF FAIL" light to light. When the valves are motored fully closed, the right and left fuel valve open relays are energized open; then the "FUEL SHUT OFF FAIL" light goes out. If the light remains lighted, either valve or possibly both valves did not close completely.

If either the right or left engine fire wall shutoff switch "FIRE PULL" handle is pulled, the respective engine emergency fire shutdown relay is energized and overrides the engine master switch control of the respective fuel shutoff valve. The fuel shutoff valve will close and the "FUEL SHUT OFF FAIL" light will react in exactly the same manner as it did when the respective engine master switch was moved to OFF.

5. Fuel Cross-feed Valves

A. Tank Cross-feed Valve

The tank cross-feed valve is a 28-volt dc motor-driven, 2-inch gate-type valve, mounted on the rib that separates the right and left boost pump sump areas of the right and left wing center sections. This allows either boost pump to pump fuel from both wing tanks. The valve should open or close in about one second and operate satisfactorily on 18- to 30-volt dc. When this valve is actuated open, fuel can flow by gravity from one fuel tank to the other. This directly exposes the fuel level of one tank to the fuel level of the other tank without the effect of the higher head of fuel in the completely full booster pump compartment of the operating pump.

B. Boost Cross-feed Valve

The boost cross-feed valve is a 28-volt dc motor-driven, one-inch gate-type valve, mounted on the top of the wing center section between the cabin floor and the top of the center section. This valve should also open or close in about one second and operate satisfactorily on 18- to 30-volt dc. When this valve is actuated open, either boost pump (right or left) can supply fuel to both engine-driven fuel pumps. Internal limit switches within these valves open the circuit to the motor when the valve reaches a fully opened or fully closed position and, if the valve does not run to the position selected by the cross-feed switch or the jettison system, on NA-370-2 through -6, -8, and -9 airplanes, will energize a caution light indicating system.

6. Fuel Heater - NA-370-1 and -7, NA-380 Airplanes

Fuel heaters are installed in recessed areas in each pylon. Fuel for each engine is routed through its respective fuel heater before going through the engine fuel filter and then to the engine. Engine fuel is heated to eliminate the possibility of ice crystals forming in the fuel filter. The temperature of the fuel flowing through the heater is sensed internally within the heater. When the fuel temperature reaches 35°F, the thermal valve within the heater opens and allows engine bleed air to circulate through the fuel heater manifold and exhaust overboard through a cutout in the pylon lower skin at the forward end of the fuel heater recess. There are no cockpit switches to control the fuel heater operation, it is totally automatic dependent on the temperature of the fuel.

FUEL SYSTEM DISTRIBUTION - TROUBLE SHOOTING

1. Trouble Shooting Fuel Control System

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. UNSTEADY OR ERRATIC FUEL BOOST PUMP PRESSURE.		
(1) Loose connection.	Run continuity check.	Repair as needed.
(2) Voltage variation.	Run voltage check.	Repair or replace as necessary.
(3) Seizing bearing.		Replace pump.
(4) Poor commutation.		Replace pump.
(5) Foreign material in line.	Make obstruction check.	Remove foreign material.
B. FUEL FAILS TO TRANSFER FROM FUEL BOOST PUMP SUMP AREAS TO ENGINE-DRIVEN FUEL PUMPS.		
(1) Faulty fuel boost pump.	Check pump operation.	Replace pump.
(2) Faulty shutoff valve.	Check shutoff valves.	Replace shutoff valve.
(3) Faulty boost pump wiring	Run continuity check. Check voltage at terminals.	Correct wiring. Replace pump.
(4) Faulty boost pump relay.	Pump fails to operate. Use jumper around relay or try substitute relay.	Replace relay.
(5) "FIRE PULL" handle pulled.	Check visually.	Push "FIRE PULL" handle in.
C. FUEL BOOST PUMP PRESSURE TOO LOW.		
(1) Low voltage.	Check for 28-volt dc at boost pump.	Check power supply. Correct wiring or replace circuit breaker.
(2) Worn brushes.	Check for noisy operation (arcing sound). Check for fluctuation in current drain.	Repair or replace pump.
(3) Faulty line connector or faulty casting.	Make pressure check.	Tighten fitting or replace casting.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
(4) Brushes not seated properly after replacement.	Listen for high-pitched sound and too much noise.	Run pump under load until pressure reaches normal.
(5) Foreign material at pump inlet.	Make obstruction check.	Remove foreign material.
(6) Armature shaft binding.		Replace pump.
(7) Fuel in motor showing possible bearing failure.	Check for fuel leakage at vent plug.	Replace pump.
D. FUEL BOOST PUMPS FAIL TO OPERATE.		
(1) Faulty relay in circuit.	Make continuity check.	Replace relay.
(2) Pump brushes worn out.	High and intermittent current drain.	Repair or replace pump.
(3) Faulty or maladjusted engine fuel ignition switch (aft switch) in throttle quadrant. NOTE: Switch should open when throttle is advanced.	Make continuity check.	Adjust or replace switch.
(4) Loose connection, or high resistance in connection.	Run voltage check at terminals.	Clean and tighten all connections.
(5) Pump motor burned out.	Check for dead short in wiring.	Replace pump.
E. FUEL BOOST PUMP BURNED OUT.		
(1) Clogging of seal chamber drain line.	Visually check for open drain.	Replace pump.
(2) Faulty bearing, or failure of pump seals.	Rotate pump and listen for rough operation. Check leakage from drains.	Replace pump.
F. FUEL LEAKING FROM FUEL BOOST PUMP SEAL DRAIN.		
(1) Failure of seal or foreign object on seal.	Run pump for short time. Hold a short-looped wire to seal drain to cause fuel to form drop.	Replace pump.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
G. FUEL LEAKING FROM FUEL BOOST PUMP MOTOR VENT.		
(1) Fuel leaking into motor through motor cover gasket.	Check visually after removing pump.	Replace pump.
H. FUEL BOOST PUMP NOISY		
(1) Loose mounting.	Remove pump and check for loose mounting bolts.	Tighten mounting bolts and connecting lines.
(2) Worn bearing.	Rotate pump and feel for too much vibration.	Replace pump.
I. BOOST CROSS-FEED VALVE FAILS TO OPEN.		
(1) "FUEL-X-FEED - BOOST" circuit breaker defective.	Place all airplane switches at OFF or airplane-parked positions. Remove all external power from airplane. Remove left overhead forward dc essential circuit-breaker panel and engage "FUEL-X-FEED - BOOST" circuit breaker; then check continuity across circuit-breaker terminals.	Replace circuit breaker.
(2) Defective "FUEL" switch.	Check continuity of switch.	Replace "FUEL" switch.
(3) Defective boost cross-feed valve.	Check boost cross-feed valve operation.	Replace boost cross-feed valve.
(4) Defective relay.	Make continuity check to determine defective relay.	Replace defective relay.
J. TANK CROSS-FEED VALVE FAILS TO OPEN.		
(1) "FUEL-X-FEED - TANK" circuit breaker defective.	Place all airplane switches at OFF or airplane-parked positions.	Replace circuit breaker.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
	Remove all external power from airplane. Remove left overhead forward dc essential circuit-breaker panel and engage "FUEL-X-FEED - BOOST" circuit breaker; then check continuity across circuit-breaker terminals.	
(2) Defective "FUEL" switch.	Check continuity of switch.	Replace "FUEL" switch.
(3) Defective tank cross-feed valve.	Check tank cross-feed valve operation.	Replace tank cross-feed valve.
(4) Defective relay.	Make continuity check to determine defective relay.	Replace defective relay.
K. "FUEL SEL SW NOT NORMAL" CAUTION LIGHT NOT ON; FUEL PUMP OR FUEL TANK CROSS-FEED FAIL CAUTION LIGHT REMAINS ON.		
(1) Cross-feed valve fails to open fully.	Check cross-feed valve operation.	Replace cross-feed valve.
(2) Defective cross-feed valve limit switch.	Check that valve motor continues to run after valve is fully open. Check that circuit between cross-feed valve pins "A" and "C" remains open.	Replace cross-feed valve.
	Check that valve motor continues to run after valve is fully closed. Check that circuit between cross-feed valve pins "B" and "D" remains open.	Replace cross-feed valve.
L. "FUEL SEL SW NOT NORMAL" CAUTION LIGHT DOES NOT COME ON; FUEL PUMP AND FUEL TANK CROSS-FEED FAIL CAUTION LIGHTS OFF (FUEL MODE SELECTOR SWITCH AT RH TANK OR LH TANK).		
(1) Left or right fuel boost pump did not shut off.	Listen for sound of boost pump operation.	Replace LHFB or RHFB relay or repair relay circuits.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
(2) RFBM or LFBM relay faulty.	Check for 28-volt dc at pin "X1" of relay and check that pin "X2" is properly grounded.	Replace relay.
M. TOO MUCH FUEL REMAINS IN RIGHT OR LEFT CENTER WING SECTION AFTER FUEL SYSTEM HAS BEEN DEFUELED.		
(1) Improper equipment used to defuel airplane.	Check that defueling hose has a restrictor line (3/8-inch O.D. by 0.035 inch by 8 feet) to keep boost pump from transferring fuel faster than fuel is supplied to pump inlet.	If restrictor line is not available, jack up nose of airplane 8 to 9 degrees to provide gravity flow to boost pump sump.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
		CAUTION: REMEDY TO BE USED ONLY WHEN IT IS KNOWN THAT FUEL EJECTOR PUMPS ARE FUNCTIONING PROPERLY.
(2) Check valve in fuel transfer ejector line reversed.	Remove access plate on rear wing spar and inspect check valve for proper installation.	Install check valve properly.
(3) Boost pump fuel bleed flow line not installed properly.	Remove access plate on rear wing spar and inspect line for proper installation at slip joint or "B" nut.	Install line properly.
(4) Ejector outlet to boost pump sump area improperly installed.	Remove access plate on rear wing spar and inspect for proper installation of ejector outlet.	Install ejector to boost pump sump line properly.
(5) Ejector inlet screen covered.	Remove sump drain valve beneath ejector and check for foreign object.	Remove foreign object and check operation of ejector. (Refer to 28-20-3.)
(6) Fuel booster pump alternate ejector port cover plate not installed.	Remove boost pump access door and check that cover plate is installed.	Install ejector port cover plate.
(7) Cap not installed on inboard "T" fitting of fuel bleed line check valve.	Remove boost pump access door and check that fuel line cap is installed.	Install fuel line cap.
(8) Blocked induced flow line.	Remove boost pump access door and check for plugged line by slipping a piece of hose over end of line and blowing into hose.	If line is plugged, disconnect line at ejector and blow air and fuel through line until obstruction is removed. (Refer to 28-20-2.)
(9) Ejector nozzles plugged.	Remove boost pump and ejector pump access doors and check for plugged nozzles.	Clear plugged nozzles. (Refer to 28-20-2.)

FUEL SYSTEM DISTRIBUTION

MAINTENANCE PRACTICES

Replacement of Fuel Boost Pumps

Removing Fuel Boost Pump

NOTE

The boost pumps are accessible through access plates on the rear spar. Removal and installation of the right and left boost pumps are similar procedures.

1. Defuel fuel supply system and remove the wheel well dish pan.

WARNING

Make sure battery switch is off and external electrical power is disconnected from airplane; failure to do this may result in an explosion.

2. When removing left boost pump, remove left fuel pump access door (see Installing Fuel Boost Pump).
3. When removing right boost pump, remove aileron sector guard, relieve aileron cable tension; then remove aileron bellcrank interconnecting rod and outboard bolt, nut, and washer from right and left aileron bellcrank brace. Push brace up until access door is cleared. Remove right fuel pump access door.

CAUTION

Failure to relieve aileron control cable tension before disconnecting the inboard sector support link may cause damage to the aileron cable sector support fitting.

4. Trace booster pump wiring from mounting pad to terminals on noise filter. Disconnect wires from terminals, and all necessary clamps from wire bundle. Clear wires until they can be moved freely.
5. Remove outlet fitting mounting bolt. The bolthead is accessible in the fuel outlet line elbow, aft of the booster pump mounting pad. Remove fuel filter outlet line.
6. Remove fuel drain line from 90-degree drain fitting on pump mounting pad.
7. Through access hole, disconnect fuel bleed line at check valve. Hold check valve with wrench when loosening B nut.
8. Slide fuel bleed line outboard until line is clear of check valve; then pull line up and out. Cap line. Temporarily spot-tie wires in front of access hole. Remove check valve and cap both ends of check valve and end of line.
9. Remove the five bolts and washers from booster pump mounting pad on aft wing spar. The remaining bolt should be the one most easily removed.
10. Reach through access hole and support booster pump; then remove last bolt and washers.
11. Rotate pump until top portion is in a horizontal position and comes out access holes first. Feed wire group through hole and pull out with pump.
12. Subsequent to removing the RH boost pump, lubricate the sector interconnect push rod end bearings (see Installing Fuel Boost Pump). Also lubricate the push-pull rod end that connects to the outboard sector.
13. Remove and discard O-ring from pump flange.
14. Place protective covering over cell opening. Do not use tape; removal of tape may cause a spark.

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MAINTENANCE PRACTICES

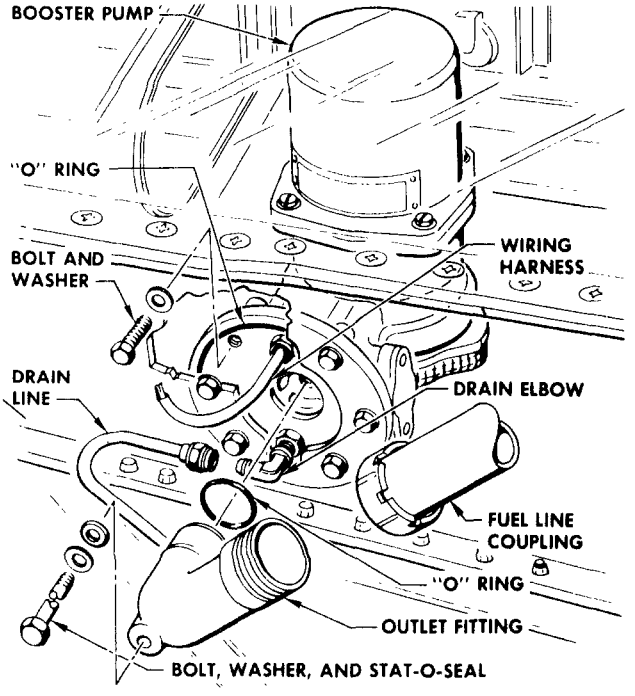
Installing Fuel Boost Pump

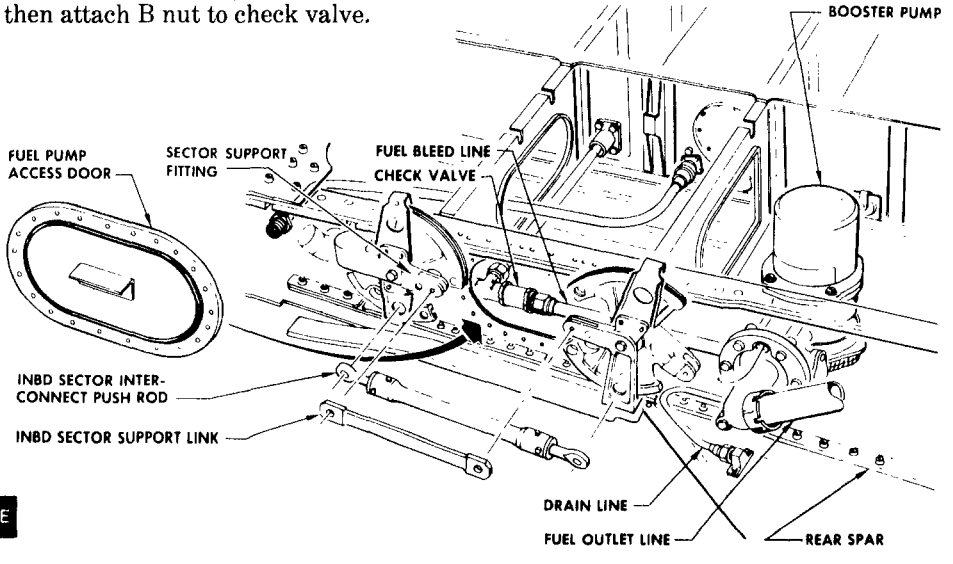
1. Observe precaution for fuel system maintenance.

WARNING

Failure to disconnect electrical power to airplane may result in explosion.

2. Make sure that battery switch is off and that external power is disconnected from airplane.
3. Remove protective coverings from booster pump and airplane access.
4. Install MS29513-237 O-ring packing in booster pump mounting flange. Hold O-ring packing in place with petrolatum.
5. Feed pump wiring harness in through wing access holes and back out through booster pump hole in rear spar; then while holding booster pump in horizontal position, with top portion aft, work pump through access holes. Position pump on mounting pad, and install six AN4-5A bolts and 2W17-416M washers. Torque bolts to 50-70 inch-pounds.
6. Place MS29513-24 O-ring packing in pump outlet. Install washer and 600-001-5/16 Stat-O-Seal washer on outlet fitting mounting bolt. Check that drain outlet elbow is loosely installed.
7. Position outlet fitting so that it aligns with fuel line, and install mounting bolt. Torque bolt to 110-130 inch-pounds.
8. Install fuel outlet line coupling. Torque coupling nut to 120-180 inch-pounds.
9. Align drain elbow with drain line, and connect drain line elbow. Torque B nut to 40-65 inch-pounds, and tighten jam nut on elbow.
10. Remove caps from check valve and from fuel bleed line which connects to it. Carefully slide line into booster pump; then attach B nut to check valve.
11. Remove foreign materials from tank, and install rear spar access door. Torque access door bolts to 25-35 inch-pounds.
12. On right-hand installation, reinstall aileron sector support link and sector interconnecting push rod.
13. On right-hand installation, tension aileron control cables and perform aileron rigging check (refer to chapter 27).
14. Install and connect booster pump wiring (refer to Wiring Manual).





MAJOR CHANGE

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MAINTENANCE PRACTICES (Continued)

Replacement of Fuel Transfer Ejectors

NOTE

The right fuel transfer ejector is in the right wing center section, immediately aft of the front wing spar. The left fuel transfer ejector is in the left wing center section, immediately aft of the front wing spar. Removal and installation of the right and left fuel transfer ejectors are similar procedures.

1. Defuel fuel supply system.

WARNING

Make sure battery switch is off and external electrical power is disconnected from airplane; failure to do this may result in an explosion.

2. Remove access door, 24 or 34, depending on which ejector is to be removed, (refer to chapter 6, Figure 6-00-3). Inside bay, remove square stress door on aft bulkhead (see Figure 28-20-401).
3. Through square stress door opening, remove access door on wing center section front spar. Save Stat-O-Seals on bolts and protect Gask-O-Seal on door.
4. Disconnect large B nut from ejector and rotate away from fitting.
5. Disconnect small B nut from ejector and pull away from fitting.
6. Remove six bolts and washers from ejector mounting plate.
7. Rotate ejector until it will slide through access hole. Remove ejector.
8. Place protective covering over cell opening. Do not use tape; removal of tape may cause a spark.

Installing Fuel Transfer Ejector

WARNING

Make sure battery switch is off and external electrical power is disconnected from airplane; failure to do this may result in an explosion.

1. Remove protective covering.
2. Place ejector in tank and position on mounting plate (see Figure 28-20-401).
3. Install mounting bolts and washers.
4. Connect small B nut to ejector fitting.

NOTE

If B nut does not align with fitting, check alignment of B nut and fitting. Use a long-handled mirror and a vaporproof flashlight. If line must be re-formed to align B nut, be careful not to kink line.

5. Connect large B nut to ejector fitting.
6. Check both lines to make sure they were not kinked during B nut connection. Use a mirror and a vaporproof flashlight.

NOTE

The Gask-O-Seal is an integral part of the access door and is not replaceable. If the Gask-O-Seal is damaged in any way, the access door must be replaced.

7. Check that area is clean; then install spar access door using serviceable Stat-O-Seals on each bolt. Torque bolts to 25-35 inch-pounds.
8. Install stress door in fuselage bay; then install access door.

Obstruction Check of Fuel Transfer Ejector Nozzles.

NOTE

The following procedure should be used whenever the ejectors are believed to be plugged. This procedure will also serve to clear the ejector nozzles if they are found to be plugged. See Figure 28-20-402 and proceed as follows:

WARNING

Make sure battery switch is off and external electrical power is disconnected from airplane; failure to do this may result in an explosion.

1. Remove right- and/or left-hand fuel pump access door from wing center section rear spar, depending on which ejector is to be checked (see Installing Fuel Boost Pump).
2. Disconnect ejector motive fuel flow line, running from boost pump compartment forward baffle at fuel bleed line T fitting inboard of access door. Slide line out of adapter on forward baffle.

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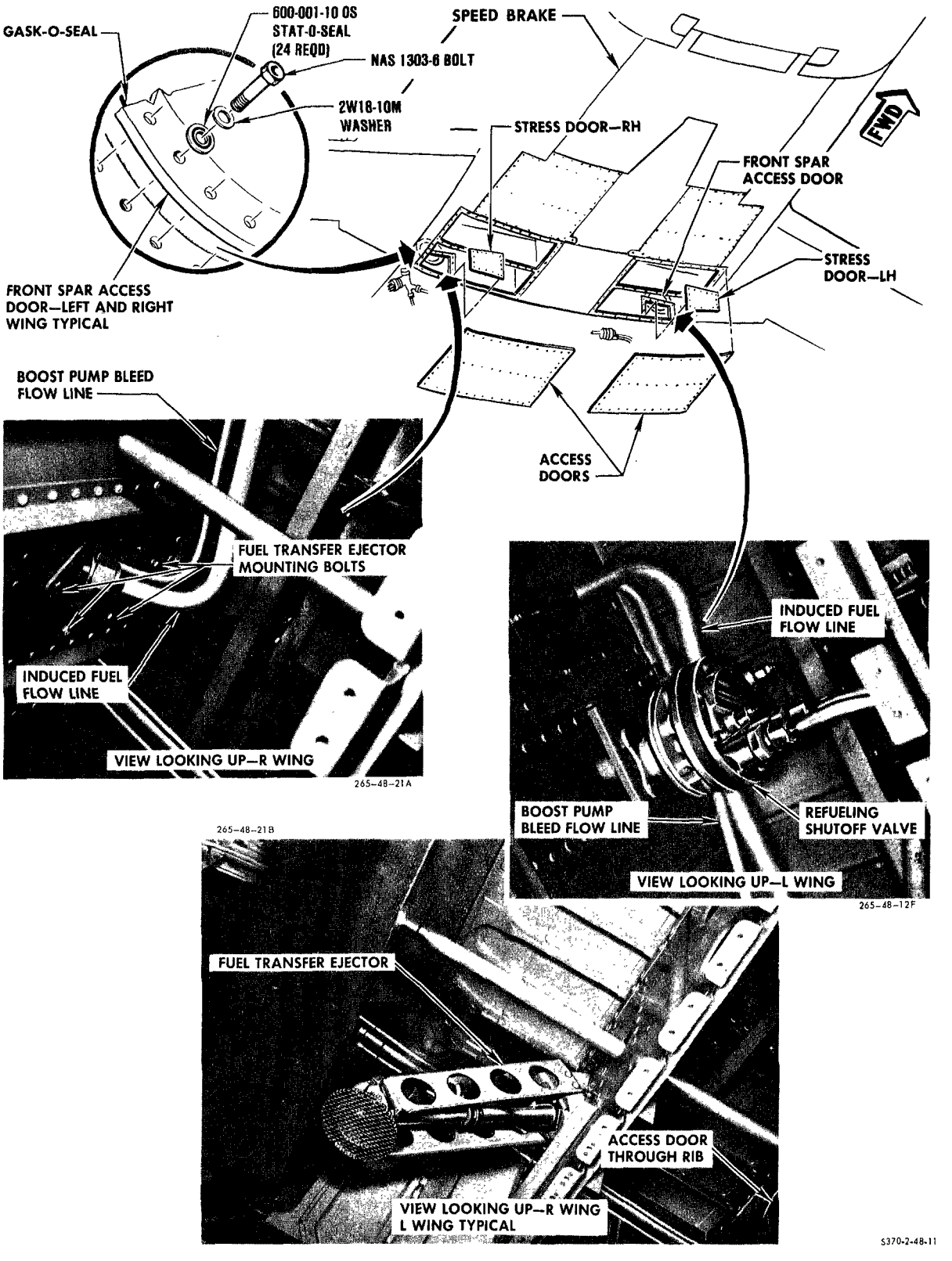

Figure 28-20-401 — Fuel Transfer Ejector Installation

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MAINTENANCE PRACTICES (Continued)

Obstruction Check of Fuel Transfer Ejector Nozzles (Cont'd)

3. Install a fabric bag over end of fuel line adapter.
4. Remove access doors 23 or 24, as shown in Figure 6-00-3, depending on which ejector is to be checked.
5. Remove square stress door on aft bulkhead (see Figure 28-20-401).
6. Working through stress door opening, remove access door on wing center section front spar. Save Stat-O-Seals on bolts and protect Gask-O-Seal on door.
7. Disconnect large induced flow line B nut from ejector and rotate away from fitting.
8. Insert end of a 4-foot-long hose, with 7/16-inch OD and 3/16-inch ID, into top of ejector, sealing against nozzle (see Figure 28-20-402). Hold hose securely against nozzle during check.
9. Apply air pressure to hose and note that airflow is detected at fuel line adapter.
10. If no airflow is detected, increase air pressure until contaminants are blown out of nozzle, and air flows freely at fuel line adapter.
11. Remove bag from fuel line adapter and note entrapped contaminants.
12. Temporarily install fabric bag over end of fuel line adapter. Hold hand over end of adapter to seal off fuel line. Fill hose with fuel or other suitable solvent and apply air pressure. Allow air pressure to build up; then remove hand from end of fuel line adapter. Repeat this procedure until all contaminants are removed.
13. Reinstall fuel motive flow line.
14. Remove hose from ejector, and install ejector return line.
15. Reinstall fuel pump access door.
16. Reinstall fuel ejector access door.
17. Perform check-out of fuel transfer ejectors. (refer to 28-20-03.)

REPLACEMENT OF FUEL SHUTOFF VALVES AND BOOST CROSS-FEED VALVE ASSEMBLY

NOTE

The fuel shutoff valves and the boost cross-feed valve are mounted as an assembly between the wing attach beam and a fuselage cross-member (directly beneath the access panel in the cabin floor) and the top of the wing center section.

1. Make sure fuselage tank is empty and wing tanks are no more than half full.

WARNING

Make sure battery switch is off and external electrical power is disconnected from airplane; failure to do this may result in an explosion.

2. Remove access panel in cabin floor directly above assembly.
3. Through access opening, manually actuate all valves to open position (see Figure 28-20-403).
4. At forward inboard corners of right and left wheel wells, place a 5-gallon container under each main fuel supply line drain cock.
5. Through cabin floor access opening, locate the 1/4-inch lines that connect fuel boost pump lines and pressure switches on forward fuselage cross-member. Loosen B nuts attached to pressure switches.
6. Open drain cocks above 5-gallon containers. Fuel should drain out. Do not shut off drain cocks until all flow stops. This drainage empties lines that must be disconnected to allow removal of this assembly.
7. After lines have been emptied, shut off drain cocks.
8. Through access hole, disconnect the three electrical connectors to the three valves and the two electrical connectors to the two pressure switches. Cap connectors and plugs.
9. Cut safeties on all Wiggins couplings to the two shutoff valves.
10. Remove the two 1/4-inch lines that connect fuel boost pump lines and pressure switches on forward fuselage cross-member. Cap fittings and plug B nuts.
11. Disconnect the four Wiggins couplings to fuel shutoff valves. Cover all openings. Do not use masking tape.
12. On right side of assembly, disconnect small tubular support member attached to bottom of bracket. Remove one bolt, nut, and washer.
13. Remove wire bundle tie-down clamps from top of bracket assembly. Remove three bolts, nuts, and washers that attach assembly bracket to aft attach point (beam).
14. Remove three bolts, nuts, and washers that attach assembly bracket to forward cross-member. Support assembly while removing last bolt.
15. Gently maneuver assembly to necessary position; then remove assembly.
16. Temporarily replace cabin floor access panel.

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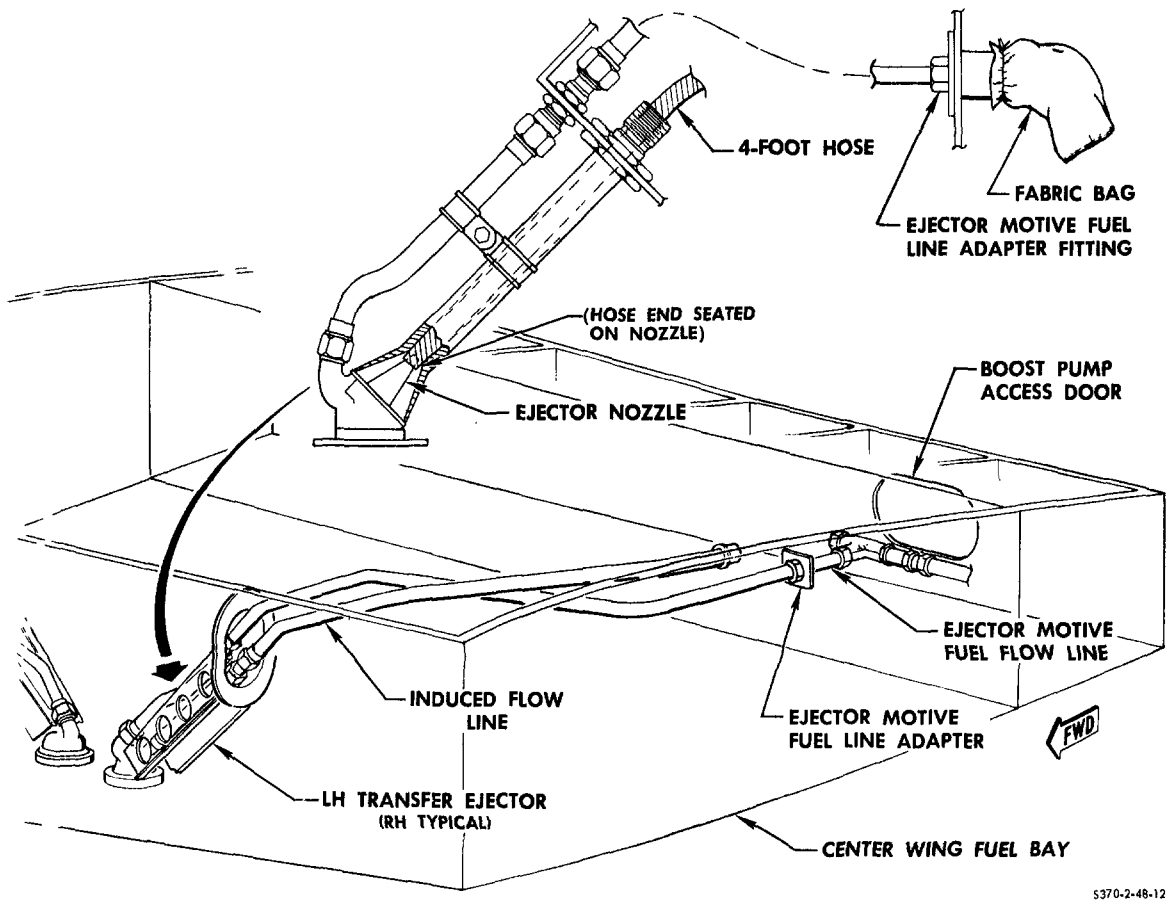

Figure 28-20-402 — Fuel Transfer Ejector Nozzle Obstruction Check

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MAINTENANCE PRACTICES (Continued)

Installing Fuel Shutoff Valves and Boost Cross-feed Valve Assembly

WARNING

Make sure battery switch is off and external electrical power is disconnected from airplane; failure to do this may result in an explosion.

1. Remove cabin floor access panel.
2. Position bracket assembly mounting bolts and washers in an accessible location for ease of installation. Make sure all valves are in a closed position (see Figure 28-20-403).
3. Through cabin floor access panel, position assembly and insert a bolt and washer in each end. Install a nut on each bolt to hold assembly in position. Make sure aft wire bundle is on top of bracket assembly.
4. Install remaining four bolts and washers. Install nuts; then torque.
5. Attach small tubular support member to bottom of right side of bracket.
6. Remove all coverings on Wiggins fittings, B nuts, pressure switch fittings, and any openings on shutoff valves.
7. Connect four Wiggins couplings to fuel shutoff valves. Torque and safety couplings.
8. Install the two 1/4-inch lines that connect fuel boost pump lines and pressure switches on forward fuselage cross-member.
9. Remove caps from valve electrical plugs and from attaching connectors. Install connectors to plugs, and safety. Install two wire bundle tie-down clamps on top of bracket assembly.
10. Temporarily install access panel.
11. Check operation of assembly.
12. Install access panel.

REPLACEMENT OF TANK CROSS-FEED VALVE

Removing Tank Cross-feed Valve

WARNING

Make sure battery switch is off and external electrical power is disconnected from airplane; failure to do this may result in an explosion.

1. Defuel fuel supply system.
2. Remove any lines necessary to provide access to booster pump access plates in right and left wheel wells. Remove plates.
3. Remove female plug from cross-feed electrical cable assembly protruding through rear spar. It is immediately to the right of the booster pump access in the left wheel well.
4. Remove jam nut and lock washer from receptacle vacated in step 3; then slide electrical fitting through mounting hole into booster sump area.
5. Remove four bolts, washers, and nuts from cross-feed valve.
6. Remove cross-feed valve with electrical cable, adapter fittings, and tubes. Check Figure 28-20-404 to ensure removal of all parts from booster sump area.
7. Place protective covering over cell opening. Do not use tape; removal of tape may cause a spark.

Installing Tank Cross-feed Valve

WARNING

Make sure battery switch is off and external electrical power is disconnected from airplane; failure to do this may result in an explosion.

1. Remove protective covering from cell openings and electrical connector.
2. See Figure 28-20-404. With O-rings installed, insert cross-feed valve with electrical cable, adapter, and tube into left tank. Insert tube (red band forward) into flapper valve. Align adapter plate, with tube installed, and cross-feed valve with holes in rib. Install bolts.
3. Insert tube with adapter plates into right tank. With gasket in place, insert tube (red band forward) into flapper valve, align adapter plate on bolts, and install washers and nuts.
4. Place electrical connection through its mounting hole to the right of left wheel well access hole. Install lock washer and jam nut.
5. Connect electrical plug to cross-feed valve connector.
6. Install boost pump access doors in both wheel wells.

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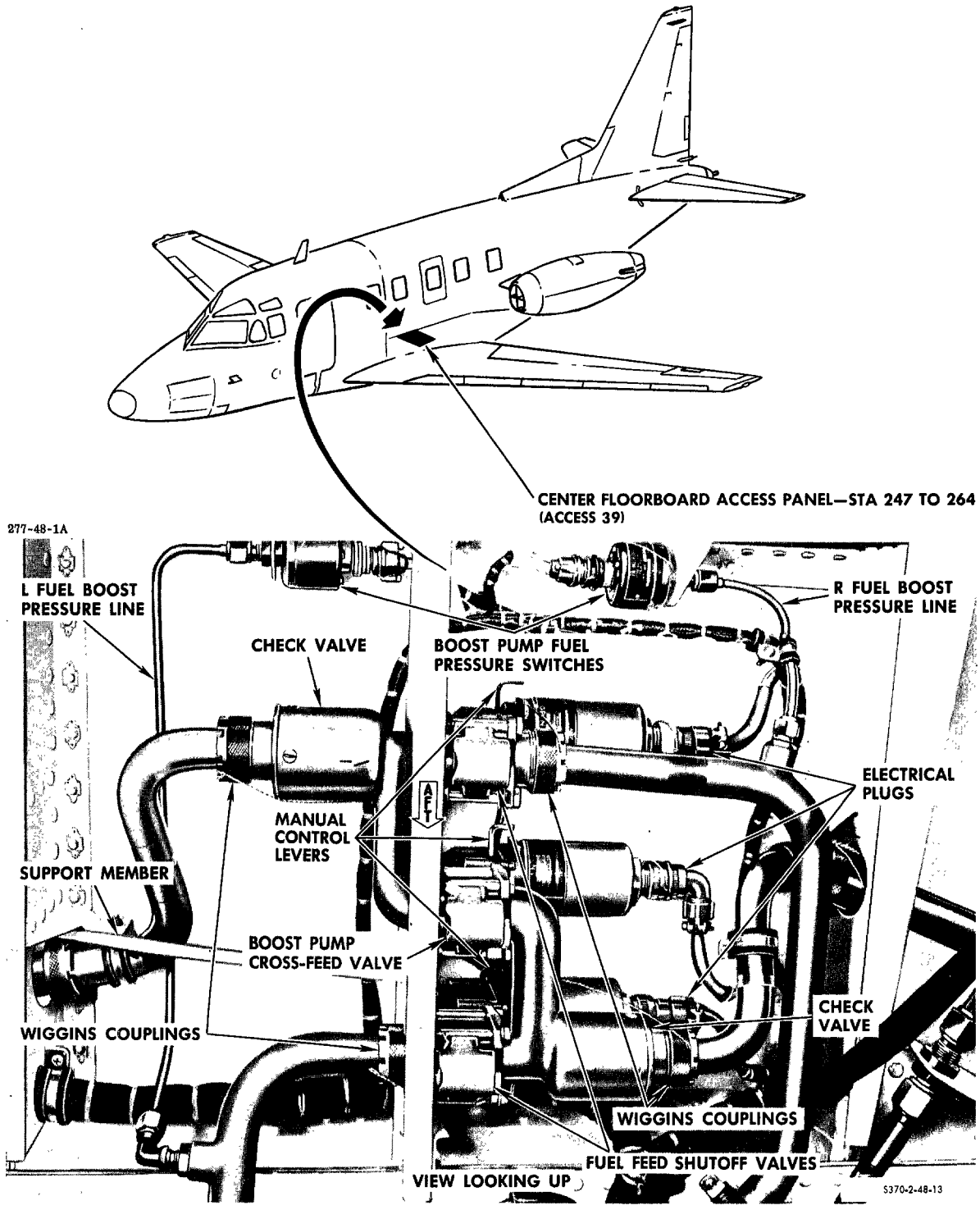

Figure 28-20-403 — Installing Fuel Shutoff Valves and Boost Cross-feed Valve Assembly

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MAINTENANCE PRACTICES (Continued)

REMOVAL OF FUEL HEATER — S/N 370-1, -7, 380-1 AND SUBS

1. Open engine access doors.
2. Disconnect and cap fuel inlet and outlet lines at fuel heater.
3. Disconnect engine bleed-air line at fuel heater.
4. Remove the four bolts securing fuel heater support brackets to inbd side of fuel heater recess.
5. Rotate fuel heater outbd from recess and remove.

INSTALLING FUEL HEATER — S/N 370-1, -7, 380-1 AND SUBS

1. Rotate fuel heater inbd and position against inbd side of fuel heater recess.
2. Install and tighten four bolts through fuel heater support brackets and inbd side of recess.
3. Connect engine bleed-air line to fuel heater.
4. Connect fuel inlet and outlet lines to fuel heater. Torque to 40 in.-lbs.
5. Close engine acess doors.

DRAINING FUEL HEATERS — S/N 370-1, -7, 380-1 AND SUBS

1. Open access doors in pylon directly below fuel heaters.
2. Remove water drain plugs (fwd plugs) from heater. (See Figure 28-20-406.)
3. Pressurize fuel system for 15 minutes.
4. Check drain ports for evidence of fuel. If evidence of fuel appears at ports, replace fuel heater(s).

NOTE

The airplane should be positioned so that bottom of fuel heater slopes toward the drain plugs to allow gravity flow. The ambient temperature of the fuel heater should be 40°F. (±4°F) or higher when performing this check.

5. Remove electrical power from airplane and reinstall water drain plugs using new gaskets. Torque plugs to 120 in.-lbs and safetywire.
6. Remove fuel drain plug from center of heater. Allow fuel to drain; then, replace plug using new O-ring. Torque plug to 120 in.-lbs and safetywire.
7. Close access doors.

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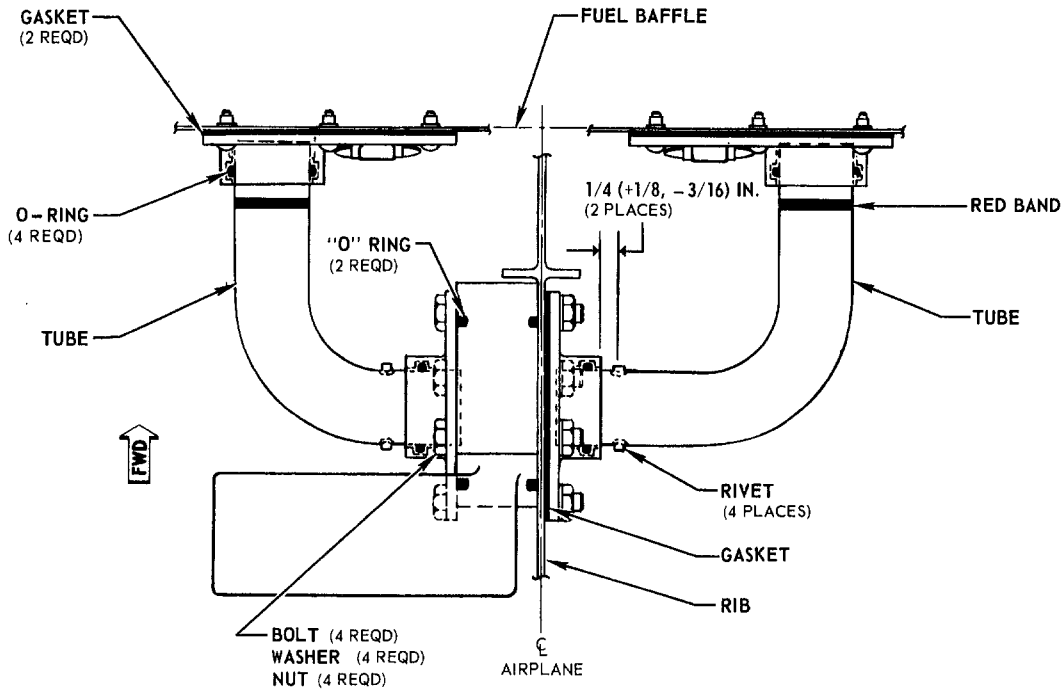

Figure 28-20-404 — Installing Tank Cross-feed Valve Assembly

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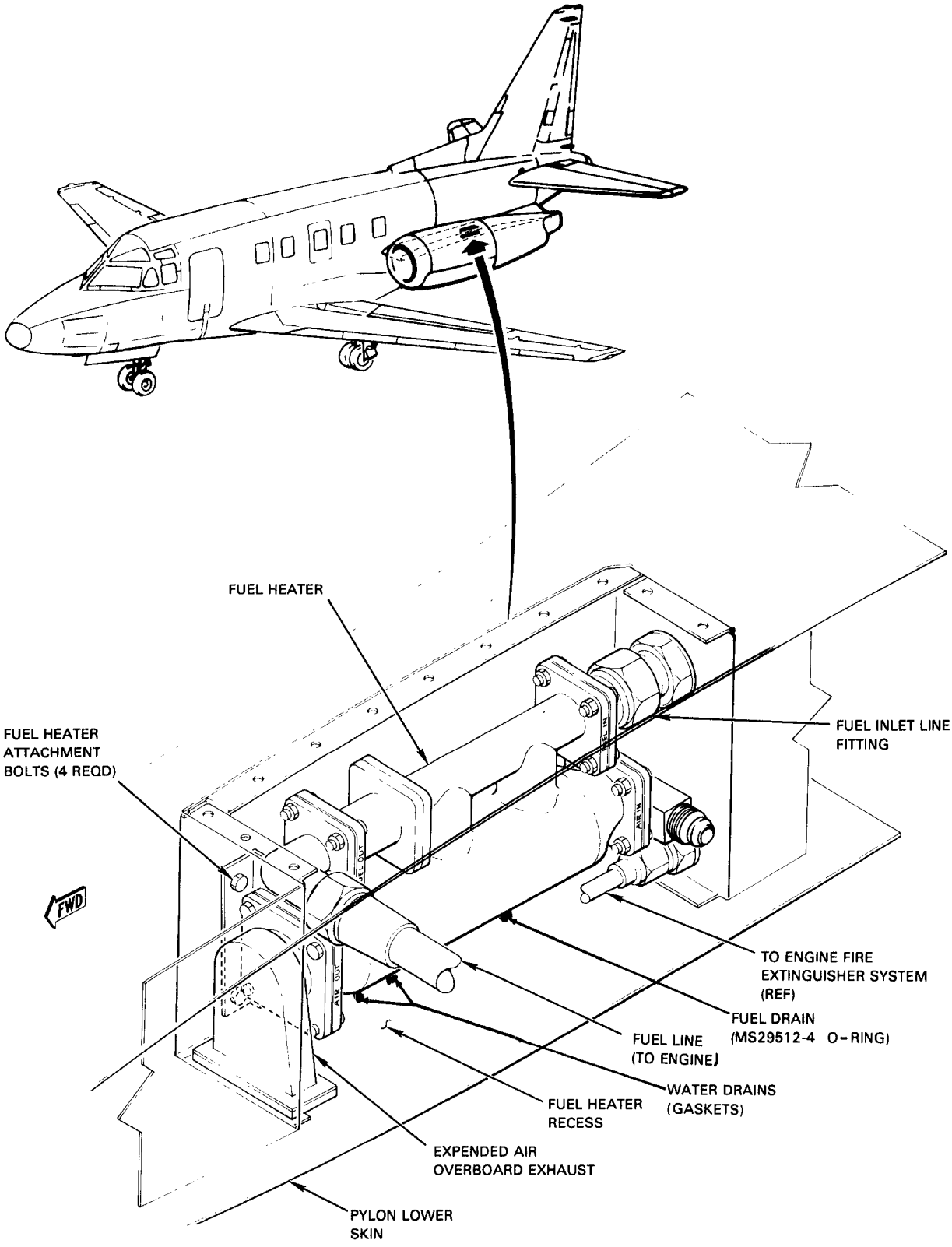

Figure 28-20-405 — Installing Fuel Heater — S/N 370-1, -7, S/N 380-1 and Subs

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1. Fuel Boost Pump Check-out Procedures

A. Fuel Boost Pump Electrical Check

The fuel boost pump operational checking procedure consists of applying a source of 28-volt dc electrical power to the electrical system, engaging the necessary circuit breaker, and then listening at each pump. If the pump can be heard, it is working. If it cannot be heard, the electrical system is at fault, or the pump is faulty. (See figure 28-00-2.)

(1) Make sure "LH ENGINE - ENGINE MASTER" switch on left engine switch panel and "RH ENGINE - ENGINE MASTER" switch on right engine switch panel are at OFF and "LH ENGINE - IGN & BOOST" and "RH ENGINE - IGN & BOOST" circuit breakers on left overhead forward dc essential circuit-breaker panel are disengaged. Disengage LH and RH "BOOST PUMP" circuit breakers.

(1A) Move power lever forward and position ENGINE MASTER switches to ON.

(2) Connect 28-volt dc external electrical power source to airplane. On left overhead forward dc essential circuit-breaker panel, engage "BOOST PUMP - LH FUEL" circuit breaker.

(3) Have helper in left wheel well to listen at forward bulkhead (wing rear spar). If the left boost pump can be heard when it starts and stops as the man in the cockpit engages and disengages the circuit breaker, the pump is working.

(4) Disengage "BOOST PUMP - LH FUEL" circuit breaker and engage "BOOST PUMP - RH FUEL" circuit breaker.

(5) Have helper in right wheel well to listen at forward bulkhead (wing rear spar). If the right boost pump can be heard when it starts and stops as the man in the cockpit engages and disengages the circuit breaker, the pump is working.

(6) Remove external power source.

(7) Restore circuit breakers and switches to normal positions.

B. Fuel Boost Pump Leakage Check

The seal drain on the left boost pump is aft of the rear spar of the wing center section in the left wheel well. The seal drain on the right boost pump is aft of the rear spar of the wing center section in the right wheel well. These drains must be visually inspected for leakage. The pump should not leak more than two drops in 3 minutes. If leakage is greater, the cause may be foreign particles under the seal, an extra-dry seal, or a faulty seal.

Foreign matter can become lodged between the surfaces of the moving seal, causing separation great enough to allow leakage. A pump seal may become extra dry when the airplane has been idle for a long time. These conditions can be eliminated by turning the pump on for a short period. If running the pump does not stop leaking, the seal is faulty and the pump must be replaced.

C. Fuel Boost Pump Pressure Check

(1) Disconnect external power from airplane.

(2) In right wheel well, remove drain cock from main fuel line at about fuselage station 268.

(3) Install 0 to 50 psi gage.

(4) On left overhead forward dc essential circuit-breaker panel, disengage "LH ENGINE - IGN & BOOST" and "RH ENGINE - IGN & BOOST" circuit breakers. On left and right engine switch panels, make sure "LH ENGINE - ENGINE MASTER" and "RH ENGINE - ENGINE MASTER" switches are at ON.

(5) Connect 28-volt dc external power source to airplane.

(6) On left overhead forward dc essential circuit-breaker panel, engage "BOOST PUMP - LH FUEL" circuit breaker.

(7) The left boost pump should run and the static pressure on gage should read between 17 and 30 psi.

(8) Disengage "BOOST PUMP - LH FUEL" circuit breaker, remove test gage, and reinstall drain cock.

(9) In left wheel well, remove drain cock from main fuel line at about fuselage station 268.

(10) Install 0 to 50 psi gage.

(11) On left overhead forward dc essential circuit-breaker panel, engage "BOOST PUMP - RH FUEL" circuit breaker.

(12) Right boost pump should run and static pressure on gage should read between 17 and 30 psi.

(13) Disengage "BOOST PUMP - RH FUEL" circuit breaker. Remove external power.

(14) Remove test gage and install drain cock.

(15) Restore circuit breakers and switches to normal positions.

D. Checking Fuel Transfer Ejectors

NOTE: At the conclusion of the first 20 hours of airplane operation, perform an operational check of the ejector pumps. If pumps are found to be operating properly and are free of contaminants, another inspection must be made at 100 hours of airplane operation. If pumps pass this second inspection, another inspection must be made at the conclusion of the next 200 hours of airplane operation. If the pumps are still operating properly at this

time, the inspection may be performed every 600 hours thereafter.

- Any time an ejector pump is found inoperative because of contaminants in the nozzle, ejector operation should be checked after every 10 hours of airplane operation, until the ejectors are found free of contaminants. When the ejectors are found free of contamination, the inspection interval should revert back to 100 hours then 200 hours, and then 600 hours as previously metioned.

(1) Check that fuel boost pump pressure is 17 to 30 psi before starting ejector pump check. (Refer to paragraph 1.C. for fuel boost pump pressure check.)

(2) With airplane at normal ground attitude (0 degree 46 minutes nose-up and wings level laterally), make sure there is about 335 pounds (100 gallons) of fuel in each wing tank.

(3) Disconnect left fuel feed line at left engine (or fuel feed line in aft fuselage - airplanes NA380-38 and later and airplanes changed by SB74-23). Connect a restrictor made of 3/8 inch OD tubing with 0.035-inch wall thickness, 8 feet long, to fuel feed line. Extend restrictor and insert open end of restrictor into empty fuel container capable of holding all wing fuel. Restrictor tubing should not be obstructed in any way (dents, crimps, bends, etc). The restrictor is the rate-of-flow measuring device that measures the fuel flow at maximum engine rpm at the most critical altitude. The restrictor should be positioned as near level with the boost pump as possible (for proper head pressure).

(4) Connect 28-volt dc external power source to airplane.

(5) On left overhead forward dc essential circuit-breaker panel, make sure the following circuit breakers are engaged.

(a) "LH ENGINE - IGN & BOOST"

(b) "RH ENGINE - IGN & BOOST"

(c) "LH ENGINE - MASTER"

(d) "RH ENGINE - MASTER"

(e) "BOOST PUMP - LH FUEL"

(f) "BOOST PUMP - RH FUEL"

(6) On electrical master and emergency overhead switch panel, on NA-370-2 through -6, -8, and -9 airplanes, make sure "EMERGENCY - FUEL JETTISON" switch is at OFF. On right engine switch panel, make sure "FUEL" switch is at NORMAL.

NOTE: When power is on the 28-volt dc essential bus, the fuel cross-feed valve indicator light and, on NA-370-2 through -6, -8, and -9 airplanes, the jettison control valve indicator light must be extinguished.

(7) With ignition selector switch at normal, place "LH ENGINE - ENGINE MASTER" switch at ON and advance left throttle to IDLE. Continue operation of left boost pump until fuel flow ceases from left fuel feed line.

(8) Return left throttle to CLOSE position and place "LH ENGINE - ENGINE MASTER" switch at OFF.

CAUTION: DO NOT PERMIT FUEL BOOST PUMPS TO OPERATE MORE THAN 1/2 MINUTE WHEN TANKS ARE DRY. HEAT GENERATED BY THE UN-COOLED MOTOR MAY CAUSE PUMP DAMAGE.

(9) Drain fuel remaining in left wing center section through tank sump drain to determine quantity. If the amount drained is more than 4 gallons, the ejector pump is not functioning properly. If too much fuel remains, make an obstruction check of fuel transfer ejector nozzles. (Refer to paragraph 2.C. in 28-20-2.)

(10) Reconnect left fuel feed line to left engine.

(11) Repeat steps (3) through (10) for check of right wing ejector pump.

(12) Remove external power and position switches at normal positions.

2. Fuel Cross-Feed Valve Check-out Procedure

A. Checking Right Fuel System, Boost Cross-feed Valve, and Wing Fuel Leveling

(1) On left overhead forward dc essential circuit-breaker panel, engage the following circuit breakers:

(a) "LH ENGINE - IGN & BOOST"

(b) "RH ENGINE - IGN & BOOST"

(c) "LH ENGINE - MASTER"

(d) "RH ENGINE - MASTER"

(e) "BOOST PUMP - LH FUEL"

(f) "BOOST PUMP - RH FUEL"

(g) "FUEL-X-FEED - TANK"

(h) "FUEL-X-FEED - BOOST"

(2) On NA-370-2 through -6, -8, and -9 airplanes, on electrical master and emergency overhead switch panel, make sure "EMERGENCY - FUEL JETTISON" switch is at OFF. On right engine switch panel, make sure "FUEL" switch is at NORMAL. (See figure 28-20-601.)

NOTE: When power is on the 28-volt dc essential bus, the fuel cross-feed valve indicator light and the jettison control valve indicator light* must be extinguished.

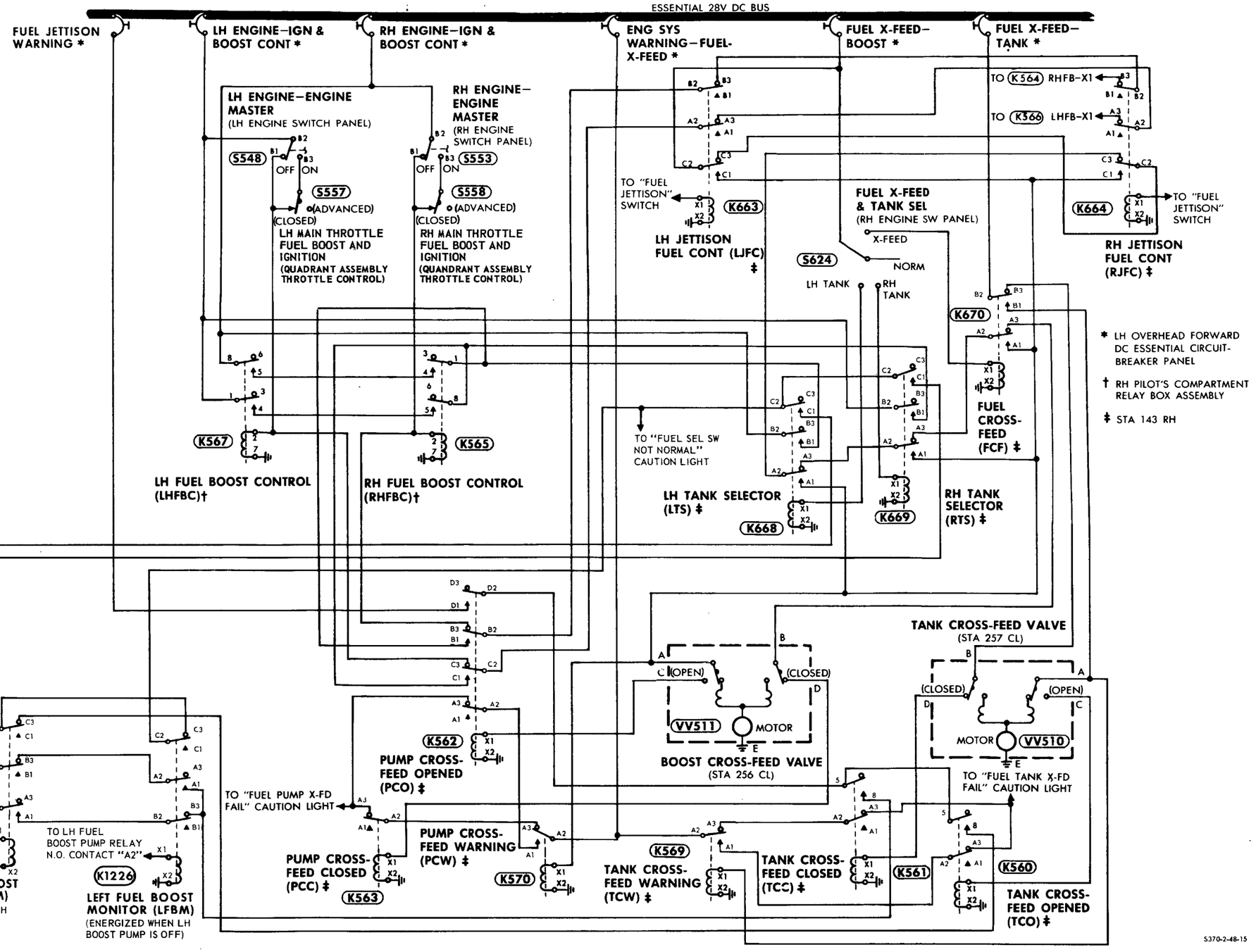
*The fuselage tank must be full to provide the necessary head of fuel to conduct the following gravity-flow tests.

(3) Connect 28-volt dc external power source to airplane.

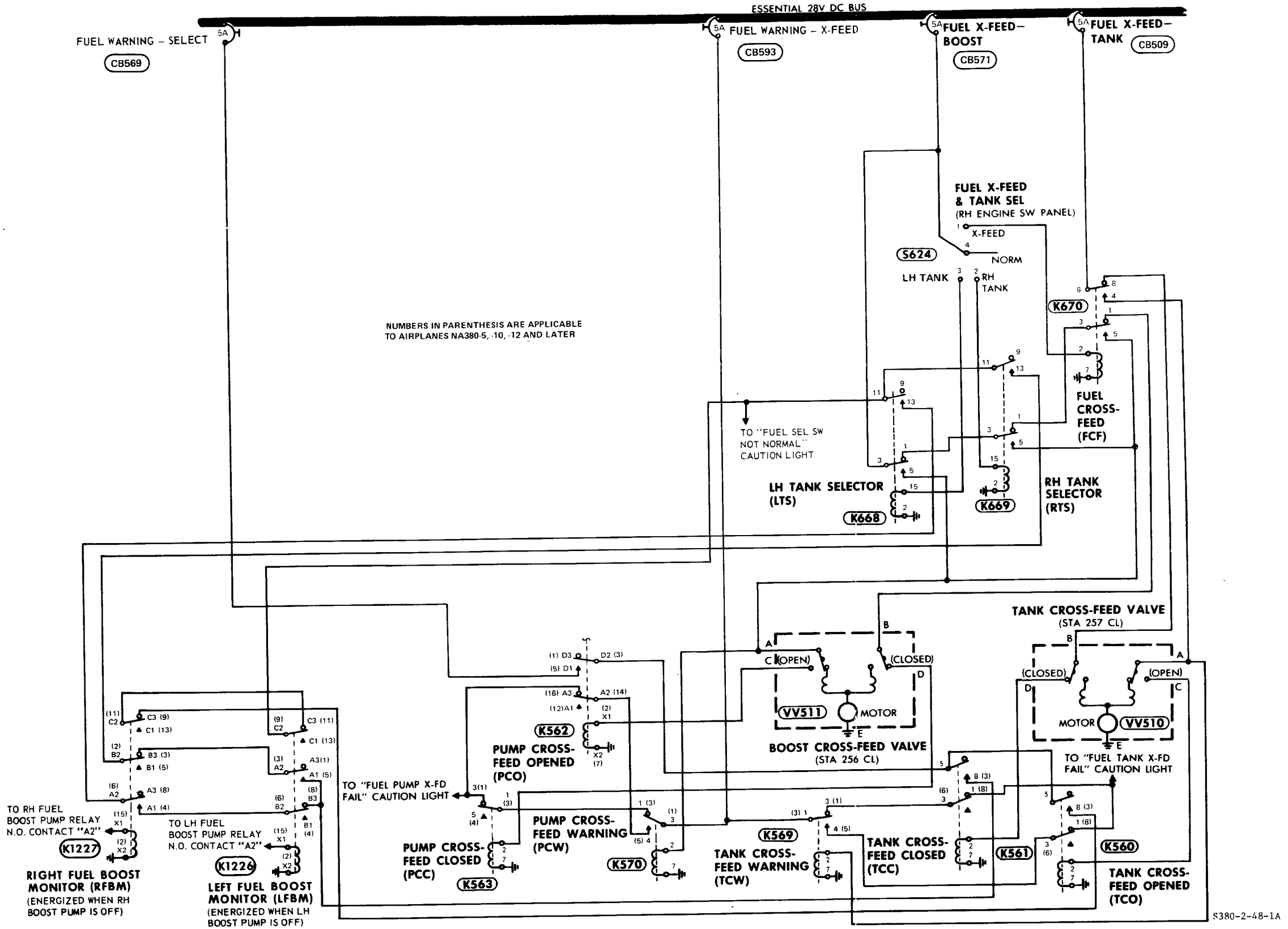
(4) Disconnect right fuel feed line at right engine. Place a 5-gallon container in a position to receive fuel from this open line.

(5) Actuate "RH ENGINE - ENGINE MASTER" switch to ON. Check that fuel flows from open feed line, indicating that right fuel shutoff valve has been energized to open position.

*NA-370-2 through -6, -8 and -9



Fuel Cross-feed System Electrical Schematic -
NA-370-2 Through -6, -8 and -9
Figure 28-20-601



Fuel Cross-feed System Electrical Schematic -
NA-370-1 and -7, NA-380-1 and Later Airplanes
Figure 28-20-602

(6) Advance right engine throttle to IDLE. Check that fuel flow increases, indicating that right boost pump has been energized.

(7) Return right engine throttle to CLOSE and check that fuel flow decreases, indicating that right boost pump was de-energized.

(8) Return "RH ENGINE - ENGINE MASTER" switch to OFF. Check that fuel flow stops, indicating that right fuel shutoff valve was closed.

(9) On left overhead forward dc essential circuit-breaker panel, disengage "FUEL-X-FEED - TANK" and "BOOST PUMP - RH FUEL" circuit breakers.

(10) Actuate "RH ENGINE - ENGINE MASTER" switch to ON. Check that fuel flows from the open feed line, indicating that right fuel shutoff valve was open.

(11) Actuate "FUEL" switch to -X-FEED. Advance left engine throttle to IDLE. Check that fuel flow from open right feed line increases, indicating that boost cross-feed valve is open and left boost pump is energized and is furnishing fuel to right engine.

(12) Return left engine throttle to CLOSE and "FUEL" switch to NORMAL. Check that fuel flow decreases and stops, indicating that left boost pump was de-energized and that boost cross-feed valve was closed. Return "RH ENGINE - ENGINE MASTER" switch to OFF.

(13) Engage "FUEL-X-FEED - TANK" circuit breaker and disengage "BOOST PUMP - LH FUEL" circuit breaker. Actuate "RH ENGINE - ENGINE MASTER" switch to ON. Check that fuel flows from open feed line, indicating that right fuel shutoff valve was opened.

(14) Actuate "FUEL" switch to LH TANK. Advance right and left engine throttles to IDLE. Engage "BOOST PUMP - LH FUEL" circuit breaker. Check that there is a fuel flow increase from open right feed line. An increase in fuel flow from the open right feed line indicates that the left booster pump was energized.

(15) Disengage "BOOST PUMP - LH FUEL" circuit breaker and engage "BOOST PUMP - RH FUEL" cicuit breaker. Check whether there is a fuel flow decrease. A decrease in fuel flow indicates that the right booster pump was not energized.

(16) Actuate "FUEL" switch to RH TANK. Check whether there is a fuel flow increase from open right feed line. An increase in fuel flow indicates right booster pump was energized.

(17) Disengage "BOOST PUMP - RH FUEL" circuit breaker and engage "BOOST PUMP - LH FUEL" circuit breaker. Check whether there is a fuel flow decrease. A fuel flow decrease indicates that the left booster pump was not energized.

(18) Return right and left throttles to CLOSE and "RH ENGINE - ENGINE MASTER" switch to OFF. Place switches at normal positions and

disengage circuit breakers. Remove external power; then connect right fuel feed line to right engine.

B. Checking Left Fuel System and Boost Cross-feed Valve

(1) On left overhead forward dc essential circuit-breaker panel, engage the following circuit breakers:

(a) "LH ENGINE - IGN & BOOST"

(b) "RH ENGINE - IGN & BOOST"

(c) "LH ENGINE - MASTER"

(d) "RH ENGINE - MASTER"

(e) "BOOST PUMP - LH FUEL"

(f) "BOOST PUMP - RH FUEL"

(g) "FUEL-X-FEED - TANK"

(h) "FUEL-X-FEED - BOOST"

(2) On NA-370-2 through -6, -8, and -9 airplanes, on electrical master and emergency overhead switch panel, make sure "EMERGENCY - FUEL JETTISON" switch is at OFF. On right engine switch panel, make sure "FUEL" switch is at NORMAL.

NOTE: When power is on the 28-volt dc essential bus, the fuel cross-feed valve indicator light and the jettison control valve indicator light* must be extinguished.

- The fuselage tank must be full to provide the necessary head of fuel to conduct the following gravity-flow tests.

(3) Connect 28-volt dc external power source to airplane.

(4) Disconnect left fuel feed line at left engine. Place a 5-gallon container in a position to receive fuel from this open line.

(5) Actuate "LH ENGINE - ENGINE MASTER" switch to ON. Check that fuel flows from open feed line, indicating that left fuel shutoff valve has been energized to open position.

(6) Advance left engine throttle to IDLE. Check that fuel flow increases, indicating that left boost pump has been energized.

(7) Return left engine throttle to CLOSE and check that fuel flow decreases, indicating that left boost pump was de-energized.

(8) Return "LH ENGINE - ENGINE MASTER" switch to OFF. Check that fuel flow stops, indicating that left fuel shutoff valve was closed.

*NA-370-2 through -6, -8 and -9

(9) On the left overhead forward dc essential circuit breaker panel, disengage "FUEL-X-FEED - TANK" and "BOOST PUMP - LH FUEL" circuit breakers.

(10) Actuate "LH ENGINE - ENGINE MASTER" switch at ON. Check that fuel flows from the open feed line, indicating that left fuel shutoff valve was open.

(11) Actuate "FUEL" switch to -X-FEED. Advance right engine throttle to IDLE. Check that fuel flow from open left feed line increases, indicating that boost cross-feed valve is open and right boost pump is energized and is furnishing fuel to left engine.

(12) Return right engine throttle to CLOSE and "FUEL" switch to NORMAL. Check that fuel flow decreases and stops, indicating that right boost pump was de-energized and that boost cross-feed valve was closed. Return "LH ENGINE - ENGINE MASTER" switch to OFF.

(13) Connect left fuel feed line to left engine, remove external power, and place circuit breakers and switches at normal positions.

C. Checking Tank Cross-feed valve

(1) Defuel airplane.

(2) Level airplane by using jacks.

(3) Fill right wing with about 100 gallons of fuel. Use emergency (gravity) refueling method.

(4) On left overhead forward dc essential circuit-breaker panel, engage the following circuit breakers:

(a) "LH ENGINE - IGN & BOOST"

(b) "RH ENGINE - IGN & BOOST"

(c) "LH ENGINE - MASTER"

(d) "RH ENGINE - MASTER"

(e) "BOOST PUMP - LH FUEL"

(f) "BOOST PUMP - RH FUEL"

(g) "FUEL-X-FEED - TANK"

(h) "FUEL-X-FEED - BOOST"

(5) On NA-370-2 through -6, -8, and -9 airplanes, on electrical master and emergency overhead switch panel, make sure "EMERGENCY - FUEL JETTISON" switch is at OFF. On right engine switch panel, make sure "FUEL" switch is at NORMAL.

NOTE: When power is on the 28-volt DC essential bus, the fuel cross-feed valve indicator light and, on NA-270-2 through -6, -8 and -9 airplanes, the jettison control valve indicator light must be extinguished.

(6) Connect 28-volt DC external power source to airplane.

(7) Disconnect right fuel feed line at right engine. Place a 5-gallon container in a position to receive fuel from this open line.

(8) Disengage "BOOST PUMP - RH FUEL" circuit breaker.

(9) Actuate "RH ENGINE - ENGINE MASTER" switch to "ON". Actuate "FUEL" switch to "-X-FEED". Wait about one minute.

(10) Advance left engine throttle to IDLE. Check that fuel flows from open right feed line to right engine, indicating that left boost pump is energized, tank cross-feed valve is open, and boost cross-feed valve is open.

(11) Actuate "FUEL" switch to LH TANK. Wait until fuel flow decreases. This will indicate that tank cross-feed valve was closed.

(12) Return left engine throttle to CLOSE, "FUEL" switch to NORMAL, and "RH ENGINE - ENGINE MASTER" switch to "OFF". Fuel flow should decrease and stop.

(13) Place switches and circuit breakers at normal positions. Connect right fuel feed line to right engine and remove external power.

(14) Lower and remove jacks.

3. Operational Check of Fuel Heater System - Aircraft NA-370-1, -7, NA-380-1 and Later

NOTE: An operational check of the fuel heater system in the open or modulation cycle can be made only if the fuel temperature is at or below 35 degrees Fahrenheit.

A. To perform operational check of the fuel heater system when the fuel temperatures are below 35 degrees Fahrenheit, proceed as follows:

(1) Start engines (refer to chapter 71).

(2) Advance throttle to operate engine at approximately 85 percent.

(3) Check the presence of airflow at fuel heater overboard exhaust port in the engine pylon lower skin.

(4) If fuel temperatures are above 35 degrees Fahrenheit, there will be no air from the fuel heater exhaust port.

(5) Repeat operation check for opposite engine.

(6) Shut down engines.

PRESSURE REFUELING SYSTEM - DESCRIPTION AND OPERATION

1. General

The pressure refueling system single-point receptacle, in the right wing inboard leading edge, provides a single-point method for filling the integral wing and fuselage (auxiliary) fuel tanks. A test and control valve near the receptacle has primary and secondary valve test and fuselage tank fill control buttons. Wing refueling shutoff valves in the wing tank center sections shut off refueling to the wings and are actuated through sensing lines extending to the fuel vent and level control pilot valves in the wing tip ribs. Fuel to the fuselage tank is shut off by a one-piece fuel shutoff and dual level control valve. Primary and secondary valve test lines connect the refueling test control valve to the wing tip and fuselage tank level control valves, to check the primary and secondary diaphragms in the fuel shutoff valves. Refueling vent valve pressure lines route refueling pressure from the single-point receptacle to the refueling vent valves in each wing tip valve, thus permitting wing tank venting during refueling.

If the wing refueling vent valves should fail to open, the spring-loaded vent valves in the fuselage tank fitting to the vent and transfer lines (between the wing center sections and the auxiliary tank) will open after approximately 1-1/4 psi pressure build-up in the wing. This allows wing refueling venting through the auxiliary fuel tank. However, if the vent valves at the wing tips fail to open, air trapped in the wing would prevent fuel from reaching the floats in the wing refueling pilot valve. If this should happen, fuel would overflow from the vent at the tail cone and the wing affected would not be filled.

A line from the top of the wing center section to the top of the auxiliary tank aids fuel transfer from the auxiliary tank during climb.

When fuel pressure is applied to the single-point pressure refueling receptacle, the fuel is directed to the right and left wing center section tank compartments. Fuel fills the center wing compartments, flows through the interconnect tubes, and spills over through the vent passages at the top of the wing ribs into the inner compartments of the outer wing panels. From the inner compartments, the fuel flows into the intermediate compartment and into the leading edge section through the front spar. From the intermediate compartment, fuel flows into the outer compartment. Fuel is also flowing to the refueling valve test and fuselage tank refueling control valve, and to the wing tip refueling vent valves. Right after refueling flow is turned on from the refueling source, the primary and secondary refueling valve test buttons should be pressed separately. Refueling flow should stop after each button is pressed. As soon as shutoff is satisfactory, refueling should be continued.

Fuel lifting the floats in the refueling level control pilot and vent valve at each wing tip causes the remotely controlled shutoff valves in each center section compartment to shut off the fuel. (See figures 28-21-1 and 28-21-2.)

The vent portion of the level control and pilot valve is pressurized open by refueling pressure during the refueling operation, and is closed by fuel

pressure equalization when the wing fuel tank system is filled. This equalization pressure is received from the shutoff valve sensing lines when the wing refueling level control valve floats seat the needles. Flow is then directed past a shuttle valve to the opposite side of the refueling vent valve diaphragm, equalizing the refueling pressure which opened the valve and allowing the spring to return the valve to the closed position.

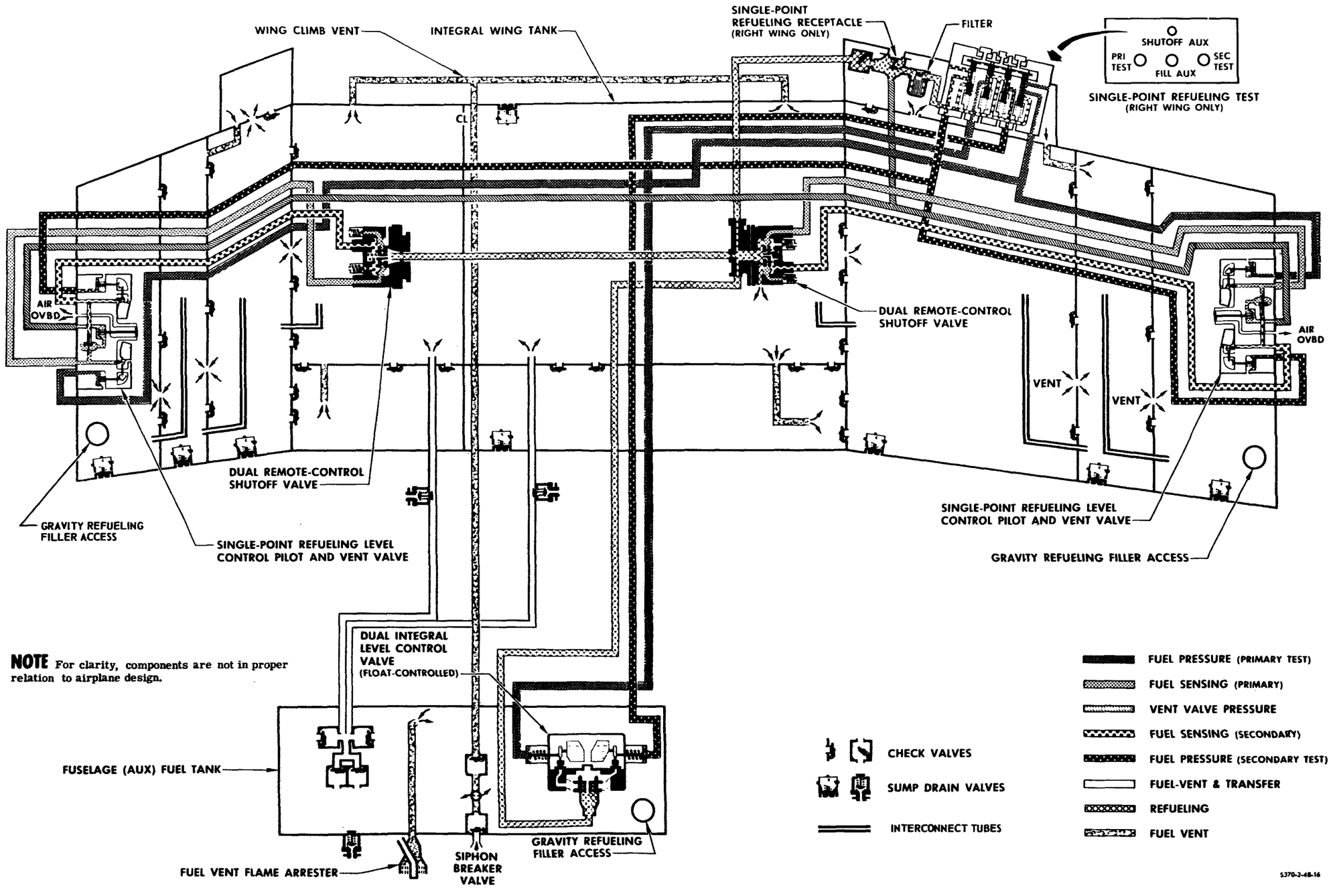
When refueling pressure is applied to the single-point pressure refueling receptacle, pressure flows to the refueling valve test and fuselage tank refueling control valve; this valve normally directs pressure to the fuselage integral dual level control valve test diaphragms. This actuates the valve closed and prevents inadvertent filling of the fuselage tank. The tank is filled by first depressing the "SHUTOFF AUX" button and then the "FILL AUX" button of the refueling valve test and fuselage tank refueling control valve, and then releasing the "SHUTOFF AUX" button first. This depressurizes the test diaphragms and allows the level valve floats to function normally, shutting off the fuel at the 163-gallon level.

2. Fuselage Tank Refueling Level Control Valve

The fuselage tank refueling level control valve is installed on the aft wall of the fuselage tank on the left side and is supported to the aft tank compartment bulkhead. A dual-float, dual-diaphragm valve, with primary and secondary test provisions, it is used to control the fuel level of the fuselage tank.

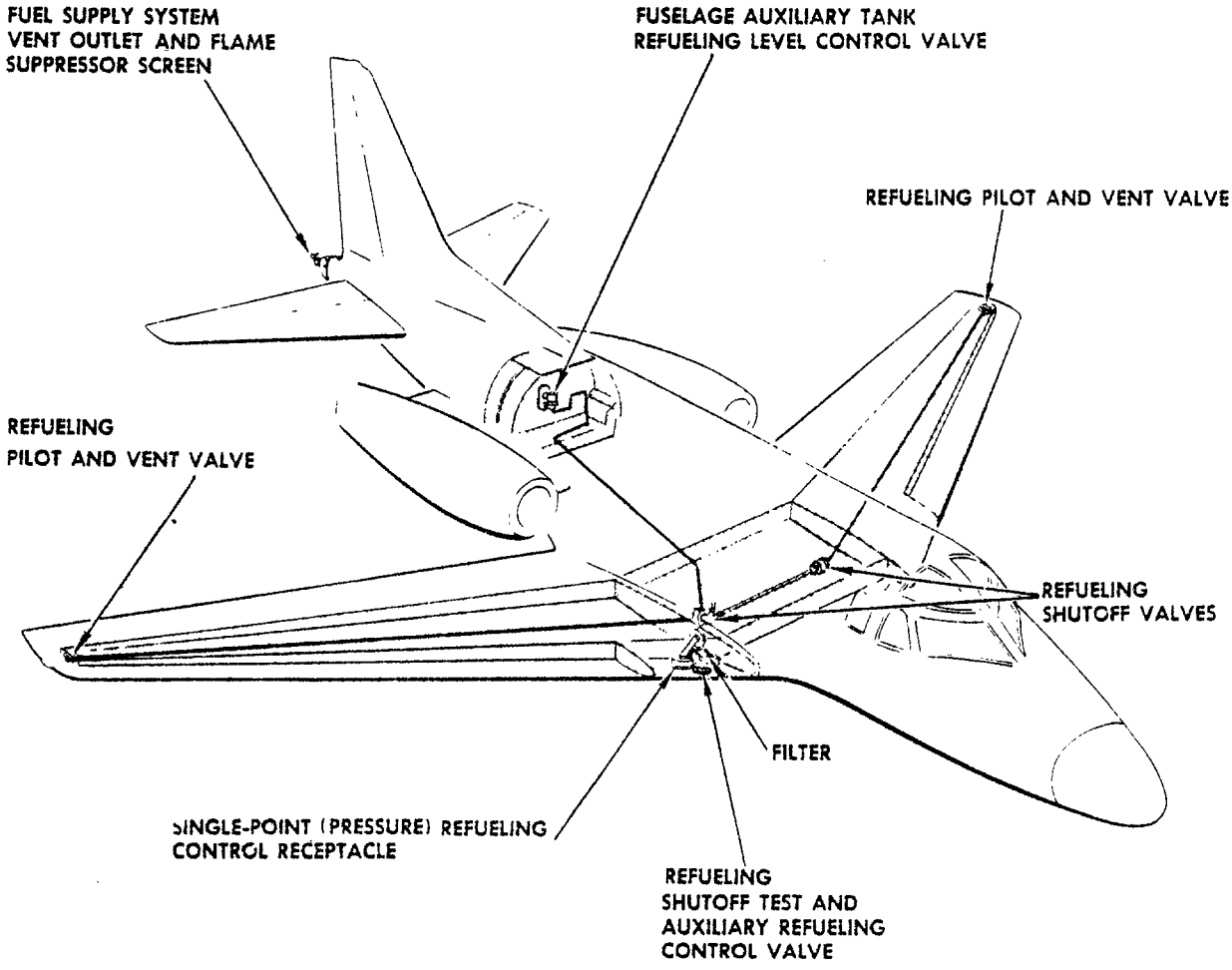
Dual-float level control valves have two floats mounted side by side. Each actuates a needle valve outside the float chamber, pressurizing dual diaphragms in the pressure chamber. The diaphragms are stacked one on the other and operate in series so that if one diaphragm fails, the other will close the valve. The pressure chamber is bounded on one side by a casting, and on the inlet side by a flexible rubberized diaphragm. The diaphragms are sandwiched between the casting, which forms one side of the pressure chamber, and a cast ring, which includes the inlet bolting flange and the valve seat. The valve face is mounted in the center of the diaphragm and is spring-loaded against the valve seat. Through the center of the valve face is a small orifice which permits some of the fuel flowing through the valve to flow into the pressure chamber. An orifice through the cast wall of this chamber permits this chamber to fill with fuel and also permits a small amount of fuel to flow through the chamber. The pressure at the valve inlet overcomes the spring holding the valve closed and fuel spills through the valve between the valve face and the valve seat, around the cast legs which support the valve inlet.

Needle valves controlled by the floats in the float chamber shut off the flow of fuel through the pressure chambers when the fuel level in the tank raises the floats. When the flow through the chambers is shut off, the pressure builds up to equal the pressure at the inlet of the valve. The pressure in the chambers is multiplied into a total force, back of the diaphragm, which is greater than the total force imposed on the valve face by the pressure in the inlet line because of the greater area of the diaphragm compared to the area of the opening of the valve seat. This greater force coupled with the spring force causes the valve to close, shutting off the fuel at the desired



NOTE For clarity, components are not in proper relation to airplane design.

Pressure Refueling System Schematic
Figure 28-21-1



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Pressure Refueling System Component Locations
Figure 28-21-2

level. Spring-loaded relief valves protect the pressure chambers from surge pressures which are imposed on the valve when the flow is shut off.

Test provisions consist of two smaller pressure chambers, primary and secondary, paralleling the needle valves. The chambers have diaphragms, spring-loaded against test pressure, and stems extending to the needle valve linkage. When pressure is applied to the diaphragm, the stem actuates the linkage to close the needle, thus testing the valve for operation.

3. Wing Refueling Shutoff Valves

The wing refueling shutoff valves are installed on branches of the refueling line in the right and left center sections. Except for the different mountings and for the fact that control and test sections of the valve are in the wing tip, these valves are similar to the auxiliary tank valve.

The control lines connecting the wing refueling shutoff valves to the wing refueling level control pilot and vent valve are important to the proper operation of the valves, and care must be taken during installation and maintenance operations for their protection. Fuel flow through the lines is necessary for the shutoff valves to remain open. Kinked or deeply indented lines may contribute to partial- or no-flow conditions. Also, malfunctions of the wing refueling level control pilot and vent valve due to icing, foreign materials, or deterioration of parts will be reflected in the operation of the refueling shutoff valves.

The surge pressure relief valves are attached to the pressure chamber, while primary and secondary sensing lines connect the pressure chamber to the level control valves in the wing tip.

4. Wing Refueling Level Control Pilot and Vent Valve

The wing refueling level control pilot and vent valve tests a wing refueling shutoff valve, provides a vent for air in the wing during single-point refueling, seals the vent opening against loss of fuel, and controls the level to which the wing will be filled when pressure-refueled. The refueling test section of the valve consists of two pressure chambers, primary and secondary, placed on opposite sides. Diaphragms across the chambers have stems attached which bear the needle valve linkage. When either primary or secondary buttons are pressed, the corresponding diaphragm is extended, actuating the needle valve linkage and closing the port.

The vent valve portion has a rubber-sealed poppet valve opening into the wing. The stem of the valve extends into a third pressure chamber, where it is attached to a diaphragm separating the chamber. A spring on one side of the diaphragm loads the valve to close. Refueling pressure directly from the single-point receptacle at 12 to 55 psi opens the valve to vent. When the wing is filled, floats in the level control section close needle valves which direct refueling pressure from the sensing lines (from the refueling shutoff valve) to the opposite side of the diaphragm equalizing the refueling pressure. This allows the spring to close the vent valve. Should one of the floats fail to admit pressure, a shuttle-type check valve closes off the failed portion of the valve, ensuring operation. For dive venting, a

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WING REFUELING LEVEL CONTROL PILOT AND VENT VALVE (Cont'd)

pressure of 4 to 5 psi differential between the atmosphere and the tank will cause the valve to admit air to the tank.

The refueling level control pilot valve section has dual floats, each attached to a needle valve. Fuel raising the floats causes the valves to seat. This stops flow from the primary and secondary sensing lines from the refueling shutoff valve in the center section area of the wing. When pressure builds up in the sensing lines, the shutoff valve is actuated to close.

5. REFUELING TEST AND CONTROL VALVE

The refueling test and control valve is flush-mounted inside the right wing inboard leading edge, on the lower surface. Two control buttons govern the primary and secondary refueling valve test procedures. The third button, which is in the center, controls fuel flow to the fuselage fuel tank, and the fourth button unlocks and holds the center button during fuselage tank refueling.

The valve contains four plungers or pistons so sectioned as to provide channels for fuel flow to the primary and secondary test diaphragms. Each of the three buttons in line presses on two plungers at one time. The outside plungers service the primary and secondary systems in the wings and fuselage tank. The two center plungers control pressure refueling for the fuselage tank. In normal position, with no buttons depressed, both fuselage tank valve test diaphragms are pressurized; the wing valves are not.

When either the primary or the secondary button is pressed, flow is directed to the wing control valves and is cut off from one or the other test system to the fuselage tank valve, thus allowing the opposite half of the fuselage tank valve to be checked. Pressing the center button, after releasing it with the lock button (SHUTOFF AUX), depressurizes both primary and secondary diaphragms on the fuselage tank control valve and allows the valve to open and refuel the tank. A filter is installed in the line from the single-point receptacle to the control valve, mounted on an adjacent rib. It prevents contamination in the fuel from reaching the control valve.

6. AFT FUSELAGE TANK MAIN VENT AND FUEL TRANSFER VALVE

The aft fuselage tank main vent and fuel transfer valve, located in the bottom of the aft fuselage fuel tank, functions in 3 modes of operation. The valve allows the aux tank fuel to gravity feed to the wing tanks. The valve also allows wing fuel to expand into the aux tank, eliminating the possibility of wing tank rupture. In the event of overpressurization, the valve allows the wings to vent into the aux tank.

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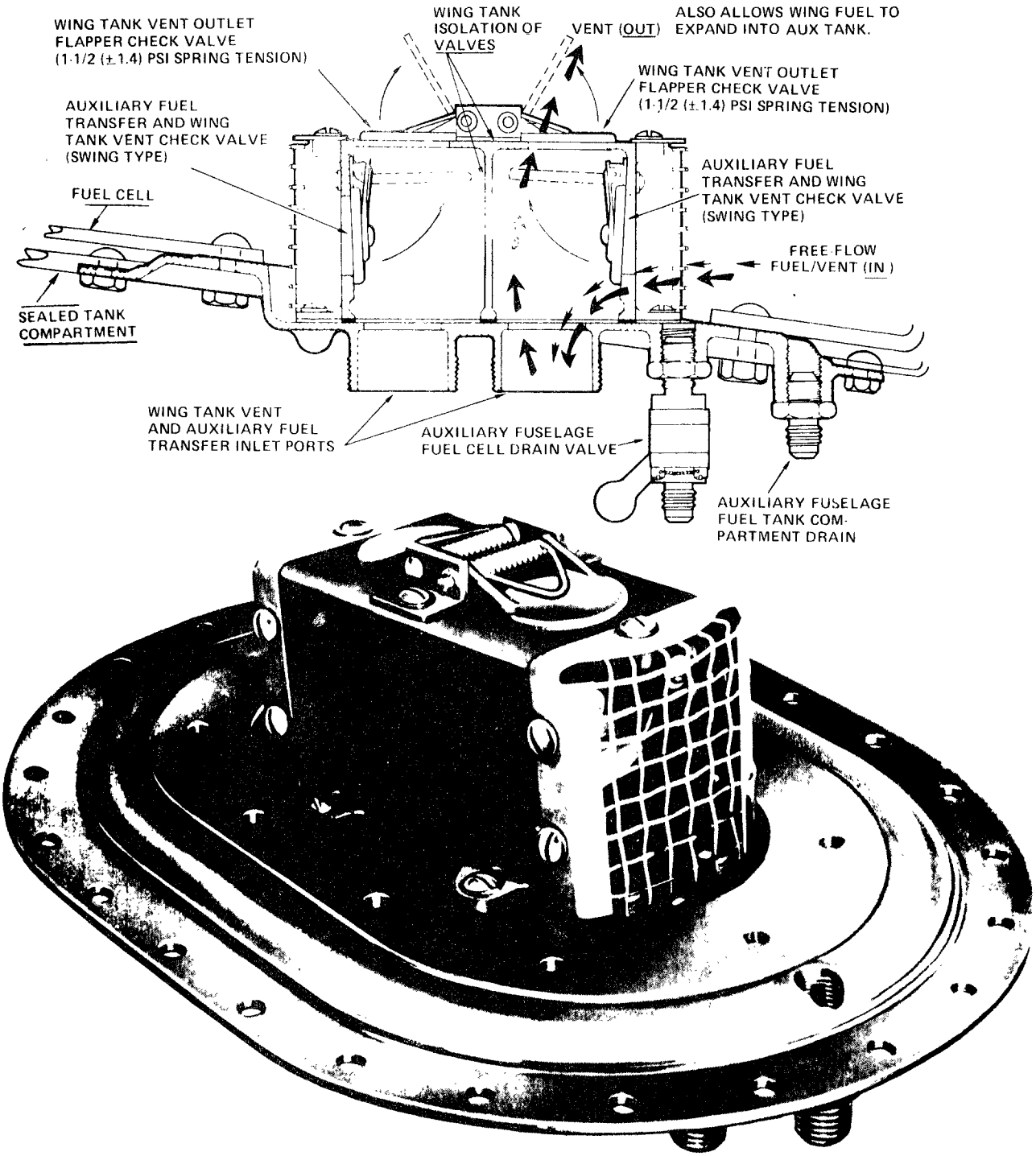
WING TANK VENT OUTLET FLAPPER CHECK VALVE (1-1/2 (±1.4) PSI SPRING TENSION)
WING TANK ISOLATION OF VALVES
VENT (OUT)
ALSO ALLOWS WING FUEL TO EXPAND INTO AUX TANK.
WING TANK VENT OUTLET FLAPPER CHECK VALVE (1-1/2 (±1.4) PSI SPRING TENSION)
AUXILIARY FUEL TRANSFER AND WING TANK VENT CHECK VALVE (SWING TYPE)
FUEL CELL
AUXILIARY FUEL TRANSFER AND WING TANK VENT CHECK VALVE (SWING TYPE)
FREE FLOW FUEL/VENT (IN)
SEALED TANK COMPARTMENT
WING TANK VENT AND AUXILIARY FUEL TRANSFER INLET PORTS
AUXILIARY FUSELAGE FUEL CELL DRAIN VALVE
AUXILIARY FUSELAGE FUEL TANK COMPARTMENT DRAIN

965-2-48-31

NEW

Figure 28-21-3 — Aft Fuselage Tank Main Vent and Fuel Transfer Valve

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PRESSURE REFUELING SYSTEM - TROUBLE SHOOTING

1. Trouble Shooting Pressure Refueling System

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. REFUELING PILOT AND VENT VALVES (FOR WING TANKS) OR FUSELAGE AUXILIARY TANK REFUELING LEVEL CONTROL VALVE (FOR FUSELAGE AUXILIARY FUEL TANK) DOES NOT STOP REFUELING FLOW WHEN REFUELING VALVE TEST BUTTONS ARE PRESSED.		
(1) Defective pilot and vent valve or valve test diaphragm.	Remove "B" nut on pressure line of suspected diaphragm. Check for flow of fuel when refueling control test button is pressed.	If flow exists, open chamber and check diaphragm. If O.K., check level control valve or shutoff valve.
(2) Ruptured shutoff valve diaphragm.	Listen for operation of valve at each wing. If valve operates and flow continues, valve is defective.	Replace valve.
(3) Test valve plunger seals cut or worn.	Remove "B" nuts at wing tip primary and secondary test fittings. Check for pressure.	Repair or replace test control valve.
B. REFUELING PILOT AND VENT VALVE VENTS FAIL TO OPEN.		
(1) Ruptured diaphragm in vent valve.	During refueling, check for airflow from wing tip lower surface. If no flow, remove "B" nut at valve and check for refueling pressure. If O.K., refer to remedy.	Replace valve.
(2) Loose "B" nut on line to refueling control receptacle.	If there was no pressure or pressure in previous isolation procedure, defuel airplane, remove access door, and check "B" nuts. If nut is tight, pressure-check line.	Tighten as necessary.
(3) Line passage blocked.	Check for flow at vent valve.	Repair or replace line.
C. REFUELING PILOT AND VENT VALVE FAILS TO CLOSE.		
(1) Broken valve return spring.	Check to see whether valve stays open without pressure on system.	Replace valve.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
D. FUSELAGE TANK FILLS INADVERTENTLY.		
(1) Failed control valve.	Check for fuel shutoff during refueling.	Replace valve.
(2) Wing vent valve fails to open.	Check for airflow from wing tip during refueling.	Refer to trouble B.
(3) Failed test and control valve "SHUTOFF AUX" control button.	With pressure refueling in progress, loosen primary and secondary sensing lines at B-nut connections on level control valve and check for fuel flow; if no flow exists, check for disconnected sensing lines between test and refueling control valve and refueling level control valve.	Connect or repair faulty sensing line, or lines, or replace faulty refueling shutoff test and control valve.
E. FUEL FLOWS FROM MAIN VENT OUTLET AFTER ALL LEVEL CONTROL VALVES HAVE CHECKED O.K.		
(1) Wing tip fill vent valve failed to open.	Refer to trouble B.	
F. FUSELAGE TANK FAILS TO FILL.		
(1) Test and control valve not operated properly.	Be sure to unlock "FILL AUX" button by pressing "SHUTOFF AUX" button first.	
(2) Test and control valve not functioning properly.	Check that there is no pressure at primary and secondary test diaphragms.	Replace test and control valve.
G. WING TANKS FAIL TO FILL.		
(1) Shutoff valves actuated.	With refueling pressure on airplane, check for fuel flow from lines separately disconnected at wing refueling level control pilot and vent valve, with and without test buttons being pressed.	If flow is O.K., replace shutoff valve. If no flow, check for source of line stoppage (kinks, dents, or foreign matter).

PRESSURE REFUELING SYSTEM - MAINTENANCE PRACTICES

1. Replacement of Fuselage Tank Refueling Level Control Valve

A. Removing Fuselage Tank Refueling Level Control Valve

This valve is accessible through aft lower fuselage equipment access doors 1 and 55. (Refer to Chapter 6, figure 6-00-3.)

(1) Remove environmental system ducting and units mounted in front of valve.

(2) Disconnect the three lines attached to valve and line support clips.

(3) Remove valve bracket support ring and two gaskets. Preserve for re-use.

(4) Remove the five countersunk screws attaching valve to cell.

(5) Lift valve from cell. Save "O" ring for re-use.

(6) Provide cover for cell opening. Do not use tape.

B. Installing Fuselage Tank Refueling Level Control Valve (See figure 28-21-401.)

2. Replacement of Wing Refueling Shutoff Valves

A. Removing Wing Refueling Shutoff Valves

A wing refueling shutoff valve is in the forward outer bay of each wing center section. The right valve is attached to a branched fitting and the left is mounted singly.

(1) Defuel airplane.

(2) Remove access door to bay to right or left of nose wheel well.

(3) Inside bay, remove square stress door on aft bulkhead of bay.

(4) Through access doors, remove access door on wing center section front spar. Save Stat-O-Seals on bolts and protect Gask-O-Seal on door.

(5) Disconnect lines to valve; then remove the six attach bolts.

(6) Remove valve from wing.

(7) Replace spar access door.

B. Installing Wing Refueling Shutoff Valves (See figure 28-21-402.)

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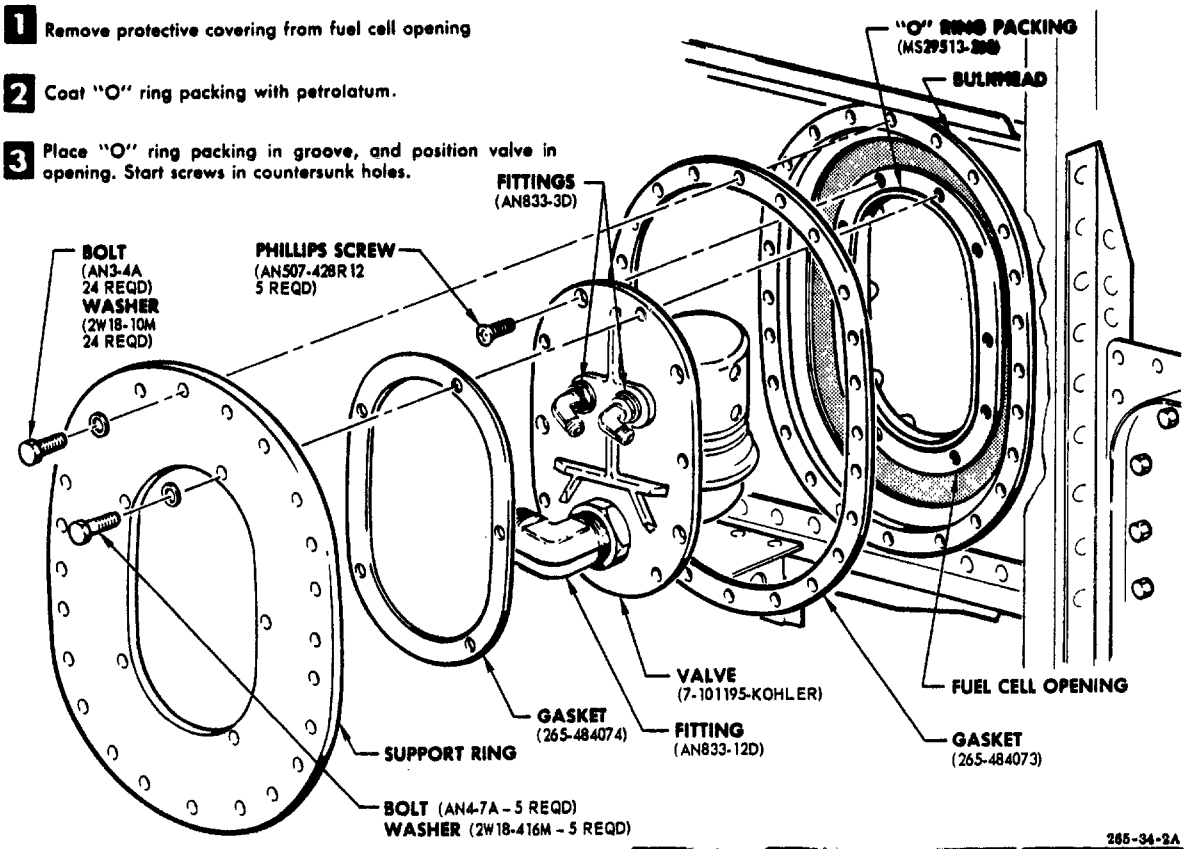
1 Remove protective covering from fuel cell opening

2 Coat "O" ring packing with petrolatum.

3 Place "O" ring packing in groove, and position valve in opening. Start screws in countersunk holes.



4 Torque five Phillips screws to 50-70 inch-pounds.

5 Position gaskets and valve support ring over valve and opening.

6 Start the five bolts through support ring, gasket, and valve mounting flange.

7 Start bolts (24) through support ring, gasket, and bulkhead.

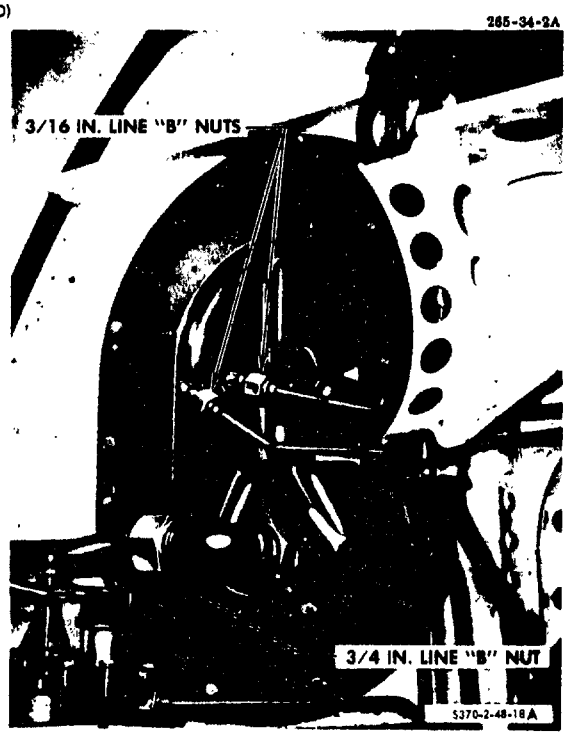
8 Draw up bolts into valve, and then into bulkhead.

9 Torque all bolts to 50-70 inch-pounds.

10 Connect "B" nuts to valve fittings. Torque 3/16-inch line "B" nuts to 25-35 inch-pounds; torque 3/4-inch line "B" nut to 300-500 inch-pounds.

11 Attach line support clamps.

12 Reinstall environmental system units.



Installing Fuselage Tank Refueling Level Control Valve
Figure 28-21-401

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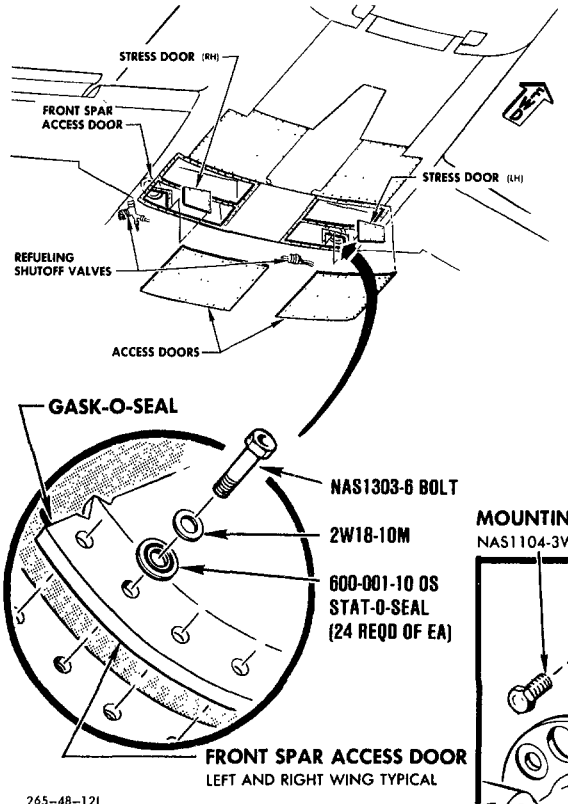
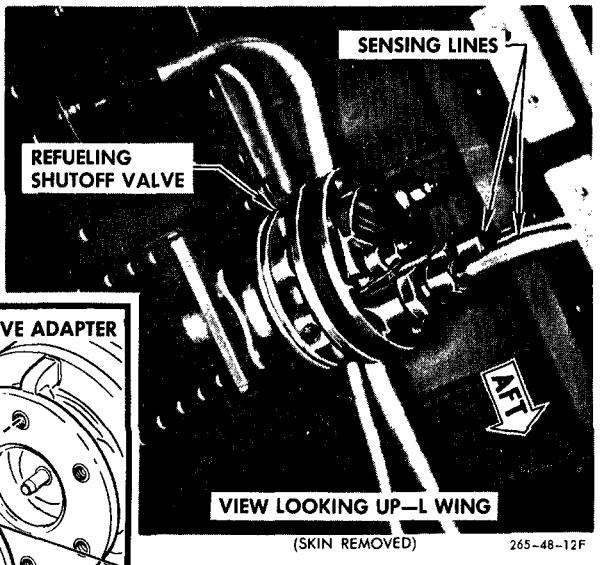
1. Check valve serviceability, and inspect valve cure date to determine that valve is not outdated.Check that two AN919-6D fittings are installed.
2. Remove access door from front spar.
3. Refer to note, and position valve so that it aligns with the proper sensing lines. Start bolt into mounting flange.

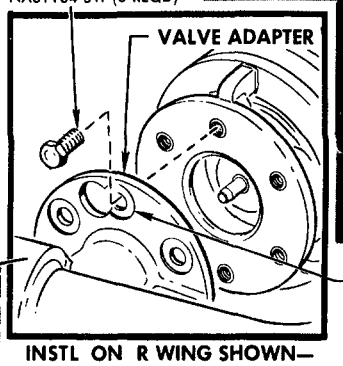
NOTE

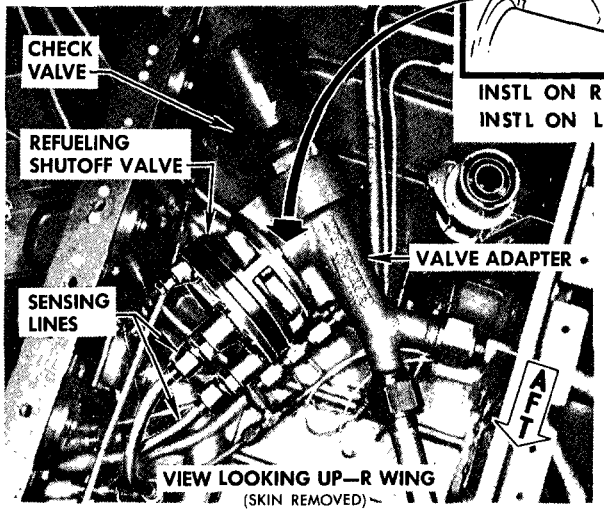
Two of the six bolt holes in the valve mounting flange are slotted. When removing the valve, loosen the two bolts in the slotted holes only a few turns; completely remove the other four bolts. The valve can then be rotated and removed (or installed). The two bolts that are to be installed in the slotted mounting holes should be installed on the replacement valve. The valve can then be properly positioned, with its fittings aligned with the proper sensing lines.

4. Start B nuts on lines, and tighten them by hand.
5. Tighten and torque mounting bolts to 50-75 inch-pounds; torque 3/8-inch line B nuts to 75-125 inch-pounds.









NOTE

The Gask-O-Seal is an integral part of the access door and is not replaceable. If the Gask-O-Seal is damaged in any way, the access door must be replaced.

6. Ensure that the area is clean. Ensure that the Gask-O-Seal and Stat-O-Seals are serviceable; then install access door. Torque bolts 25-35 inch-pounds.
7. Install stress door in fuselage bay and access door.

Figure 28-21-402 — Installing Wing Refueling Shutoff Valves

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REMOVAL AND INSTALLATION (Continued)

REPLACEMENT OF WING REFUELING LEVEL CONTROL PILOT AND VENT VALVE

Removing Wing Refueling Level Control Pilot and Vent Valve

1. Defuel airplane.
2. Remove wing tip.
3. Disconnect and cap all lines to valve.
4. Remove valve mounting bolts.
5. Lift and slide valve from wing, remove O-ring and air-dry before re-use.
6. Place a suitable cover over wing opening. Do not use tape. (Removing it could cause a static spark and explosion.)

Installing Wing Refueling Level Control Pilot and Vent Valve (see Figure 28-21-403).

REPLACEMENT OF REFUELING TEST AND CONTROL VALVE

Removing Refueling Test and Control Valve

1. Defuel airplane.
2. Remove access door, lower surface, wing inboard leading edge.
3. Through access door and lightening hole, disconnect the five B nuts from test and control valve.

CAUTION

Care should be exercised when disconnecting lines in this area so that small lines are not excessively bent or displaced, causing crossed lines or kinks that could result in line stoppage and the necessity for replacement.

4. Remove the 10 screws attaching valve to wing skin.
5. Remove valve Gask-O-Seal from wing interior.
6. Provide covers for openings, do not use tape.

Installing Refueling Test and Control Valve (see Figure 28-21-404). Cleaning and Replacing

Refueling Test and Control Valve Filter Element

This filter should require changing only at the test and control valve overhaul periods or in case of valve failure, since there is relatively no flow through the filter.

1. Loosen B nuts to both sides of filter (see Figure 28-21-404) unless valve is removed; if valve is removed, take off inlet line only.

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MAINTENANCE PRACTICES (Continued)

1. Observe fuel maintenance precautions.
2. Inspect the cure dates on new valve to determine that valve is not outdated, and check its serviceability.
3. Lubricate "O" ring packing, using petrolatum.
4. Place "O" ring in groove of valve mounting flange, and slide valve into position in wing.
5. Start attach bolts and tighten them evenly; then torque bolts to 25 inch-pounds.
6. Connect all lines to valve and start "B" nuts by hand; then torque 3/8-inch line "B" nuts to 75-125 inch-pounds, and torque 3/16-inch line "B" nuts to 25-35 inch-pounds.
7. Install wing tip.

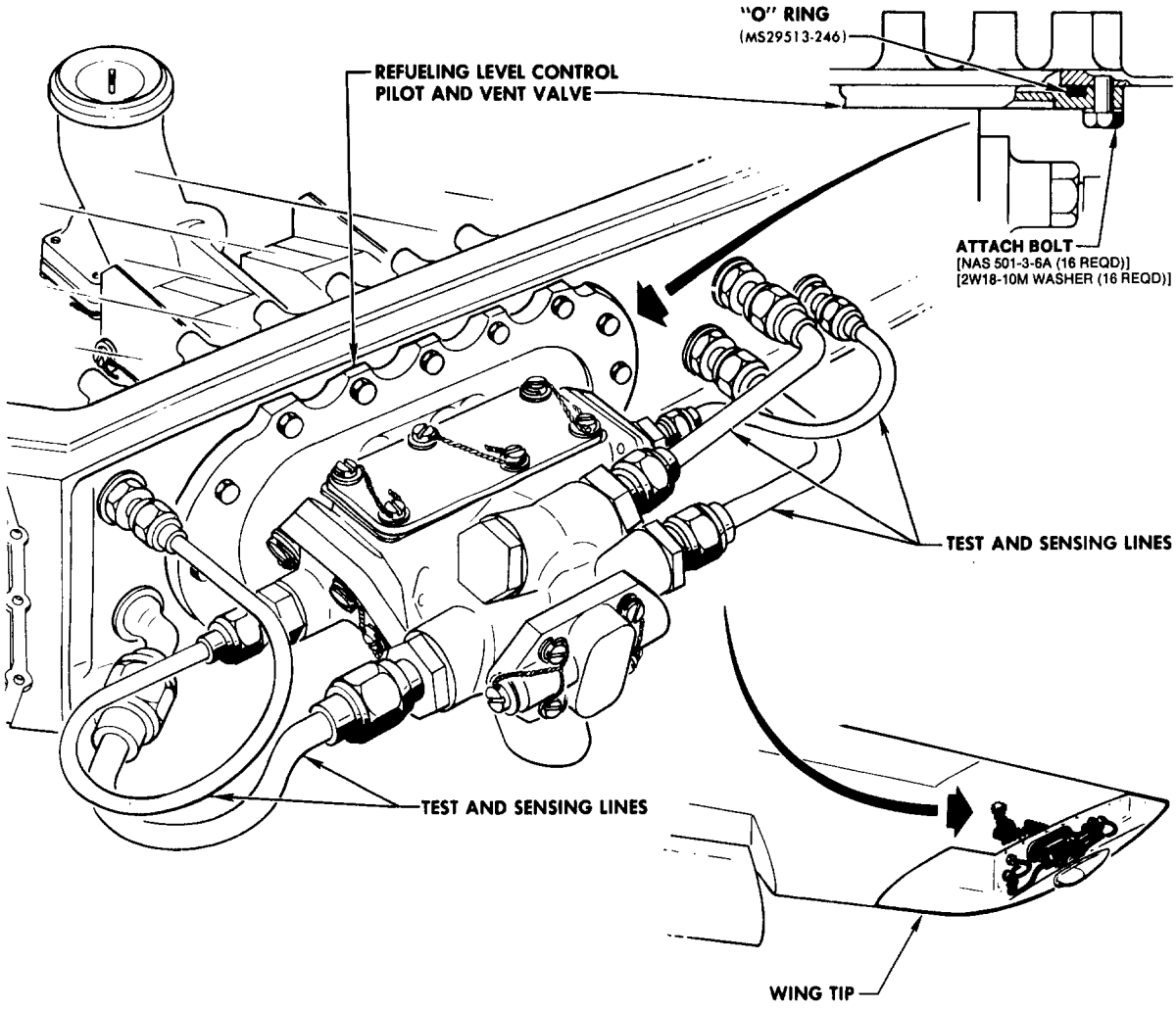

Figure 28-21-403 — Installing Wing Refueling Level Control Pilot and Vent Valve

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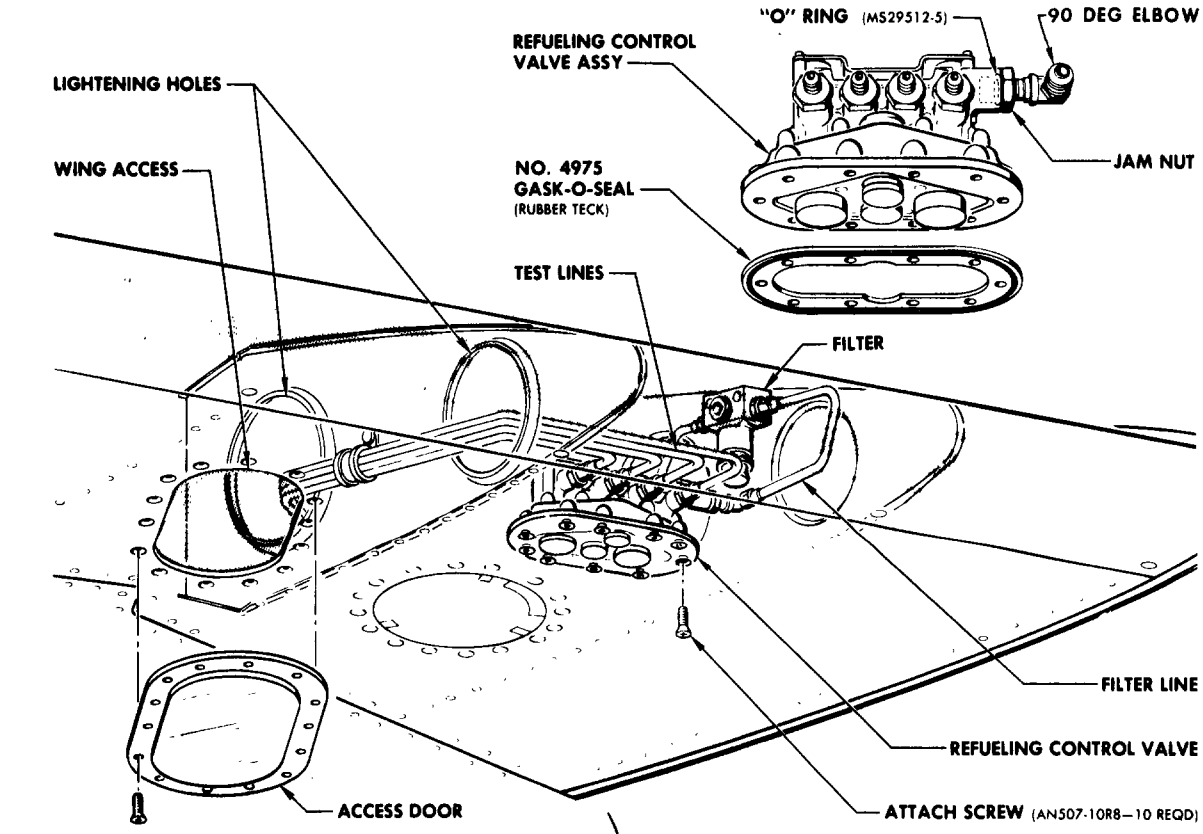
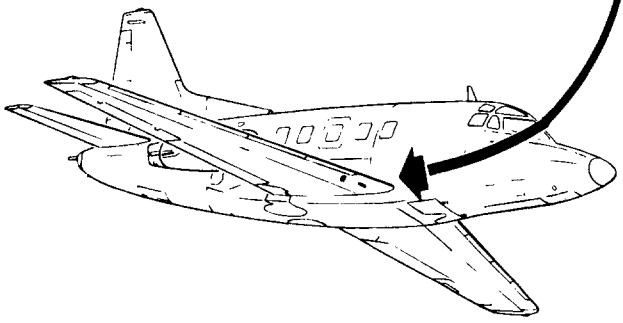
1. Observe fuel maintenance precautions.
2. Remove cover from wing opening.
3. Inspect valve serviceability and check the cure date, to determine that valve is not outdated.
4. Install 90-degree elbow and "O" ring in inboard (inlet) port of valve, leaving jam nut loose so that it can be aligned after it is installed.
5. Lubricate Gask-O-Seal (Rubber Teck No. 4975) with petrolatum, and position seal in wing.

NOTE Whenever the refueling control valve assembly is removed or replaced, the fuel filter element P/N042788 should also be replaced.



6. By reaching through access door and lightening hole, position valve in wing.
7. Check that Gask-O-Seal is in place; then start the 10 attach screws.
8. Tighten attach screws to 15 inch-pounds.
9. Align 90-degree elbow with line from filter, start "B" nuts, and tighten jam nut on elbow.
10. Connect remaining lines to valve. Torque 5/16-inch line "B" nut to 60-80 inch-pounds and torque the four 3/16-inch line "B" nuts to 25-35 inch-pounds.
11. Using low air pressure (25 psi), check to determine whether there are leaks in valve lines.
12. Check that wing is clean and install access door.



5370-2-48-21B

Figure 28-21-404 — Installing Refueling Test and Control Valve

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(2) Remove the two mounting bolts attaching valve to rib.

(3) Remove filter body from bay.

(4) Loosen nut and remove element from filter body.

(5) Slosh element and bowl in solvent, clean with air blast, and recheck for cleanness and general condition.

(6) Reinstall element and bowl.

(7) Tighten retaining nut and safety with lockwire.

(8) Place valve in position in wing and start "B" nuts.

(9) Install valve attach bolts.

(10) Torque "B" nuts 60 to 80 inch-pounds.

(11) If balance of installation is complete, check for foreign matter and install access door.

5. Replacement of Single-point Refueling Receptacle and Adapter

A. Removing Single-point Refueling Receptacle and Adapter

The single-point refueling receptacle adapter is accessible after the dust cover is removed. The receptacle housing is considered as part of the structure and is sealed by the groove sealing method. Removal of the housing would require removal of the right inboard wing leading edge.

(1) Make sure that fuel level is below refueling shutoff valves. (If airplane is left fueled, check valve may drip fuel.)

(2) Remove receptacle access door.

(3) Drain refueling line by manually opening poppet valve in receptacle adapter.

(4) Remove the six screws around receptacle adapter. The receptacle adapter and "O" ring should fall out of the recess.

B. Installing Single-point Refueling Receptacle and Adapter

(1) Check O-ring seal packing (MS29513-328) for serviceability.

(2) Coat "O"-ring with petrolatum (Federal Specification VV-P-236).

(3) Place "O" ring in groove and position receptacle adapter (MS24484-1) into housing.

(4) Install screws around receptacle adapter and attach cover chain.

(5) Install dust cover.

6. Replacement of Pressure Refueling Line Check Valve Assembly

A. Removing Pressure Refueling Line Check Valve Assembly

This valve is accessible through a fuselage bay and a door in the right center section front spar. Refer to procedure for removing wing refueling shutoff valve for information on disconnecting lines to right refueling shutoff valve.

(1) Disconnect the two lines (1-1/4 inch "B" nuts) to branched fitting.

(2) Remove the six bolts attaching check valve adapter fitting to rib.

(3) Work valve assembly in a circular motion to the rear and inboard. When it is free, remove it from bay.

NOTE: Care should be exercised to ensure that refueling line does not leave its slide fitting in the front spar and fall into wing. Recovery and repositioning of line may be difficult.

(4) Temporarily install acess door on front spar.

B. Installing Pressure Refueling Line Check Valve Assembly

(1) Remove center section access door.

(2) Lubricate "O" rings with petrolatum.

(3) Place "O" ring in groove of check valve adapter fitting.

(4) Check that arrow on check valve assembly is on top and pointing to rear when positioned in center section and that shutoff valve is to left (when facing in direction of arrow) of branched fitting.

(5) Check that refueling line is engaged in wing front spar fitting, then position check and shutoff valve assembly over refueling line into cutout in rib. Start all six mounting bolts; then tighten them.

(6) Connect wing refueling shutoff valve sensing lines. Torque 3/8-inch line "B" nut 75 to 125 inch-pounds.

(7) Connect refueling lines to branched fitting. Torque 3/4-inch line "B" nut 300 to 500 inch-pounds.

(8) Check area for foreign materials; then install front spar access door.

(9) Install stress door in fuselage bay.

(10) Install access door.

(11) Run operational check-out on pressure refueling system.

1. Check-out Procedure

A. Perform check-out and testing of pressure refueling system as follows:

(1) Connect single-point refueling source nozzle to single-point refueling receptacle and turn nozzle clockwise to hold it in place. Actuate nozzle lever clockwise to latch nozzle and open nozzle valve. (See figure 28-21-1.)

(2) Apply 25 psi fuel pressure to single-point pressure refueling receptacle. Make sure fuel flows into wing fuel tank system, but does not flow into fuselage tank. During filling operation, make sure venting occurs at vent underneath each wing tip and at airplane fuel vent outlet (screened) at tail cone.

(3) Depress "SHUTOFF AUX" and "FILL AUX" buttons of single-point refueling test and fuselage tank refueling control valve. Release "SHUTOFF AUX" button; then "FILL AUX" button. Make sure fuel flows into fuselage tank.

(4) Increase pressure to 50 (±5) psi and make sure fuel flow continues. Depress and release "SHUTOFF AUX" button and make sure that fuel flow to the fuselage tank stops, but continues to the wing tanks.

(5) With 50 (±5) psi fuel pressure applied to single-point pressure refueling receptacle, depress and release "PRI TEST" button of refueling test and fuselage tank refueling control valve. Make sure flow into wing tank system stops when button is depressed, and starts when it is released. Check that fuel is not flowing into the fuselage tank during this test.

(6) Perform step (5), except depress and release "SEC TEST" button instead of "PRI TEST" button. The identical fuel flow characteristics should occur.

(7) Depress "SHUTOFF AUX" and "FILL AUX" buttons. Release "SHUTOFF AUX" button, then "FILL AUX" button, and completely fill fuel tank system.

(8) Continue fuel filling until flow ceases automatically, signifying that tanks are full. Then press "SHUTOFF AUX" button. Maintain fuel pressure for a short period after flow has ceased and make sure no fuel overflows at any of the vent outlets.

(9) Depress poppet valve of single-point refueling receptacle. Check that fuel flows from receptacle at approximately 1/4 gpm.

CAUTION: CARE MUST BE TAKEN DURING THIS PROCEDURE TO PREVENT DAMAGE TO THE POPPET VALVE.

WARNING: PERSONNEL SHOULD REMAIN AT A SAFE DISTANCE FROM THE REFUELING RECEPTACLE WHEN THE POPPET VALVE IS INITIALLY DEPRESSED, SINCE A SMALL AMOUNT OF FUEL WILL FLOW FROM THE RECEPTACLE AT CONSIDERABLE PRESSURE.

(10) Check the following drain valves for proper operation:

(a) Wing section tanks
- Two in LH center section
- Two in RH center section
- Three in LH outer panels
- Three in RH outer panels

(b) Fuselage section lines
- One in LH fuel line
- One in RH fuel line
- Two in LH fuselage fuel cell fuel vent and transfer line
- Two in RH fuselage fuel cell fuel vent and transfer line
- One in bottom flange of fuselage cell
- One in RH fuel jettison line

NOTE: The amount of fuel required to fill the system should be approximately 1106 gallons (7189 pounds) as indicated on cockpit fuel quantity gages.

2. Testing Pressure Refueling System

The following procedures assist in locating and correcting the cause of a malfunction. Basic malfunctions encountered in the pressure refueling system occur because the system either fails to refuel, refuels too slowly, or fails to shut off. Some of the major causes of malfunctions are pressure leaks in the vent valve, primary and secondary test system, or primary and secondary sensing lines, due to loose "B" nuts, restrictions in the lines, or lower pressure output from the refueling unit than is indicated on the unit pressure gage. When tests of the fuel system are performed, the fuel level in the wing tanks should be below 1500 pounds in each tank.

A. Testing Refueling Pilot and Vent Valve System

(1) Defuel airplane, if necessary. (Refer to 28-10-2.)

(2) Remove wing tip.

(3) Disconnect "B" nut on vent valve pressure line (265-488201-27 left or 265-488201-37 right) from refueling pilot and vent valve, and loosen "B" nut on opposite end of line and rotate upward. (See figure 28-21-405.)

(4) Connect calibrated pressure gage to vent valve pressure line disconnected in step (3) and tighten "B" nuts.

(5) Connect calibrated pressure gage to pressure refueling nozzle. (Refer to Chapter 12, figure 12-13-201.)

(6) Remove single-point refueling control receptacle dust cover, insert nozzle grounding plug into jack on airplane, connect pressure refueling nozzle to single-point refueling control receptacle, and apply fuel pressure to start fuel flow into tanks. (Refer to 12-13-0.)

(7) Note pressure indication on refueling nozzle pressure gage; then check pressure indication on pressure gage connected to vent valve pressure line at wing tip. Pressure differential should not exceed 5 psi. If pressure differential is greater than 5 psi, the same check should be made at the opposite refueling pilot and vent valve in an effort to determine the area of line loss (leakage).

NOTE: The refueling pilot and vent valve should open between 12 and 55 psi. If the valve does not open upon application of 55 psi (maximum pressure), it is defective and must be replaced.

(8) Check for possible leakage at "B" nuts within wing tip tank area. To check for leakage, remove gravity refueling filler access cap, depress flapper, and then, with refueling pressure applied and the other mechanic depressing primary and secondary test buttons and holding them depressed, listen for sounds of leakage within wing tip tank areas.

(9) Since refueling pilot and vent valve is easier to remove than center wing access panels, remove valve and check all "B" nuts within wing tip tank areas for tightness before investigating other areas.

(10) To check for displaced O-ring seals at centerline wing rib (265-130073), defuel airplane, remove access covers 23 and 35 (refer to Chapter 6, figure 6-00-3) and the two forward drain valves, and check "O" rings on refueling pilot and vent valve pressure line on both sides of bulkhead. These lines can be seen when the drain valves are removed and a vaporproof flashlight is used.

NOTE: To check for leakage in O-ring seal area, attach one end of a flex hose to refueling pilot and vent valve pressure line at wing tip and other end of hose to a container of fuel. Raise container high enough above wing tip to induce fuel flow into refueling pilot and vent valve pressure line. This pressurizes the O-ring seals and shows leakage if the "O" rings are not seated.

(11) Check refueling pilot and vent valve pressure line connections in outboard ends of wing center section by removing fuel valve access doors. (See figures 28-21-402 and 28-21-405.)

(12) Check refueling pilot and vent valve pressure line connections at, and adjacent to, single-point refueling control receptacle by removing wing leading edge fuel bay access door. (See figure 28-21-404.)

(13) At conclusion of tests, remove test equipment used for testing; then reconnect lines, torque "B" nuts, and install drain valves, access covers, and doors that were removed.

(14) Perform operational check of pressure refueling system.

B. Testing Pressure Refueling Primary and Secondary Test System

(1) Defuel airplane, if necessary.

(2) Remove single-point refueling control receptacle dust cover, insert nozzle grounding plug into jack on airplane, connect pressure refueling nozzle to single-point refueling control receptacle, and apply fuel pressure to start fuel flow into tanks. (Refer to Chapter 12, figure 12-13-201.)

(3) If fuel flow is audible from top of fuselage auxiliary fuel tank, discontinue refueling. Check refueling pilot and vent valves for venting. If one or both valves do not vent, the fuselage auxiliary fuel tank will fill inadvertently. If refueling pilot and vent valve does not vent, refer to paragraph A.

(4) If both refueling pilot and vent valves are venting, ensure that "FILL AUX" selector button is not engaged. Remove gravity refueling cap from fuselage auxiliary tank and listen for audible fuel flow. If fuel flow is audible from top of tank, discontinue refueling.

(5) Disconnect primary test line from fuselage auxiliary fuel tank refueling level control valve and connect a pressure gage to test line. (See figure 28-21-401.) Apply refueling pressure; check reading on refueling nozzle gage and test line gage. The pressure differential should not exceed 5 psi. If pressure differential is greater, check test line for leaks at all "B" nuts within fuselage and at wing fittings. A reading of 12 psi or greater indicates that the primary float should be raised. If the fuselage auxiliary fuel tank continues to fill from the fuselage auxiliary tank refueling level control valve, the valve must be replaced. Pressure should be available in the test lines as soon as refueling pressure is applied.

(6) Repeat step (5), using secondary test line. If further trouble shooting of system for failure to shut off during test is to be accomplished before replacement of a defective fuselage auxiliary tank refueling level control valve, isolate fuselage auxiliary fuel tank by disconnecting primary test, secondary test, and single-point refueling lines from fuselage auxiliary tank refueling level control valve, and cap all three lines. (See figure 28-21-401.)

(7) Reapply refueling pressure and check that right and left wing refueling shutoff valves close when primary and secondary test buttons are depressed. If they do not, proceed to the following steps.

(8) Remove wing tip.

(9) Disconnect affected test line (primary or secondary) from refueling pilot and vent valve and rotate line upward. (See figure 28-21-405.)

(10) Connect pressure gage to open end of test line and tighten "B" nuts.

(11) Have helper apply refueling pressure while depressing appropriate primary or secondary test button. Note pressure on wing tip gage. A minimum of 9 psi is required to lift the float. If 9 psi or greater is obtained, perform check of primary or secondary sensing lines according to steps (17) through (23). If fuel shutoff valves close during sensing line check, the float valve or diaphragm is defective in the refueling pilot and vent valve.

NOTE: If less than 9 psi is obtained, there is a leak or complete obstruction in the test line or test selector valve.

(12) Check for possible leakage at "B" nuts within wing tip tank area. To check for leakage, remove gravity refueling filler access cap, depress flapper, and then, with refueling pressure applied and other mechanic depressing primary and secondary test buttons and holding them depressed, listen for sounds of leakage within wing tip tank area.

(13) Since refueling pilot and vent valve is easier to remove than center wing access panels, remove valve and check all "B" nuts within wing tip for tightness before investigating other areas. (See figure 28-21-405.) If loose "B" nuts are found, tighten "B" nuts and reinstall pilot and vent valve.

(14) To check for displaced O-ring seals at centerline wing rib (265-130073), defuel airplane, remove access covers 23 and 35 and the two forward drain valves, (refer to Chapter 6, figure 6-00-3), and check "O" rings on primary and/or secondary test lines. These lines can be seen when the drain valves are removed and a vaporproof flashlight is used.

NOTE: To check for leakage in O-ring seal area, attach one end of a flex hose to primary and/or secondary test line at wing tip and other end of hose to a container of fuel. Raise container high enough above wing tip to induce fuel flow into vent valve pressure line. This pressurizes the O-rings and shows leakage if the "O" rings are not seated.

(15) Check primary and/or secondary test line "B" nuts in outboard ends of wing center section by removing fuel valve access doors. (See figures 28-21-402 and 28-21-405.)

(16) Check primary and/or secondary test line "B" nuts at, and adjacent to, single-point refueling control receptacle by removing wing leading edge fuel bay access door. (See figure 28-21-405.)

(17) Disconnect primary and secondary sensing lines from refueling pilot and vent valve, and cap refueling pilot and vent valve ports.

(18) Apply refueling pressure.

(19) Close off primary sensing line with thumb. Fuel flow to the wing should cease. Release thumb pressure to open sensing line.

(20) Close off secondary sensing line with thumb. Fuel flow to the wing should cease. If the shutoff valve fails to close when either of the sensing lines is closed off, there is leakage in the sensing lines, or the shutoff valve diaphragm is defective and the valve must be replaced.

(21) Since refueling pilot and vent valve is easier to remove than center wing access panels, remove valve and check all "B" nuts within wing tip tank area for tightness before investigating other areas. (See figure 28-21-405.) If loose "B" nuts are found, tighten "B" nuts and reinstall refueling pilot and vent valve, leaving primary and secondary sensing lines disconnected.

(22) Remove fuel valve access doors and check primary and/or secondary sensing line B-nut connections to refueling shutoff valve in outboard ends of wing center section. If no loose "B" nuts are found, replace refueling shutoff valve access doors.

(23) If loose "B" nuts on sensing lines of refueling shutoff valve are found, tighten "B" nuts, reinstall access door removed in step (22), and reapply refueling pressure. Close off primary and/or secondary sensing line with thumb. Fuel flow should shut off. If it does not shut off, defuel airplane and replace refueling shutoff valve.

(24) At conclusion of tests, remove test equipment, reconnect lines, torque "B" nuts, and install drain valves and access doors that were removed. Perform operational check of pressure refueling system.

C. Sensing Line Check

(1) Defuel airplane if necessary.

(2) Disconnect primary and secondary sensing lines from both pilot and vent valves, and rotate lines downward. Provide containers at each wing tip to catch fuel.

(3) Add 1000 pounds of fuel to airplane system. An equal amount of fuel should emit from both the primary and secondary sensing lines at both wing tips into the four containers.

(4) Check airplane fuel gages after 1000 pounds of fuel has been added. The gages should read within 50 pounds of the added fuel in the tanks.

NOTE: If gages do not read within 50 pounds of the added fuel in tanks, do steps (5) and (6) to check sensing lines for obstruction.

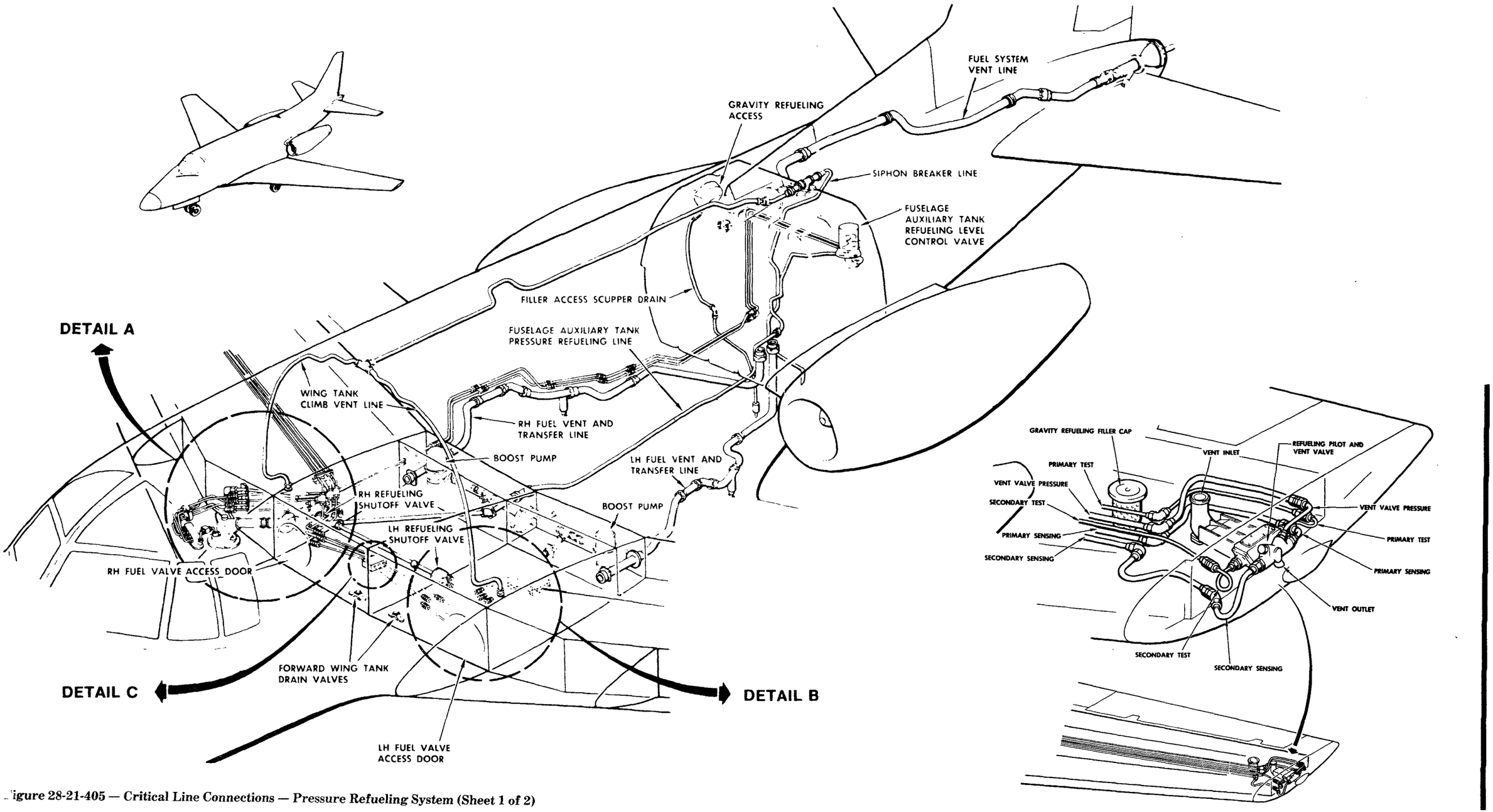

Figure 28-21-405 — Critical Line Connections — Pressure Refueling System (Sheet 1 of 2)

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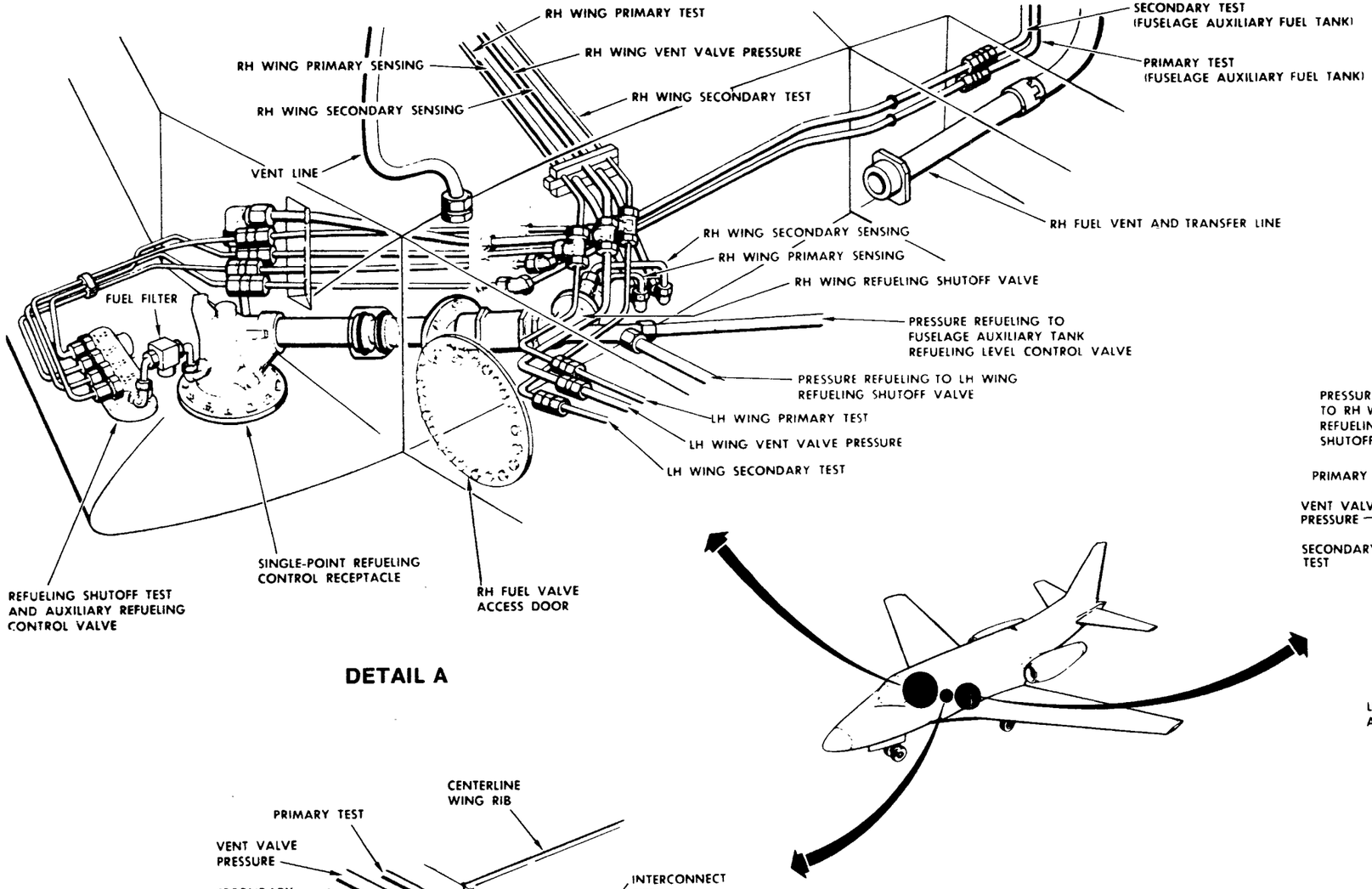
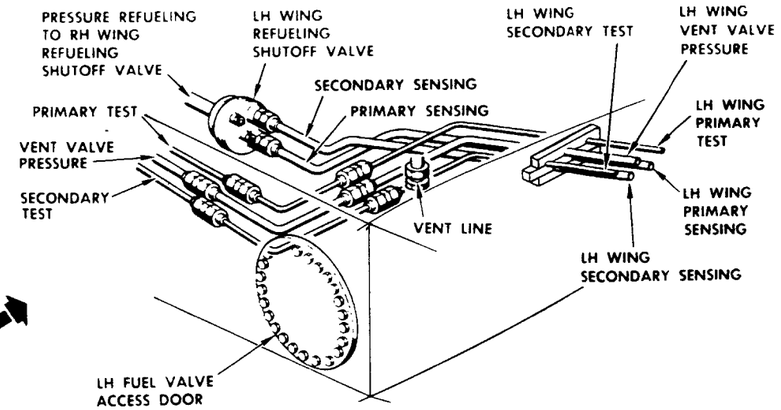
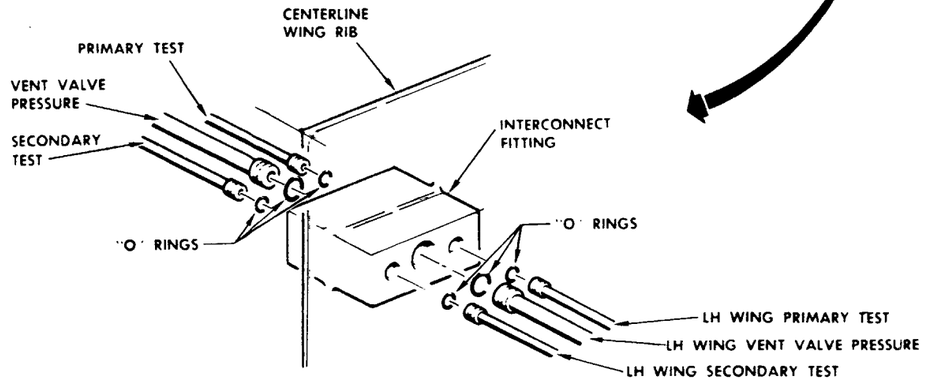


DETAIL A



DETAIL B



DETAIL C

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Figure 28-21-405 — Critical Line Connections — Pressure Refueling System (Sheet 2 of 2)

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(5) Defuel airplane; then disconnect sensing lines at fuel shutoff valve.

(6) Attach one end of a transparent hose to sensing line at wing tip and other end to a container of fuel. Raise container high enough to induce fuel flow. If there is no obstruction, the fuel can be seen to lower in the container.

(7) If sensing line is unobstructed, replace fuel shutoff valve.

(8) At conclusion of tests, remove test equipment used; then reconnect lines, torque "B" nuts, and install drain valves, access covers, and doors that were removed. Perform operational check of pressure refueling system.

D. Test Line Flow Check

(1) Defuel airplane if necessary.

(2) Apply refueling pressure. Depress first the primary and then the secondary test buttons. Fuel flow to wing tanks should cease.

(3) If fuel flow does not cease when either test button is depressed, stop refueling immediately and check "B" nuts within wing tip, at single-point refueling valve, and center wing areas for leakage. If no "B" nuts are found loose, proceed with step (4).

(4) Disconnect primary and secondary test lines at both vent valves. Add fuel. No fuel should emit from the lines. Have assistant depress first the primary, then the secondary test buttons; fuel should emit from the respective test lines.

(5) If fuel still does not flow from these lines, they should be checked for restriction. After ensuring lines are not restricted, again add fuel and depress primary and secondary test buttons. It may take as long as 2 minutes for refueling to commence after the test buttons have been depressed and released, indicating a possible restricted or defective refueling valve.

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(6) To check for restricted or defective refueling valve, disconnect a primary test line at the vent valve. Attach one end of a transparent hose to the test line and the other end to a container of fuel. Raise the container high enough to induce fuel flow. If there is no restriction, the fuel can be seen to lower. Reconnect this line; then repeat procedure with secondary test line.

(7) If there is a restriction in vent of refueling valve, defuel airplane and check for restriction or a defective refueling shutoff test and auxiliary refueling valve.

(8) At conclusion of tests, remove test equipment used; then reconnect lines, torque "B" nuts, and install drain valves, access covers, and doors that were removed. Perform operational check of pressure refueling system.

FUEL JETTISON SYSTEM - DESCRIPTION AND OPERATION - NA-370-2 THROUGH -6, -8 AND -9 AIRPLANES

1. General

The fuel jettison system consists of a jettison line (routed overboard) and items connecting this line to the fuel supply through the engine fuel boost lines. (See figures 28-30-1 and 28-30-2.) A single three-position selector switch, two solenoid-operated jettison shutoff valves, relays, and caution lights aid operation of the system. The jettison line extends from the jettison valves to the fuel vent fitting in the tail cone. The fuel jettison switch, in the cockpit, selects automatic or manual jettisoning of fuel. The switch has three positions: AUTO, MAN, and OFF. When the switch is placed at AUTO, it stays there until it is changed by the pilot, and fuel is jettisoned. Float switches in the AUTO position circuit automatically limit the level to which fuel will jettison. (This ensures that enough fuel will be retained for a specific flight at specified altitude and duration.) The MAN position is spring-loaded to OFF. As long as the switch is held in the MAN position, fuel is jettisoned until the fuel supply is exhausted. Selecting either the AUTO or the MAN position of the fuel jettison switch actuates relays connected to the cross-feed valves, closing them when the jettison valves open. Contacts in the jettison valve switches control "FUEL JETT OPEN" caution lights.

The fuel boost pumps supply fuel jettison pressure in addition to the pressure required to transfer fuel to the engines. With both pumps operating, sufficient pressure is maintained for normal engine requirements and jettisoning. A check valve is installed downstream of each jettison shutoff valve.

Operation of the fuel jettison system is done in flight with both of the fuel boost pumps in operation. The amount of fuel jettisoned will be only that amount pumped which is in excess of engine demand. After the fuel jettison switch is placed at AUTO, the supply of fuel is reduced to a predetermined level. If the MAN position of the switch is selected, the switch must be held until the desired quantity of fuel is jettisoned. Jettison should not be continued to a point where the remaining fuel supply becomes dangerously low. If the tanks contain sufficient fuel to raise the floats in the fuel level control switches [138 (±31) gallons each tank], the jettison level control relay contacts will be closed. (See figure 28-30-3.) Then, when the jettison switch is placed at AUTO, current flows from the "FUEL JETTISON - LH" and "FUEL JETTISON - RH" circuit breakers on the dc essential bus through the normally open (now closed) contacts of the right and left jettison control relays to the jettison valve solenoids, and the jettison valves open. When either or both jettison valves are "cracked" open, the "FUEL JETT OPEN" caution light switch closes, and this light and the master caution light illuminate. When the jettison switch is at MAN or AUTO, the right and left jettison fuel control relays are energized. The fuel pump cross-feed valve is then energized open. The right and left jettison fuel control relays also de-energize the right and left fuel boost

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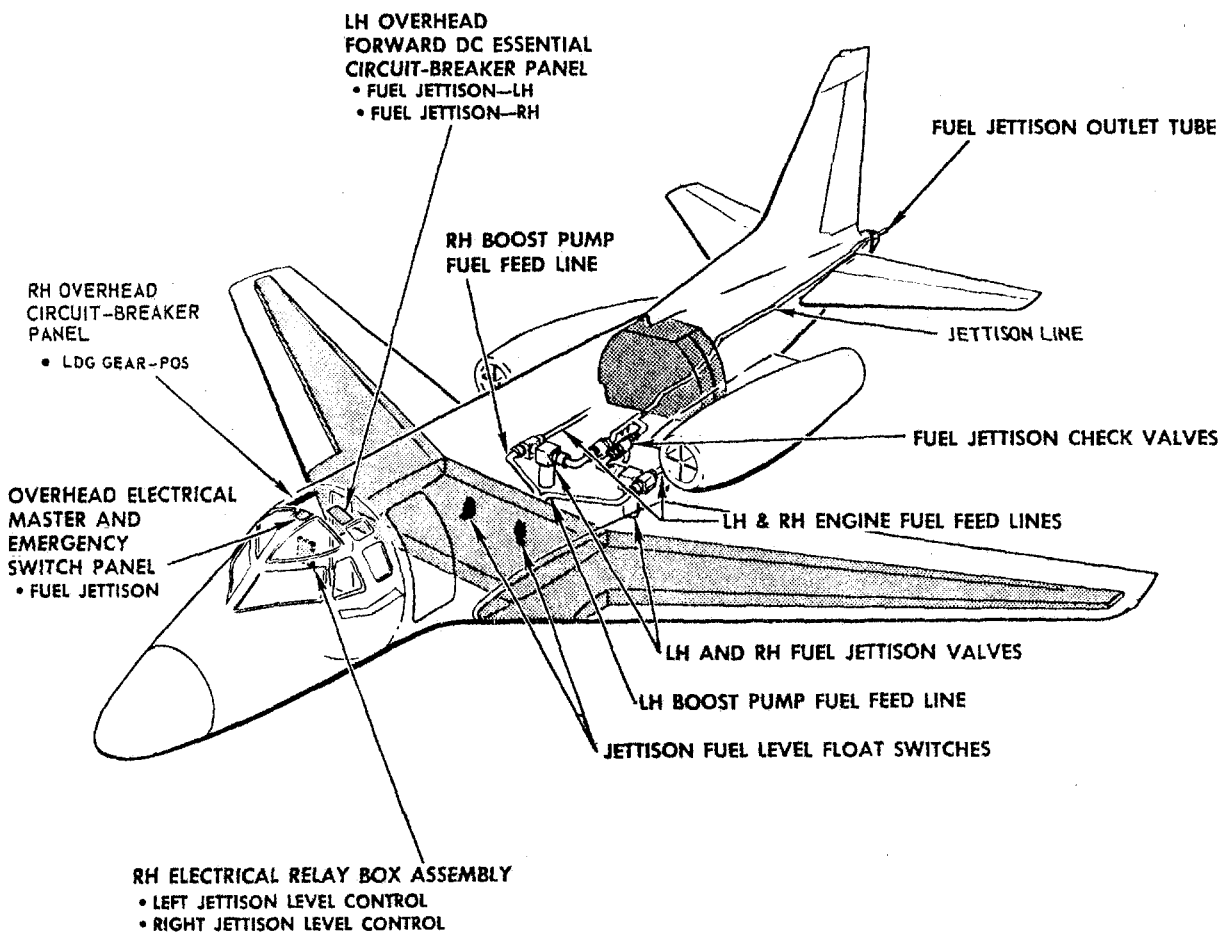
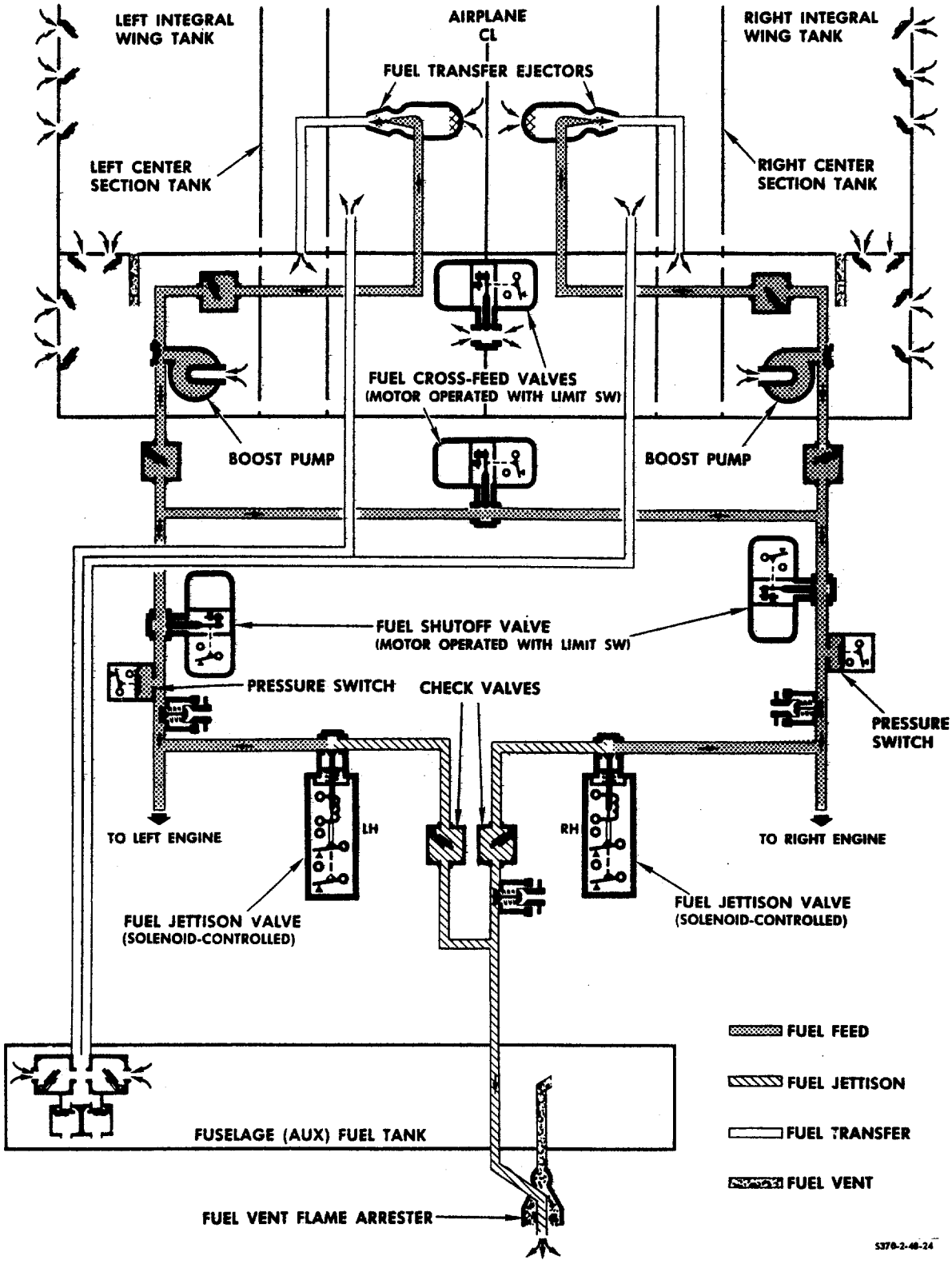


Fuel Jettison System Component Locations
Figure 28-30-1



Fuel Jettison System Schematic
Figure 28-30-2

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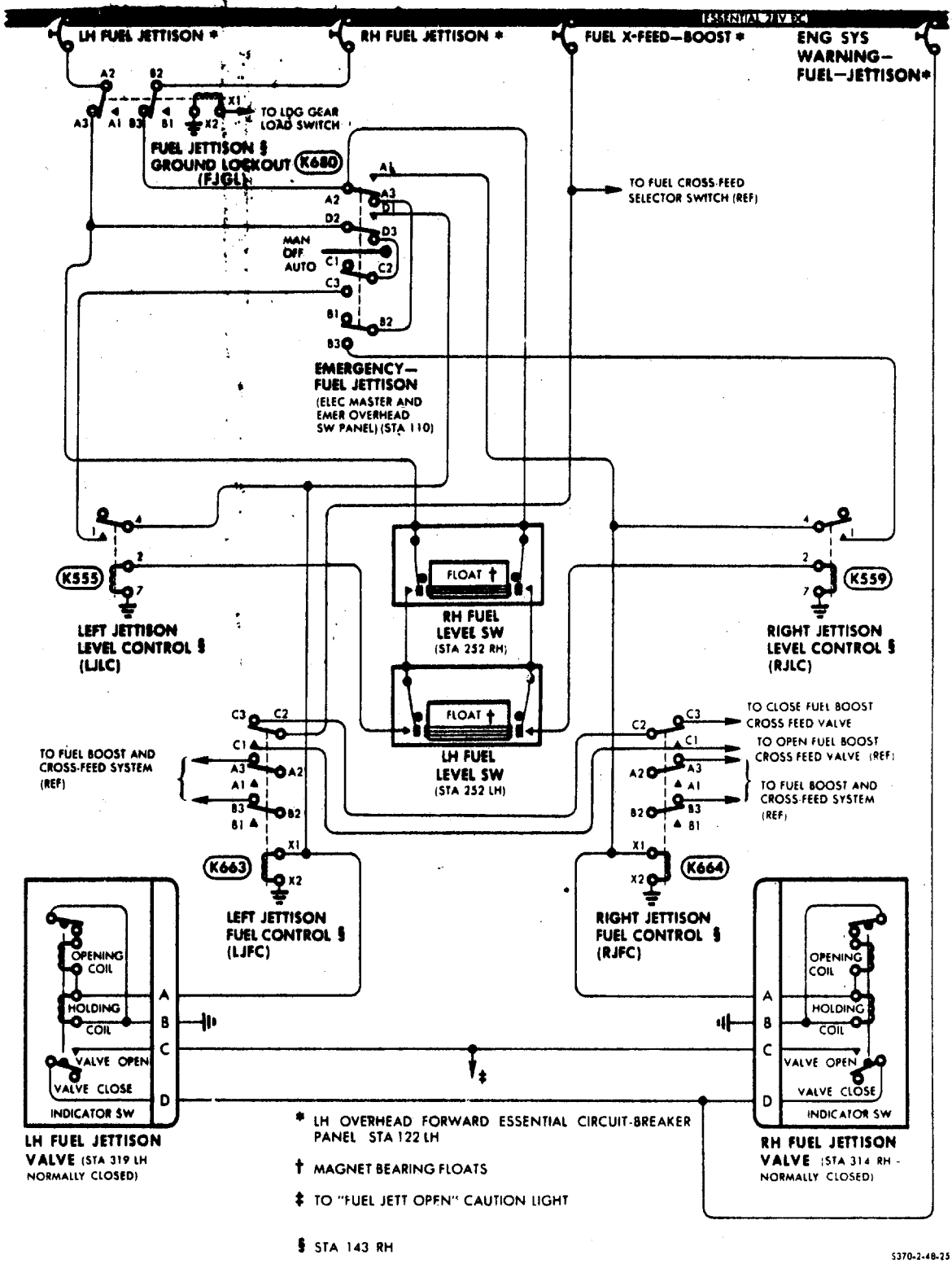

ESSENTIAL 7 LV BUS
LH FUEL JETTISON *
RH FUEL JETTISON *
FUEL X-FEED—BOOST *
ENG SYS WARNING—FUEL—JETTISON*
TO LDG GEAR LOAD SWITCH
FUEL JETTISON § GROUND LOCKOUT (FJGL) K680
TO FUEL CROSS-FEED SELECTOR SWITCH (REF)
MAN
OFF
AUTO
EMERGENCY—FUEL JETTISON
(ELEC MASTER AND EMER OVERHEAD SW PANEL) (STA 110)
K555
LEFT JETTISON LEVEL CONTROL § (LJLC)
FLOAT †
RH FUEL LEVEL SW (STA 252 RH)
LH FUEL LEVEL SW (STA 252 LH)
K559
RIGHT JETTISON LEVEL CONTROL § (RJLC)
TO FUEL BOOST AND CROSS-FEED SYSTEM (REF)
TO CLOSE FUEL BOOST CROSS FEED VALVE
TO OPEN FUEL BOOST CROSS FEED VALVE (REF)
TO FUEL BOOST AND CROSS-FEED SYSTEM (REF)
K663
LEFT JETTISON FUEL CONTROL § (LJFC)
K664
RIGHT JETTISON FUEL CONTROL § (RJFC)
OPENING COIL
HOLDING COIL
VALVE OPEN
VALVE CLOSE
INDICATOR SW
LH FUEL JETTISON VALVE (STA 319 LH NORMALLY CLOSED)
RH FUEL JETTISON VALVE (STA 314 RH - NORMALLY CLOSED)
* LH OVERHEAD FORWARD ESSENTIAL CIRCUIT-BREAKER PANEL STA 122 LH
† MAGNET BEARING FLOATS
‡ TO "FUEL JETT OPEN" CAUTION LIGHT
§ STA 143 RH
5370-2-48-25

Fuel Jettison System Electrical Schematic
Figure 28-30-3

relays. These fail-safe-to-close relays actuate to ensure that the fuel boost pumps stay on and thus provide jettison pressure, regardless of throttle switch position. If the engine master switches are at ON (with the jettison switch in AUTO or held in MAN), fuel will jettison in the air. When on the ground, a ground lockout system prevents inadvertent jettison of fuel.

When the fuel drops to the level of the dual-element float switches, the jettison level control relays are de-energized and open the circuit to the jettison valves, allowing them to close. Also, the jettison fuel control relays are de-energized and the pump cross-feed valves can close. When the valves reach their closed position, all caution lights go out.

When the fuel jettison switch is held at MAN, the system bypasses the jettison level control switches and relays, allowing fuel jettison to continue. Releasing manual pressure on the switch allows the valves to return to the closed position.

2. Fuel Jettison Valves

The fuel jettison valves are solenoid-operated gate shutoff valves. The left valve is mounted forward of the bulkhead; the right valve is mounted aft of the bulkhead at station 316. The valves are normally closed. They are spring-loaded to the closed position and electrically actuated to open. Position switches in the valves illuminate the jettison valve open light, in the cockpit, when either or both valves are not fully closed. The valve solenoid circuit contains two coils, a pull coil and a holding coil. When the valves reach the open position, a switch is opened and the pull coil is de-energized. The pull coil requires 18 amperes at 28 volts and 70°F. The holding coil draws 0.9 ampere at 28 volts and 70°F. Minimum operating voltage is 18 volts dc. Operating time is about 1/2 second to open or close at 27 volts dc. Wiggins-type fuel couplings are used to connect the valves to the fuel lines.

3. Fuel Jettison Level Control Float Switches

Two dual-element float switches (one in each wing tank) limit the level to which fuel will be jettisoned when the fuel jettison switch is positioned at AUTO. Each dual contact is in a hermetically sealed tube. One contact arm of each pair of contacts is of a material which is attracted by a magnet. A float surrounding the sealed tube carries a magnet which pulls the contacts open when the float approaches the low point. When the fuel reaches a predetermined level and contacts either switch, it breaks the jettison valve open circuit and permits the solenoid valve for that particular wing tank to close. These switches are accessible through doors in the center section rear spar and booster pump sump bulkhead access door. Flexible conduits connect the switches to connectors in the rear spar. Switch continuity may be checked at the rear spar connector.

FUEL JETTISON SYSTEM - TROUBLE SHOOTING

1. Trouble Shooting Fuel Jettison System

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. JETTISON VALVES FAIL TO OPEN.		
(1) Fuel level too to jettison with jettison switch at AUTO.	Check fuel level or select jettison switch MAN position.	Make sure tanks are half full or more.
(2) Level control switch float saturated or otherwise defective.	Fill tanks and check for proper contact through switches.	Replace switch assembly.
B. JETTISON VALVES FAIL TO CLOSE.		
(1) Foreign matter caught in gate channel of valve.	Remove valve and check visually.	Clean valve and flush lines. Reinstall valve.
(2) Circuit breaker out.	Make visual check.	Push circuit breaker in.
(3) Valve return spring weak or broken.	Remove valve and check operation.	Replace valve.
(4) Valve inoperative.	Check for voltage to pin "A" when switch position is selected manually.	Replace valve.
C. LIGHT STAYS ON AFTER FUEL JETTISON SWITCH IS PLACED AT OFF.		
(1) Valves not completely closed.	Tap valve body to aid seating.	Replace valve.
(2) Valve light switch defective.	Disconnect plug to valve; light should go out.	Replace valve.
(3) Open circuit in relay.	Run continuity check on relay.	Replace relay.
D. NO LIGHT WHEN JETTISON VALVE IS OPEN.		
(1) Switch contacts open at pin "C" on jettison valve.	Check with ohmmeter.	Replace valve.
(2) Indicator light inoperative.	Check push-to-test light.	Replace lamp.

FUEL JETTISON SYSTEM - MAINTENANCE PRACTICES

1. Replacement of Fuel Jettison Valves

   A. Removing Left and Right Fuel Jettison Valves

      The left fuel jettison shutoff valve is mounted forward of the bulkhead; the right jettison valve is mounted aft of the bulkhead at station 316.

      (1) Determine that fuel shutoff valves are closed; then disconnect battery and external power.

      (2) Provide a closed receptacle to catch fuel.

      (3) Remove access door 52. (Refer to Chapter 6, figure 6-00-3.)

      (4) Remove electrical disconnect plug from valve. Cap or bag plug and receptacle.

      (5) Remove safety wire and disconnect fuel line coupling to valve; prepare to catch fuel draining from line.

      (6) Support disconnected valve and remove support clamps; then remove valve.

      (7) Provide covering for openings into fuel system.

   B. Installing Left and Right Fuel Jettison Valves (See figure 28-30-401.)

2. Replacement of Fuel Jettison Level Control Float Switches

   A. Removing Fuel Jettison Level Control Float Switches

      (1) Defuel airplane.

      (2) On right-hand installation, relieve aileron control tension; then remove aileron bell-crank interconnecting rod and outboard bolt, nut, and washer from right and left aileron bell-crank brace. Push brace up until access door is cleared.

      CAUTION: FAILURE TO RELIEVE AILERON CONTROL CABLE TENSION BEFORE DISCONNECTING THE INBOARD SECTOR SUPPORT LINK MAY CAUSE DAMAGE TO THE AILERON CABLE SECTOR SUPPORT FITTING.

      (3) Remove rear spar access door. (See figure 28-30-402.)

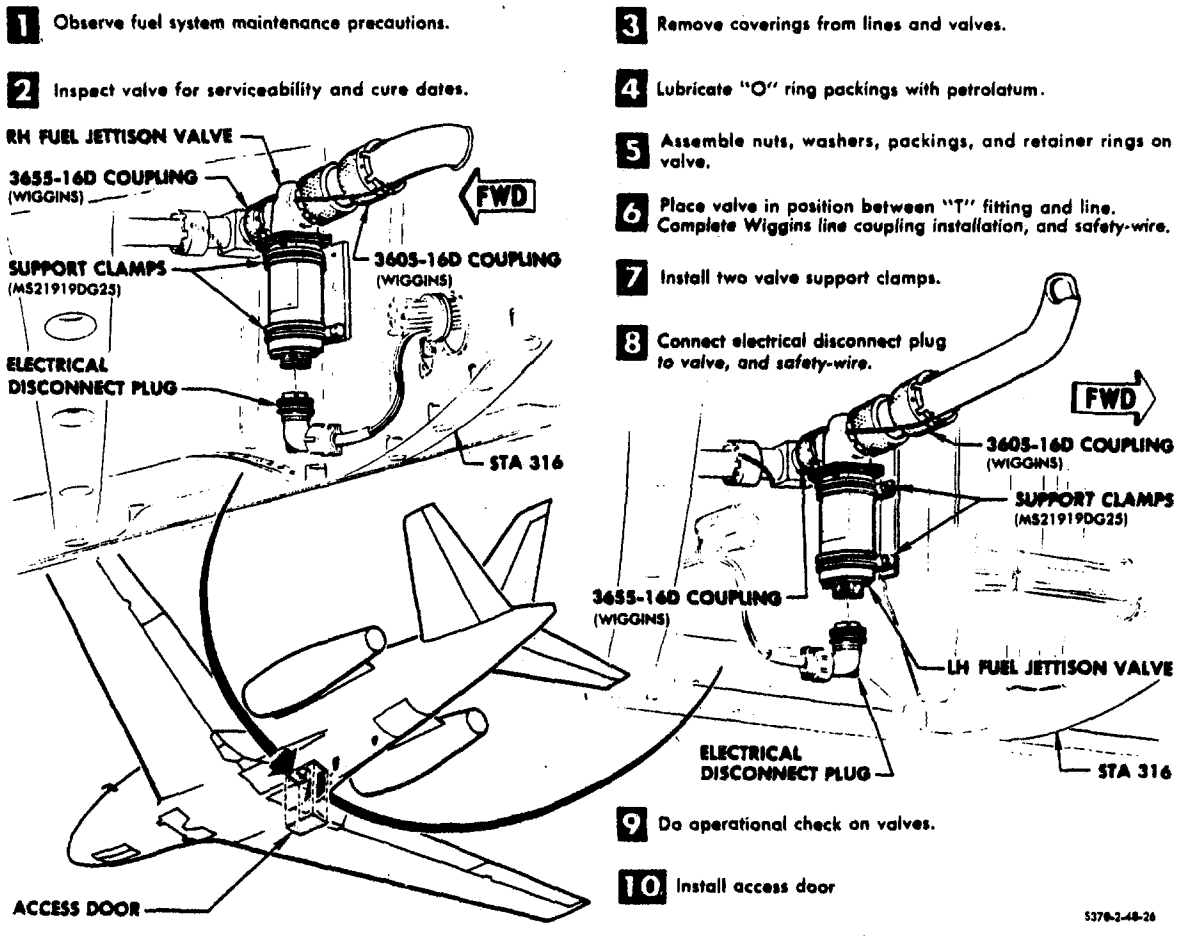
      (4) Remove access door from fuel booster pump sump bulkhead.

      (5) Remove lockwire and disconnect conduit "K" nut from float switch.

      (6) Remove two AN bolts from switch mounting bracket and lift switch from fuel bay.

      (7) Temporarily install rear spar access door.

   B. Installing Fuel Jettison Level Control Float Switches (See figure 28-30-402.)

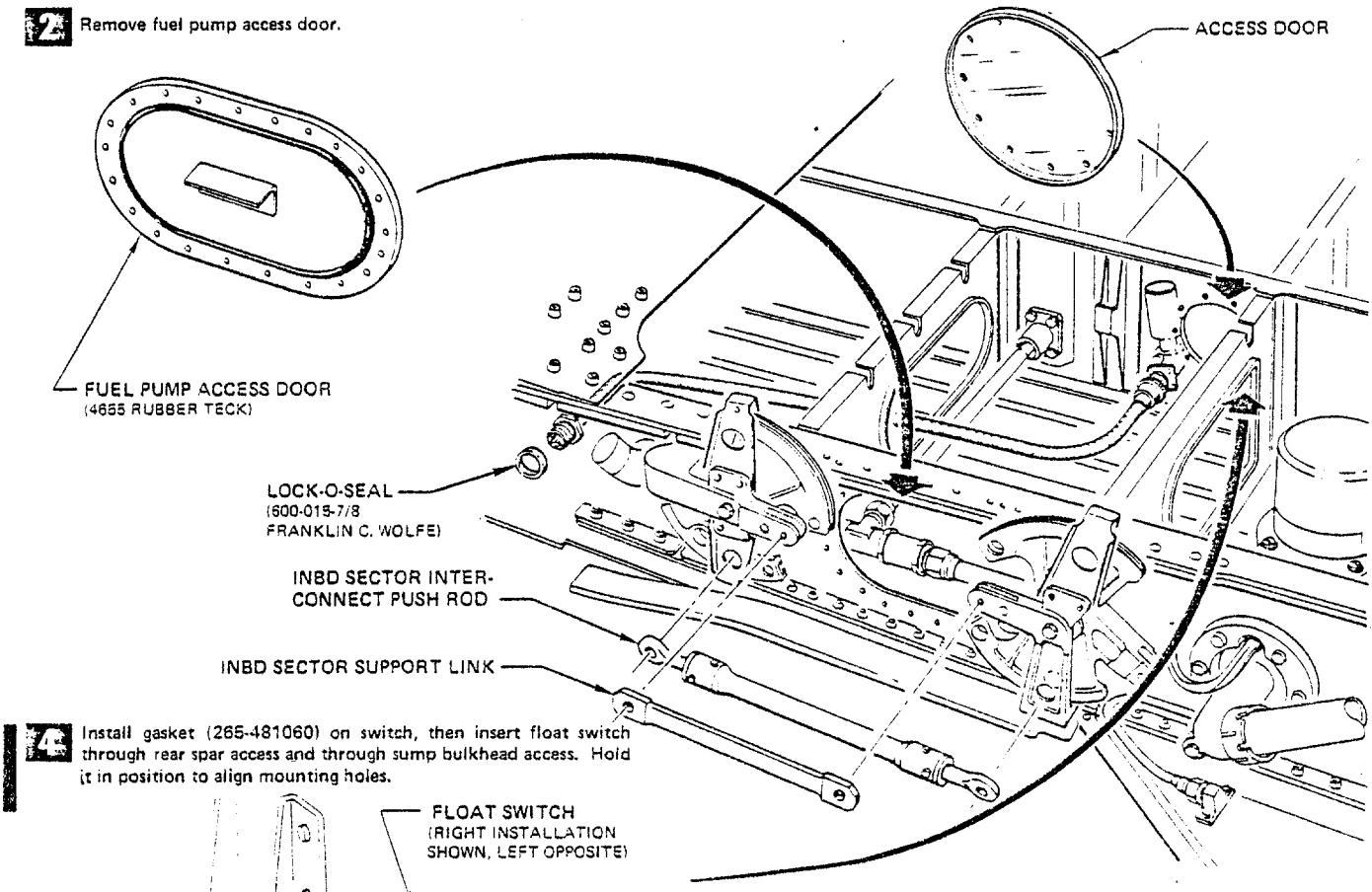

1 Observe fuel system maintenance precautions.
2 Inspect valve for serviceability and cure dates.
3 Remove coverings from lines and valves.
4 Lubricate "O" ring packings with petrolatum.
5 Assemble nuts, washers, packings, and retainer rings on valve.
6 Place valve in position between "T" fitting and line. Complete Wiggins line coupling installation, and safety-wire.
7 Install two valve support clamps.
8 Connect electrical disconnect plug to valve, and safety-wire.
9 Do operational check on valves.
10 Install access door
RH FUEL JETTISON VALVE
3655-16D COUPLING (WIGGINS)
FWD
SUPPORT CLAMPS (MS21919DG25)
3605-16D COUPLING (WIGGINS)
ELECTRICAL DISCONNECT PLUG
STA 316
ACCESS DOOR
FWD
3605-16D COUPLING (WIGGINS)
SUPPORT CLAMPS (MS21919DG25)
3655-16D COUPLING (WIGGINS)
LH FUEL JETTISON VALVE
ELECTRICAL DISCONNECT PLUG
STA 316
5378-2-48-26

Installing Left and Right Fuel Jettison Valves
Figure 28-30-401

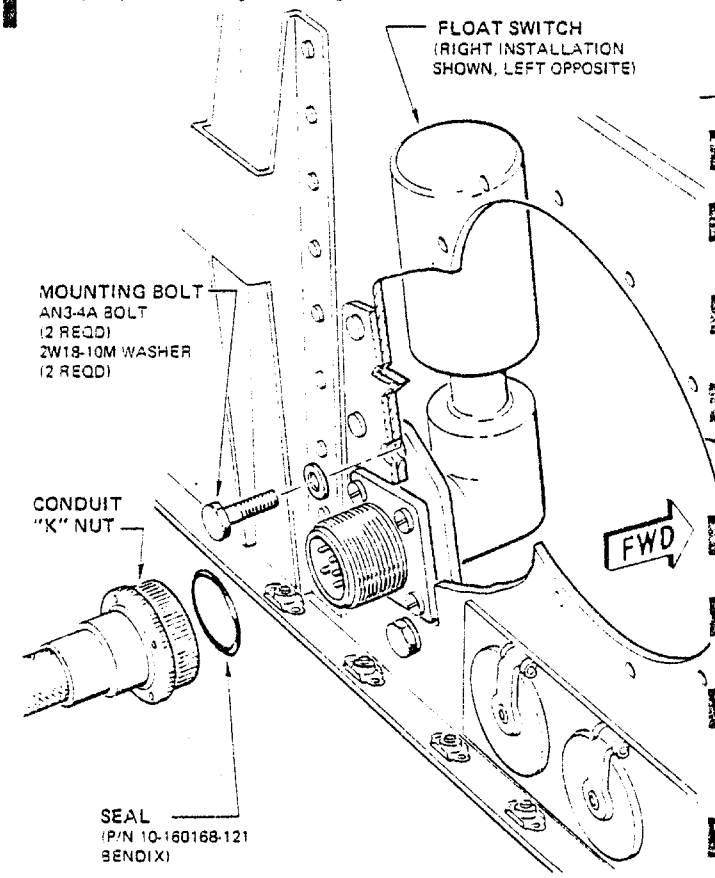
1. Observe fuel maintenance precautions.

2. Remove fuel pump access door.

3. Remove access door from booster pump sump bulkhead.



4. Install gasket (265-481060) on switch, then insert float switch through rear spar access and through sump bulkhead access. Hold it in position to align mounting holes.



5. Install two AN3 mounting bolts.

6. Install seal, and connect conduit "K" nut to switch. Install lockwire.

7. Remove any foreign matter from wing and sump.

8. Install pump sump bulkhead access door.

SUMP BULKHEAD ACCESS

9. Install fuel pump access door.

10. On right-hand installation, reinstall aileron sector support link and sector interconnecting push rod.

11. On right-hand installation, tension aileron control cables and perform aileron rigging check. (Refer to Chapter 27.)

12. Connect and safety any outside connections that may have been removed.

S370-2-48-27A

Installing Fuel Jettison Level Control Float Switches
Figure 28-30-402

FUEL SYSTEM

INSPECTION/CHECK

FUNCTIONAL TEST OF FUEL JETTISON SYSTEM

See table 601 for a functional test of the fuel jettison system.

Table 601. Functional Test of Fuel Jettison System

PROCEDURE	NORMAL INDICATION
Equipment/Tools • Set of Jacks • Jacking and Mooring Kit • 28 Vdc Electrical Power Supply • Suitable Length of Hose • Suitable Container to Recieve Jettisoned Fuel • Source of Fuel for Refueling **Consumables** • Fuel	
CAUTION ***Place airplane on jacks before fuel jettison test is performed to prevent possible damage to the airplane.***	
NOTE ***Original jettison float switches were set at 700 to 1100 lbs remaining. Approved replacement switches may be set at 1000 to 1400 lbs remaining. This could result in a fuel unbalance of up to 700 lbs in event of a switch malfunction. Use the same P/N switch as remaining serviceable switch or replace both switches with the same P/N switch if a jettison float switch must be replaced.***	
1. Jack airplane.	
2. Level airplane.	A. Fuselage.....3° nose up B. Wings ..... level laterally
3. Fuel quantity .....500 gal (3250 lbs)	Equally divided between LH and RH wing

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INSPECTION/CHECK (Cont'd)

Table 601. Functional Test of Fuel Jettison System (Cont'd)

PROCEDURE	NORMAL INDICATION
CAUTION ***Do not bend or crush jettison outlet tube. Use support for hose.***	
4. Hose.....connect to fuel jettison outlet	Extend hose to fuel container or tank truck
5. Power levers..... CUTOFF	Prevents engines from being flooded with fuel
6. Electrical power ..... connect	
7. Circuit breakers:	
LDG GEAR POS.....out	Ensures contacts of fuel jettison ground lockout relay will remain closed simulating an airborne condition if MLG wheels are not clear of ground
LH ENGINE IGN & BOOST.....out	
RH ENGINE IGN & BOOST.....out	
BOOST PUMP-RH.....in	
BOOST PUMP-LH.....out	Ensures fuel will be jettisoned from RH wing only
FUEL JETTISON-LH .....in	
FUEL JETTISON-RH .....out	A. Prevents RH jettison valve from opening B. Permits testing of LH jettison valve and contacts of one switch in the RH fuel level switch assy. (See figure 28-30-3.)
8. Switches:	
LH ENGINE-ENGINE MASTER..... ON	Ensures jettisoned fuel will flow through LH fuel shutoff valve
RH ENGINE-ENGINE MASTER.....OFF	
ELECTRICAL MASTER ..... ON	
EMERGENCY FUEL JETTISON ..... AUTO	A. Fuel ..... flows from hose B. Indicates jettisoned fuel will flow through LH jettison and boost pump crossfeed valves. C. FUEL JETT OPEN light ..... on D. Fuel ..... flow stops automatically E. Verifies proper operation of one switch in the RH fuel level switch assy. (See figure 28-30-3.)
9. Fuel quantity RH wing..... 1000 to 1400 lbs	
10. EMERGENCY FUEL JETTISON .....OFF	FUEL JETT OPEN Light .....off
11. Refuel RH wing with 50 to 100 lbs. of fuel.	RH fuel level switches..... close

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INSPECTION/CHECK (Cont'd)

Table 601. Functional Test of Fuel Jettison System (Cont'd)

PROCEDURE	NORMAL INDICATION
12. Circuit breakers:	
FUEL JETTISON-RH .....in	
FUEL JETTISON-LH .....out	A. Prevents LH jettison valve from opening. B. Permits testing of the remaining switch in the RH fuel level switch assy.
13. Switches:	
LH ENGINE-ENGINE MASTER.....OFF	
RH ENGINE-ENGINE MASTER..... ON	Ensures jettisoned fuel will flow through RH fuel shutoff valve.
EMERGENCY FUEL JETTISON ..... AUTO	A. Fuel ..... flows from hose B. Indicates jettisoned fuel will flow through RH jettison and boost pump crossfeed valves. C. FUEL JETT OPEN light ..... on D. Fuel ..... flow stops automatically E. Verifies proper operation of the remaining switch in the RH fuel level switch assy.
14. Fuel quantity RH wing..... 1000 to 1400 lbs	
15. EMERGENCY FUEL JETTISON switch ..OFF	FUEL JETT OPEN light ..... off
16. Refuel RH wing with 50 to 100 lbs of fuel.	RH fuel level switches..... close
17. Circuit breakers:	
BOOST PUMP-LH FUEL.....in	
BOOST PUMP-RH FUEL .....out	Ensures fuel will be jettisoned from LH wing only.
18. Switches:	
LH ENGINE-ENGINE MASTER.....OFF	
RH ENGINE-ENGINE MASTER..... ON	Ensures jettisoned fuel will flow through RH fuel shutoff valve
EMERGENCY FUEL JETTISON ..... AUTO	A. Fuel ..... flows from hose B. Indicates jettisoned fuel will flow through RH jettison and boost pump crossfeed valves. C. FUEL JETT OPEN light ..... on D. Fuel ..... flow stops automatically E. Verifies proper operation of one switch in the LH fuel level switch assy
19. Fuel quantity LH wing..... 1000 to 1400 lbs	
20. EMERGENCY FUEL JETTISON .....OFF	FUEL JETT OPEN light ..... off

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INSPECTION/CHECK (Cont'd)

Table 601. Functional Test of Fuel Jettison System (Cont'd)

PROCEDURE	NORMAL INDICATION
21. Refuel LH wing with 50 to 100 lbs of fuel.	LH fuel level switches ..... close
22. Circuit breakers:	
FUEL JETTISON-LH .....in	
FUEL JETTISON-RH .....out	A. Prevents RH jettison valve from opening B. Permits testing of the remaining switch in the LH fuel level switch assy
23. Switches:	
LH ENGINE-ENGINE MASTER..... ON	
RH ENGINE-ENGINE MASTER.....OFF	
EMERGENCY FUEL JETTISON ..... AUTO	A. Fuel ..... flows from hose B. Indicates jettisoned fuel will flow through LH jettison and boost pump crossfeed valves C. FUEL JETT OPEN light ..... on D. Fuel ..... flow stops automatically E. Verifies proper operation of the remaining switch in the LH fuel level switch assy
24. Fuel quantity LH wing..... 1000 to 1400 lbs	
25. EMERGENCY FUEL JETTISON .....OFF	FUEL JETT OPEN light ..... off
26. Refuel LH wing with 50 to 100 lbs of fuel.	LH fuel level switches ..... close
27. BOOST PUMP-RH FUEL circuit breaker .....in	
28. Switches:	
RH ENGINE-ENGINE MASTER..... ON	
EMERGENCY FUEL JETTISON ..... MAN	Fuel ..... flows from hose
EMERGENCY FUEL JETTISON .....OFF	Fuel ..... stops flowing
29. Circuit breakers:	
FUEL JETTISON-RH .....in	
FUEL JETTISON-LH .....out	
30. Switches:	
EMERGENCY FUEL JETTISON ..... MAN	Fuel ..... flows from hose
EMERGENCY FUEL JETTISON .....OFF	Fuel ..... stops flowing
LH ENGINE-ENGINE MASTER.....OFF	
RH ENGINE-ENGINE MASTER.....OFF	
ELECTRICAL MASTER .....OFF	
31. Electrical power .....disconnect	

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INSPECTION/CHECK (Cont'd)

Table 601. Functional Test of Fuel Jettison System (Cont'd)

PROCEDURE	NORMAL INDICATION
32. Circuit breakers:	
FUEL JETTISON-LH	in
LH ENGINE-IGN & BOOST	in
RH ENGINE-IGN & BOOST	in
LDG GEAR POS	in
33. Disconnect hose from fuel jettison outlet.	
34. Lower airplane and remove jacks.	
35. Secure airplane as desired.	

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FUEL INDICATING SYSTEM - DESCRIPTION AND OPERATION

1. Fuel Indicating System

The fuel indicating system consists of the fuel quantity indicating system, the fuel low-pressure warning system, and the fuel flow indicating system and their components.

A. Fuel Quantity Indicating System

The airplane has two independent fuel quantity indicating systems, one for each wing, with half of the fuselage tank fuel included on each indicator. (See figure 28-40-1.) Each system consists of an indicator with a self-contained transistorized amplifier, eight tank units, reference condensers, connectors, coaxial cables, wiring, and a relay control unit. One of the eight tank units has dual elements, one set of the capacitor elements being used in each system. The reference condenser capacitor elements are in one of the tank units of each system. The single-prong, bayonet-type coaxial connectors are designed to meet the high-impedance requirements of the system.

A relay control unit is installed in the system to switch the tank units and dummy condensers to give an indication of the amount of fuel in the auxiliary tank. The relay control unit is operated by the fuel quantity test switch installed on the instrument panel. During normal system operation, the relay control unit is de-energized, to connect the left wing elements to the left fuel quantity indicator and the right wing elements to the right fuel quantity indicator, when the fuel quantity test switch is at the NORMAL or TOTAL position. Half of the auxiliary tank fuel is also shown on the respective left and right fuel quantity indicators.

The fuel quantity test switch TEST position tests the left and right fuel quantity indicators. The fuel quantity test switch RH IND, RH WING, LH IND, AUX position selection energizes the relay control unit to show the amount of fuel in the auxiliary tank on the left fuel quantity indicator and the amount of fuel in the right wing tank on the right fuel quantity indicator. At this time, the left wing tank units are disconnected from the left indicator, and the right auxiliary tank unit elements are disconnected from the right indicator and reconnected to the left indicator. In addition, a dummy condenser (inside the relay control unit), equal to the dry capacitance of the left wing tank units minus the dry capacitance of the right auxiliary tank unit elements, is connected to the left indicator, and a dummy condenser (inside the relay control unit), equal to the dry capacitance of the auxiliary tank unit elements, is connected to the right indicator. Capacitor adjustments are provided on the relay control unit. (See figure 28-40-2.)

The fuel caution system consists of a panel light and an indicator low-level switch. Power for the fuel quantity indicating system is obtained from the 115-volt ac essential source. Power for the fuel-level warning system is obtained from the 28-volt dc essential source.

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The compensated capacitance-type fuel quantity indicating system operates as a self-balancing-type network. The tank units have tubular-type capacitor elements in the fuel tanks, forming a leg of the measuring network. The capacitance of the tank units varies with the fuel level and the fuel dielectric constant. The reference condenser compensates for variations of the fuel dielectric constant, making the system responsive to the mass of fuel. The reference condenser is completely immersed in the fuel being measured; therefore, its function is to improve system accuracy by compensating the indicator response to the general relationship between the dielectric constant and the density of the fuel.

A variation in the fuel level introduces an unbalanced signal which is amplified to a suitable level to drive a two-phase induction motor in the indicator. The miniature two-phase motor drives a rebalance potentiometer to balance the system through a high-reduction gear train.

The tank units are the basic sensing elements of the indicating system. The capacitor elements are characterized to match the contour of the tank, resulting in a linear response of the indicator. Because the sensing elements are fixed, the capacitance of the tank unit can be varied only by the change in the dielectric. The dielectric ratio between air and fuel is approximately 2:1; therefore, the capacitance of the sensing element is about twice as great when the tank is full as it is when the tank is empty. Any change of fuel quantity between empty and full reflects a change in tank unit capacitance.

The fuel-level caution light is energized at a fuel low-level condition by a switch in the indicator.

B. Fuel Low-pressure Warning System

Four fuel pressure switches, two in the right fuel system and two in the left fuel system, sense low fuel pressure conditions. The following applies to the right fuel system: One pressure switch is positioned downstream from the right fuel shutoff valve and monitors the right boost pump pressure. The second pressure switch (a pressure-differential type switch) is positioned in the right engine pod and monitors the differential pressure of the inlet and outlet lines of the centrifugal stage of the engine-driven fuel pump. The two fuel pressure switches in the left fuel system function the same.

When either the right or left fuel boost pump fails or if the right or left centrifugal stage of the engine-driven fuel pumps fails, the respective "RH FUEL PRESS LOW" or "LH FUEL PRESS LOW" caution light on the caution light indicator panel is lighted. The "FUEL" switch should immediately be actuated to -X-FEED. The boost cross-feed and tank cross-feed valves will open, allowing either the right or left boost pump to supply fuel to both engines. If the respective "RH FUEL PRESS LOW" or "LH FUEL PRESS LOW" caution light goes out, the designated right or left fuel boost pump has failed. If the light remains on, the designated right or left centrifugal stage of the engine-driven fuel pump has failed. On all airplanes, if the tank cross-feed or boost cross-feed valves fail to open, the "FUEL TANK X-FD FAIL" or "FUEL PUMP X-FD FAIL" caution lights will light and remain on.

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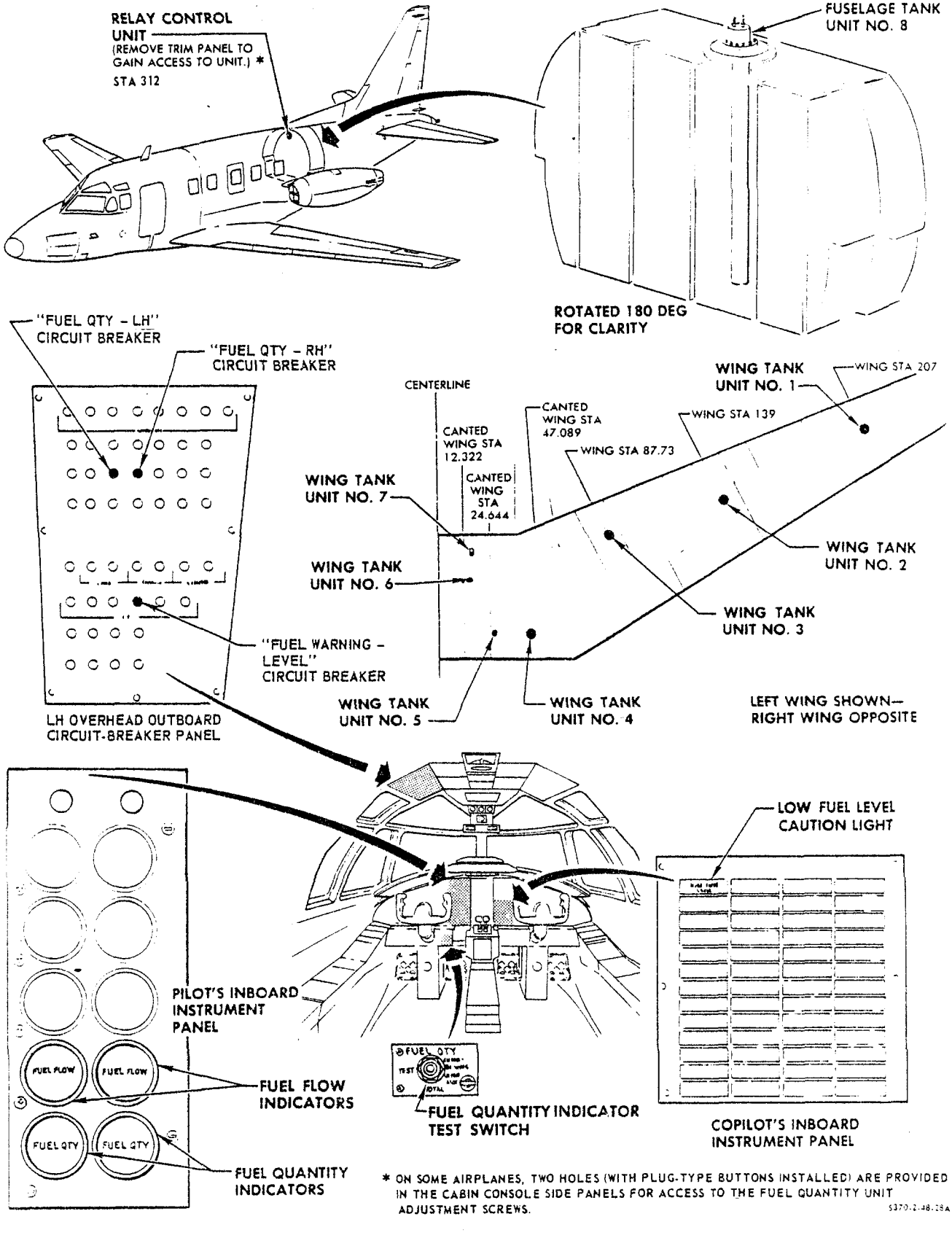
RELAY CONTROL UNIT
(REMOVE TRIM PANEL TO GAIN ACCESS TO UNIT.) *
STA 312
FUSELAGE TANK UNIT NO. 8
ROTATED 180 DEG FOR CLARITY
"FUEL QTY - LH" CIRCUIT BREAKER
"FUEL QTY - RH" CIRCUIT BREAKER
CENTERLINE
CANTED WING STA 12.322
CANTED WING STA 24.644
CANTED WING STA 47.089
WING STA 87.73
WING STA 139
WING STA 207
WING TANK UNIT NO. 1
WING TANK UNIT NO. 2
WING TANK UNIT NO. 3
WING TANK UNIT NO. 4
WING TANK UNIT NO. 5
WING TANK UNIT NO. 6
WING TANK UNIT NO. 7
"FUEL WARNING - LEVEL" CIRCUIT BREAKER
LH OVERHEAD OUTBOARD CIRCUIT-BREAKER PANEL
LEFT WING SHOWN—RIGHT WING OPPOSITE
LOW FUEL LEVEL CAUTION LIGHT
PILOT'S INBOARD INSTRUMENT PANEL
FUEL FLOW
FUEL FLOW
FUEL QTY
FUEL QTY
FUEL FLOW INDICATORS
FUEL QUANTITY INDICATORS
FUEL QUANTITY INDICATOR TEST SWITCH
COPILOT'S INBOARD INSTRUMENT PANEL

* ON SOME AIRPLANES, TWO HOLES (WITH PLUG-TYPE BUTTONS INSTALLED) ARE PROVIDED IN THE CABIN CONSOLE SIDE PANELS FOR ACCESS TO THE FUEL QUANTITY UNIT ADJUSTMENT SCREWS.

5370-2-48-28A

C. Fuel Flow Indicating System

Two fuel flow indicators are mounted on the pilot's inboard instrument panel. The indicators show the rate of fuel flow from each engine fuel control unit in pounds per hour. The indicators are calibrated from 0 to 4000 in increments of 100 pounds. The indicators are powered by 115-volt, 400-cycle ac power from the airplane essential bus system. When the engines are being operated, fuel passing through the respective engine-mounted fuel flow transmitter results in a signal to the indicators representing mass flow rate.

D. Total Fuel Consumed Indicating System

On NA-380 airplanes equipped as Airway Checker Airplanes, a total fuel consumed system has been installed (see figure 28-40-3). The total fuel consumed system is powered by the 115 VAC primary bus and is integrated with the fuel flow indicating system. Components of the total fuel consumed system consist of a digital display instrument mounted in the co-pilot's outboard instrument panel, a fuel total reset button on the co-pilot's console and associated wiring. The indicator provides a digital indication of the amount of fuel used, in pounds. Depressing the FUEL TOTAL RESET button on the copilot's console will reset the digital display unit to zero. The totalizer signal is boot strap signal from the fuel flow indicators; therefore, in the event of a fuel flow indicator failure, the totalizer will display one-half of the normal indication.

2. Fuel Cell Tank Units

The tank unit is a tubular capacitor in which the area and spacing between electrodes are fixed, and are made of three concentric tubes. The inner tube (inner electrode) serves as a plate of a capacitor. The middle tube (the other electrode) serves as the other plate. The third tube, around the other two tubes, is an electrical shield as well as a support. The two inner electrodes are electrically insulated from each other and from the outside tube, which is at ground potential. Fuel is permitted to flow freely in and out of spaces between the tubes, through drain and vent holes at each end of the unit. This fills the unit to the same fuel level as that in the fuel cell. Access to the inside of the fuselage cell is provided by removing a door from the metal compartment at fuselage station 357 and then removing the fuel cell access door. There are seven fuel tank unit distributed throughout each wing. They can be reached through wing access doors.

3. Fuel Quantity Indicators

Two fuel quantity indicators, on the pilot's inboard instrument panel, indicate in pounds of fuel. The indicators are calibrated from 0 to 4000 in increments of 100 pounds. They indicate all internal fuel. Each indicator reads one half the total wing fuel plus one half the total fuselage tank fuel regardless of the temperature or the type of fuel. The fuel quantity indicating system is powered by the 115-volt ac essential bus.

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A fuel quantity test switch is in the landing gear control switch panel assembly. The function of this switch is to test the fuel quantity indicating system for proper operation. The three switch positions are TEST; TOTAL; and RH IND, RH WING, LH IND, AUX. When the switch is placed at TEST, the pointer of the left and right fuel quantity indicators should rotate counterclockwise. When the switch is placed at TOTAL, the pointers should return to their former positions. The TOTAL position indicates on the left fuel quantity indicator the total of the fuel in the left wing tank plus half of the fuel in the auxiliary tank, and the right fuel quantity indicator shows the total of the fuel in the right wing tank plus half of the fuel in the auxiliary tank. The RH IND, RH WING, LH IND, AUX switch position indicates the quantity of fuel in the auxiliary tank on the left indicator, while the right indicator shows the quantity of fuel in the right wing tank. Failure of the pointers to respond shows a faulty indicating system.

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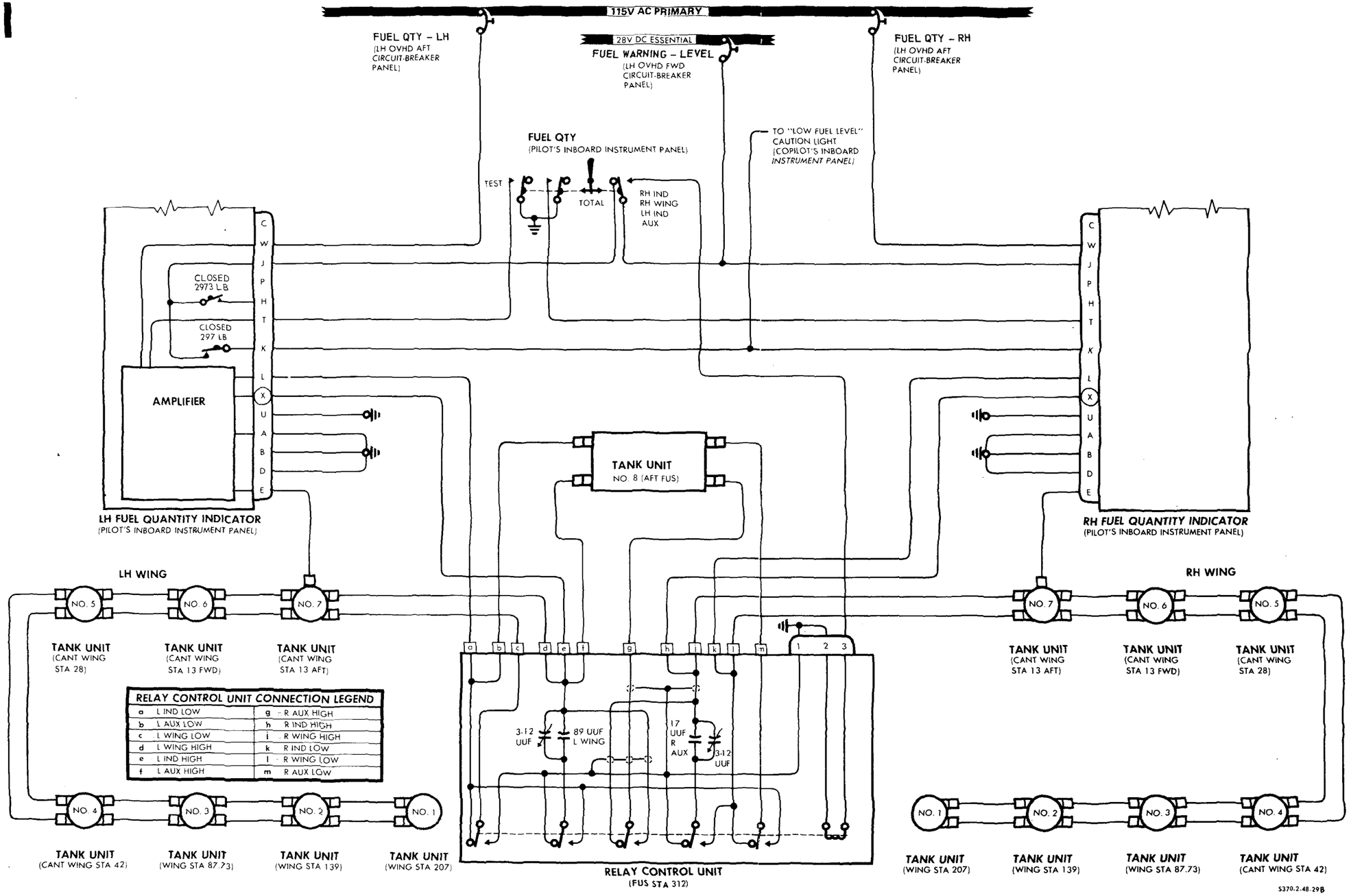


Fuel Quantity Indicating System Electrical Schematic
Figure 28-40-2

4. Fuel Pressure-sensing Switches

Four fuel pressure switches, two in the right fuel system and two in the left fuel system, indicate the low fuel pressure conditions. The following applies to the right pressure switches: One pressure switch is positioned downstream from the right fuel shutoff valve and illuminates the "RH FUEL PRESS LOW" caution light when the boost pump pressure drops to 5 (±1) psi. The second pressure switch in the right fuel system is positioned in the right engine pod and monitors the inlet pressure and outlet pressure of the centrifugal stage of the engine-driven fuel pump. This is a pressure-differential type of switch. The pressure rise of about 40 psi across the centrifugal portion of the engine-driven pump creates a pressure differential of 40 psi. When the pressure differential across the pump drops to 6 (±1) psi, the pressure switch will close and light the "RH FUEL PRESS LOW" caution light. The light will go out on a pressure rise of 3 psi above 6 (±1) psi. The two fuel pressure switches in the left fuel system, one switch downstream from the left fuel shutoff valve and one in the left engine pod, operate in the same manner.

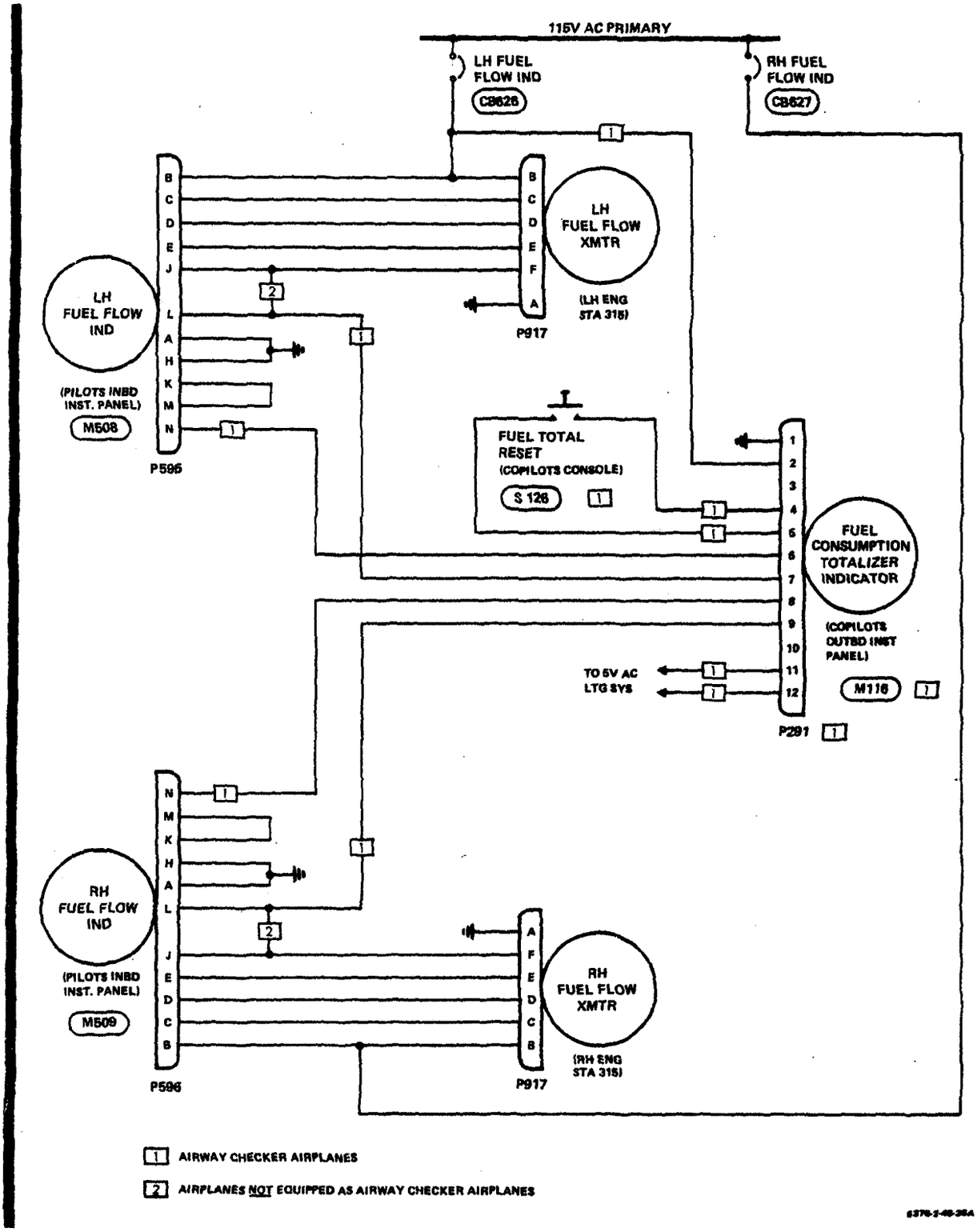
5. Fuel Flow Indicators

The fuel flow indicators consist of a selsyn receiver for rotating the pointer and a stable frequency power supply for the transmitter impeller motor, to compensate for external frequency changes in the airplane electrical system. The indicators are integrally lighted, hermetically sealed, and are designed to indicate the flow rate of engine fuel in pounds per hours. Fuel flow indicators have a range of 0 to 4000 pounds per hour. (See figure 28-40-3.)

6. Fuel Flow Transmitters

Elements of the transmitters include flow-sensing means, a synchronous drive motor, a restraining spring, and a transducer that converts mechanical rotation to an electrical position signal.

Angular momentum proportional to impeller speed is imparted to each unit mass of fuel. By recovering this angular momentum, a mechanical torque is developed which is proportional to the product of mass flow rate and impeller speed. This torque is converted to an electrical signal which is transmitted to the fuel flow indicator. The fuel flow transmitters are installed on the engines and may be reached through engine access doors.



Right and Left Engine Fuel Flow Indicating Systems (Typical)
Figure 28-40-3

FUEL INDICATING SYSTEM - TROUBLE SHOOTING

1. Trouble Shooting Fuel Indicating System

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
	FUEL QUANTITY INDICATING SYSTEM	
A. NO INDICATOR RESPONSE OR READING, OR INDICATOR DOES NOT RESPOND WHEN FUEL QUANTITY TEST SWITCH IS ACTUATED.		
(1) No ac power to power unit.	No indication when fuel quantity test switch is actuated.	Repair ac power circuit.
(2) Faulty power unit (amplifier).	No indication when fuel quantity test switch is actuated.	Replace indicator.
(3) Faulty indicator.	No indication when fuel quantity test switch is actuated.	Replace indicator.
(4) Faulty wiring in fuel quantity test circuit.	Make continuity check.	Repair as needed.
B. FUEL QUANTITY INDICATOR READS HIGH WITH EMPTY FUEL CELLS, OR POINTER ROTATES CLOCKWISE WITHOUT STOPPING.		
(1) Faulty power unit (amplifier in indicator).	Substitute known good indicator. If pointer stops rotating, old indicator is bad.	Replace indicator.
(2) Faulty reference condenser or associated wiring.	Make continuity check of wires.	Repair or replace wire or connector as required.
	Make continuity check to ground.	Replace reference condenser.
C. FUEL QUANTITY INDICATOR POINTER ROTATES COUNTERCLOCKWISE SLOWLY WITHOUT STOPPING WITH FUEL CELLS EMPTY.		
(1) Faulty power unit (amplifier in indicator).	Substitute known good indicator. If pointer stops rotating, old indicator is bad.	Replace indicator.
(2) Faulty tank unit(s) or associated wiring.	Isolate faulty wire or connection. Perform a continuity check across the two pairs of contacts. Use variable-capacitance tester to determine whether tank has	Replace or repair wires or connectors as required. Replace tank unit.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
	the proper capacitance value. If tank unit does not show correct value, unit is faulty.	
D. INDICATOR READS LOW IN RESPECT TO FUEL IN CELLS.		
(1) One or more tank units not working in circuit.	Refer to isolation procedure for probable cause (2) under trouble C.	Repair or replace wires, connectors, or tank units as required.
E. FUEL QUANTITY INDICATOR POINTER STUCK.		
(1) Pointer drags on dial face or glass cover.	No fuel indication when system and all components, except indicator, are functioning properly.	Replace indicator.
F. FUEL QUANTITY INDICATOR POINTER OSCILLATES, CREEPS SLOWLY ACROSS DIAL SCORE, OR IS STOPPED AT ZERO OR FULL END OF SCALE.		
(1) Wires short-circuited or broken.	Isolate faulty wire connection or break by continuity check.	Repair or replace wires or connections as required.
G. LOW READING OR NO READING AT FUEL QUANTITY INDICATOR.		
(1) Poor connection at tank unit or indicator.	Isolate faulty connection by making continuity check.	Tighten, repair, or replace as required.
(2) Soap solution used in pressure check of fuel supply system in cable connections.	Isolate faulty connection by making continuity check.	Wash connections with water and dry with solvent.
(3) Damaged pivots or jewels in indicator.		Replace fuel quantity indicator.
FUEL LOW-PRESSURE WARNING SYSTEM		
H. LH OR RH FUEL PRESS LOW LIGHT REMAINS ON.		
(1) Boost pump does not supply sufficient fuel pressure.	Make pressure check. (Refer to procedure for boost pump pressure check.)	Replace boost pump.
(2) Centrifugal stage of the engine-driven fuel pump does not increase boost pump		Replace centrifugal stage of engine-driven fuel pump.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
pressure more than 12 (±1) psi.		
(3) Defective boost pump pressure switch.	Check for fuel leakage from pressure switch.	Replace pressure switch.
(4) Defective engine-driven fuel pump pressure differential switch.	Check for fuel leakage from pressure switch.	Replace pressure switch.
	Check for kinked or damaged inlet or outlet lines to switch.	Repair or replace lines.
I. "LH FUEL PRESS LOW" OR "RH FUEL PRESS LOW" CAUTION LIGHT DOES NOT COME ON WHEN POWER IS ON SYSTEM AND BOOST PUMP IS INOPERATIVE OR ENGINE IS NOT RUNNING.		
(1) "FUEL WARNING - BOOST" circuit breaker defective.	Place all airplane switches at OFF.	Replace circuit breaker.
	Remove all external power from airplane. Remove left outboard forward dc essential circuit-breaker panel and engage "FUEL WARNING - BOOST" circuit breaker; then check continuity across circuit-breaker terminals.	
FUEL FLOW INDICATING SYSTEM		
J. INDICATOR POINTER SLUGGISH OR IN ERROR.		
(1) Low ac voltage.	Check for correct power supply.	
(2) Faulty indicator.	Interchange right and left engine indicators.	Replace faulty indicator.
(3) Faulty wire harness.	Check continuity of wire harness.	Repair as necessary.
K. INDICATOR POINTER SWINGS BACK AND FORTH ACROSS DIAL.		
(1) Faulty wiring, indicator, or transmitter.	Check continuity of wiring, or change indicator and/or transmitter. (See figure 28-40-3.)	Repair wiring or replace indicator and/or transmitter.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
(2) Fuel leaks at transmitter fittings.	Check for loose end fittings and cut O-ring seals on transmitter.	Replace O-ring seals and tighten fittings.
L. SLIGHT OR NO MOVEMENT OR INDICATOR POINTER SPINS.		
(1) Shorted or intermittent circuit, or faulty indicator.	Check continuity of wiring, or interchange right and left engine indicators.	Repair wiring or replace indicator.
M. INTERMITTENT OR NO INDICATION OF FUEL FLOW.		
(1) Broken impeller pivots in transmitter.	Check for indication during engine operation.	Replace fuel flow transmitter.

FUEL INDICATING SYSTEM - MAINTENANCE PRACTICES

WARNING: TO PREVENT POSSIBLE EXPLOSIONS DURING THE FOLLOWING REPLACEMENT PROCEDURES, MAKE SURE THAT ALL AIRPLANE SWITCHES ARE AT OFF OR NORMALLY MAINTAINED SAFE POSITIONS, AND THAT ALL SOURCES OF ELECTRICAL POWER ARE REMOVED FROM AIRPLANE BUS SYSTEM.

NOTE: Subsequent to replacement of a fuel quantity indicator or probe, the fuel quantity system should be calibrated.

1. Replacement of Wing Tank Units

A. Removing Wing Tank Units

The right and left wing intermediate and outboard wing tank units may be reached from the top of the right and left wings through access covers. (Refer to Chapter 6, figure 6-00-3.) The wing center section fuel cell, containing the right and left wing inboard tank units, is equipped with access doors for maintenance of the fuel system units. Access to wing tank units #6 and #7 is provided by removing door assembly, P/N 4655, from inboard rear spar. Access to the #5 wing tank unit is provided by removing P/N 4980 door from inboard front spar.

(1) After locating fuel quantity indicating tank unit or units to be removed, defuel airplane fuel system.

CAUTION: COMPLY WITH ALL NECESSARY SAFETY PROCEDURES WHEN DEFUELING AIRPLANE FUEL SYSTEM.

(2) After gaining access to selected wing tank unit, either remove wing tank unit securing screws or loosen and remove securing clamps. Then disconnect coaxial leads from unit and remove from fuel cell.

B. Installing Wing Tank Units

(1) Connect coaxial leads to wing tank unit and install in proper mounting position, using securing clamps and/or screws. Ensure that "O" ring seals are installed under head of screws.

(2) Make sure that access doors and/or plates are properly installed to prevent fuel system leaks.

(3) Adjust fuel quantity indicating system. (Refer to 28-40-3.)

2. Replacement of Fuelage Tank Fuel Probe Unit

A. Removing Fuel Probe Unit

The fuel quantity indicating fuselage tank unit can be reached through access doors.

(1) After gaining access to fuselage tank unit, remove 18 bolts and washers retaining fuel probe support to tank compartment and remove support and two gaskets.

(2) Remove the two countersunk screws retaining fuel probe to cell and remove fuel probe and cell opening "O" ring.

(3) Disconnect coaxial leads from unit and remove unit from fuel cell.

B. Installing Fuel Probe Unit

(1) Install MS29513-228 "O" ring in groove of cell fuel probe opening mount.

(2) Install fuel probe, align mounting holes, and secure with two countersunk screws.

(3) Place 265-484099 gasket on fuel probe flange and 265-484100 gasket on tank compartment mounting flange.

(4) Align holes in gaskets with mounting flange holes.

(5) Position support ring over gaskets and align holes in ring with holes in gaskets and mounting flanges.

(6) Install two bolts with washers into inner two holes of support ring. Do not tighten.

(7) Install 16 bolts with washers into outer 16 holes of support ring, and tighten.

(8) Tighten two inner bolts.

(9) Hook up fuel probe coaxial leads.

(10) Adjust fuel quantity indicating system. (Refer to paragraph 1 or 3 in 28-40-3.)

3. Replacement of Fuel Pressure-sensing Switches

A. Removing Boost Pump Fuel Pressure Switches

The boost pump fuel pressure switches are mounted on brackets directly beneath an access panel in the cabin floor. The procedure for removal of either switch is identical.

(1) Remove access panel in cabin floor.

(2) Make sure right and left shutoff valves are in closed position.

(3) Disconnect electrical connector to switch. (See figure 28-20-404.) Cap connector and plug.

(4) Remove line between switch and fuel boost pump line. Immediately cap fitting on boost pump line to prevent fuel spillage. Cap fitting on switch and plug "B" nuts on line.

(5) Remove jam nut holding switch on mounting bracket; then remove switch.

(6) Temporarily replace cabin floor access panel.

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B. Installing Boost Pump Fuel Pressure Switches

(1) Remove cabin floor access panel.

(2) Position switch through hole in mounting bracket and install jam nut. (See figure 28-20-404.)

(3) Remove plugs from "B" nuts on line to be installed between switch and boost pump lines. Remove cap from pressure switch. Install line on switch. When installing line on fitting in boost pump line, complete installation as rapidly as possible to prevent fuel spillage. Torque "B" nuts 40 to 65 inch-pounds.

(4) Remove caps from electrical connector and plug. Install connector to plug, and safety.

(5) Install access panel.

4. Replacement of Fuel Flow Indicators

A. Removing Fuel Flow Indicators

(1) Loosen round-head jackscrew.

(2) Withdraw instruments from panel, to expose connections on rear of instrument.

(3) Disconnect power source from instrument. Cover disconnect fittings.

CAUTION: ALL EXPOSED ELECTRICAL RECEPTACLES, PLUGS, AND PITOT-STATIC FITTINGS MUST BE COVERED TO KEEP OUT FOREIGN MATTER, PARTICULARLY METAL PARTICLES, THAT MAY CAUSE THE INSTRUMENTS TO BECOME INOPERATIVE.

(4) Remove instrument from panel.

(5) Remove clamp from panel by completely removing jackscrew and flush-head attaching screw.

B. Installing Fuel Flow Indicators

(1) Clean mounting clamp and instrument for bonding.

(2) Remove jackscrew and mounting screw from clamp.

(3) Position clamp on forward side of instrument panel with small lug at countersunk hole in panel.

(4) Install flush-head screw, but do not tighten it. Install jackscrew and tighten until clamp begins to close.

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(5) Connect instrument power source, and insert instrument in panel.

(6) Hold instrument in place and tighten jackscrew and mounting screw.

5. Replacement of Fuel Flow Transmitters-NA-370-2 Through -6, -8 and -9 Airplanes

A. Removing Fuel Flow Transmitters

The fuel flow transmitters are on the left side of the engine compressor inlet case. The right-hand engine must be removed before removal or installation of the right fuel flow transmitter can be accomplished.

CAUTION: BEFORE REMOVING ELECTRICAL PLUG FROM FUEL FLOW TRANSMITTER, BE SURE TO DISENGAGE "FUEL FLOW" CIRCUIT BREAKER. THIS IS TO ENSURE THAT AC ELECTRICAL POWER IS NOT IN THE CIRCUIT. IF AC ELECTRICAL POWER IS IN THE CIRCUIT WHEN THE FUEL FLOW TRANSMITTER ELECTRICAL PLUG IS REMOVED, THE FUEL FLOW INDICATOR AMPLIFIER WILL BE BURNED OUT.

(1) Disconnect electrical plug from transmitter.

(2) Disconnect fuel line fittings from transmitter.

(3) Remove transmitter retaining bolts and lift transmitter from engine.

(4) If new transmitter is to be installed, remove and retain end plates from both ends of transmitter.

B. Installing Fuel Flow Transmitters

The fuel flow transmitters are installed in an inverted position and metal drip shields are installed at the fuel inlet end of the transmitters. This installation prevents electrical interaction between the engine starter ground lead and transmitter magnetic field during engine starting.

(1) Using rear mounting boss retaining bolts, install transmitter on mounting bracket.

(1A) Rotate fuel flow transmitter so that electrical connection is positioned down and forward (up and forward on airplanes changed by Service Letter No. 52).

(2) Connect fuel line to transmitter, and torque line nuts to 750 inch-pounds.

(3) Connect electrical plug to transmitter, and lockwire.

6. Replacement of Fuel Flow Transmitters - NA-370-1 and -7, NA-380-1 and Later Airplanes

A. Removing Fuel Flow Transmitters

The fuel flow transmitter is located on the engine lower centerline forward of the fuel filter. Access to the fuel flow transmitter is through the engine lower inboard and lower outboard access doors.

(1) Open lower inboard and lower outboard engine access doors.

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(2) Make certain that all electrical ground power is removed from the airplane and that the electrical power master switch is at OFF.

(3) Disconnect electrical plug (P591) from fuel flow transmitter.

(4) Disconnect fuel line fittings from each end of transmitter and remove transmitter from engine.

B. Installing Fuel Flow Transmitters

(1) Position fuel flow transmitter between fuel lines and engage line nuts to fittings on each end of transmitter. Do not tighten nuts.

(2) Rotate fuel flow transmitter so that electrical connection is positioned aft and up. Connect electrical plug (P591) to fuel flow transmitter and safety with lockwire.

(3) Torque line nuts to 750 inch-pounds.

(4) Close engine access doors.

7. Replacement of Fuel Quantity Indicators

A. Removing Fuel Quantity Indicators.

The fuel quantity indicators are located on the pilot's inboard instrument panel and are accessible from the cockpit.

(1) Make certain electrical power is removed from airplane.

(2) Loosen round-head jackscrew.

(3) Withdraw fuel quantity indicator from instrument panel to expose power connection on rear of indicator.

(4) Disconnect electrical plug from rear of indicator and install protective caps on disconnected electrical plugs.

CAUTION: ALL EXPOSED ELECTRICAL RECEPTACLES, AND PLUGS MUST BE COVERED TO KEEP OUT FOREIGN MATERIALS, PARTICULARLY METAL PARTICLES THAT MAY CAUSE THE INSTRUMENT TO BECOME INOPERATIVE.

(5) Remove fuel quantity indicator from panel.

(6) Remove instrument clamp from panel by completely removing jackscrew and flush head attaching screw.

B. Installing Fuel Quantity Indicators.

(1) Clean mounting clamp and fuel quantity indicator for bonding.

(2) Remove jackscrew and mounting screw from clamp.

(3) Position clamp on forward side of instrument with small lug at countersunk hole in panel.

(4) Install flush head screw, but do not tighten. Install jackscrew and tighten until clamp begins to close.

(5) Connect electrical power plug to aft side of fuel quantity indicator.

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(6) Insert fuel quantity indicator in instrument panel. Hold instrument in place and tighten jackscrew and mounting screw.

CAUTION: ANYTIME A FUEL QUANTITY INDICATOR IS REPLACED THE FUEL QUANTITY INDICATING SYSTEM MUST BE CALIBRATED. (REFER TO 28-40-3.)

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1. Adjusting Fuel Quantity Indicating System, Using TTU-68/E or MD-1 Tester

The following procedure should be performed on each indicating system. The tank units must be dry when making the adjustments; therefore, make sure tanks are drained. Make empty and full adjustments (accessible on back of indicator) by sliding indicator out of panel or removing panel from frame, taking care not to damage coaxial and wire leads. Disconnect connector from indicator not being adjusted.

NOTE: See figure 28-40-2 for different fuel quantity test switch selections.

A. Empty Adjustment

(1) Defuel airplane.

(2) Connect 28-volt dc external power to airplane, and push in the following circuit breakers:

- "INVERTERS - CONTROL - NO. 2 & INST"
- "INVERTERS - PWR TRIP - NO. 2"
- "FUEL QTY - LH"
- "FUEL QTY - RH"
- "FUEL WARNING - LEVEL"

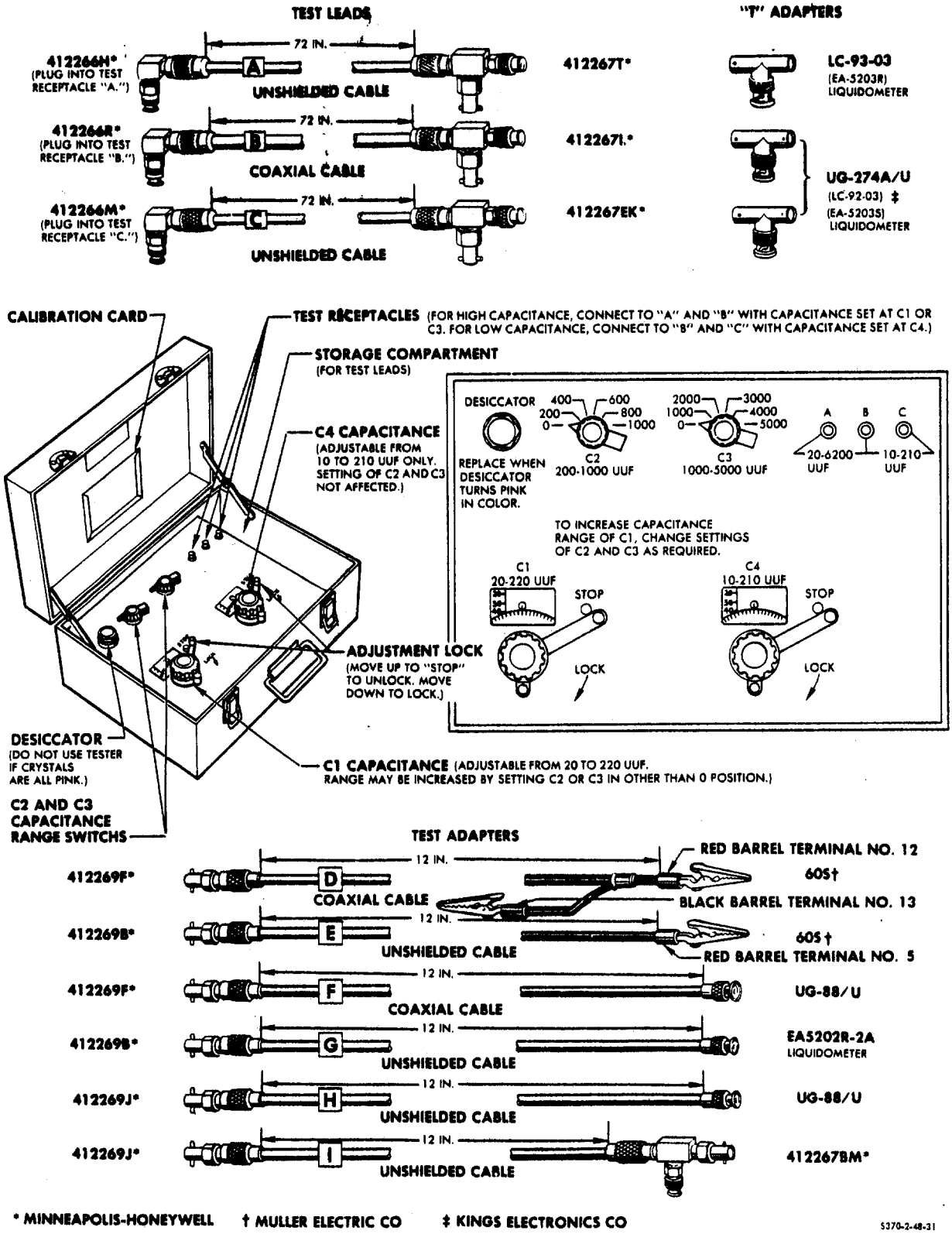
Place "INVERTER" switch at NO. 2. Allow 2 minutes for fuel quantity indicating system to stabilize before making adjustments with fuel quantity test switch at center TOTAL position.

(3) Turn "E" adjustment (on back of fuel quantity indicator) until indicator reads zero. Momentarily actuate fuel quantity test switch to move pointer off set position and check repeatability when pointer returns to zero. Look directly into indicator to avoid error due to parallax.

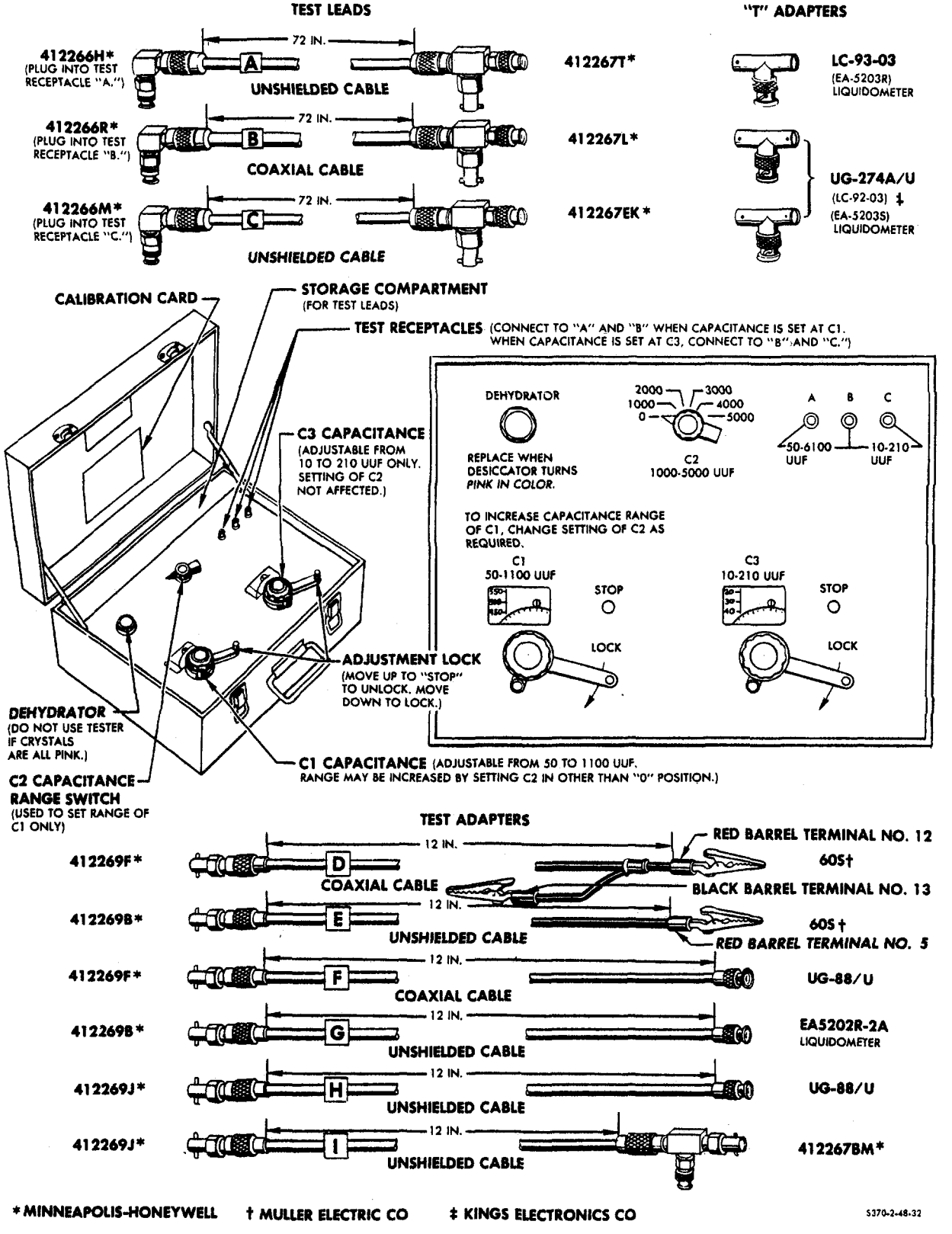
(4) Substitute "C1" section of TTU-68/E or MD-1 tester (figures 28-40-501, 28-40-502, 28-40-503, and 28-40-504) in place of airplane tank units as follows:

(a) Disconnect and ground tank unit low-impedance coaxial cable. (See figures 28-40-504 and 28-40-507.)

(b) Assemble tester wire harness (265-810084) and cable adapters as shown in figure 28-40-504.



TTU -68/E Fuel Quantity Variable-capacitance Tester
Figure 28-40-501



MD-1 Fuel Quantity Variable-capacitance Tester
Figure 28-40-502

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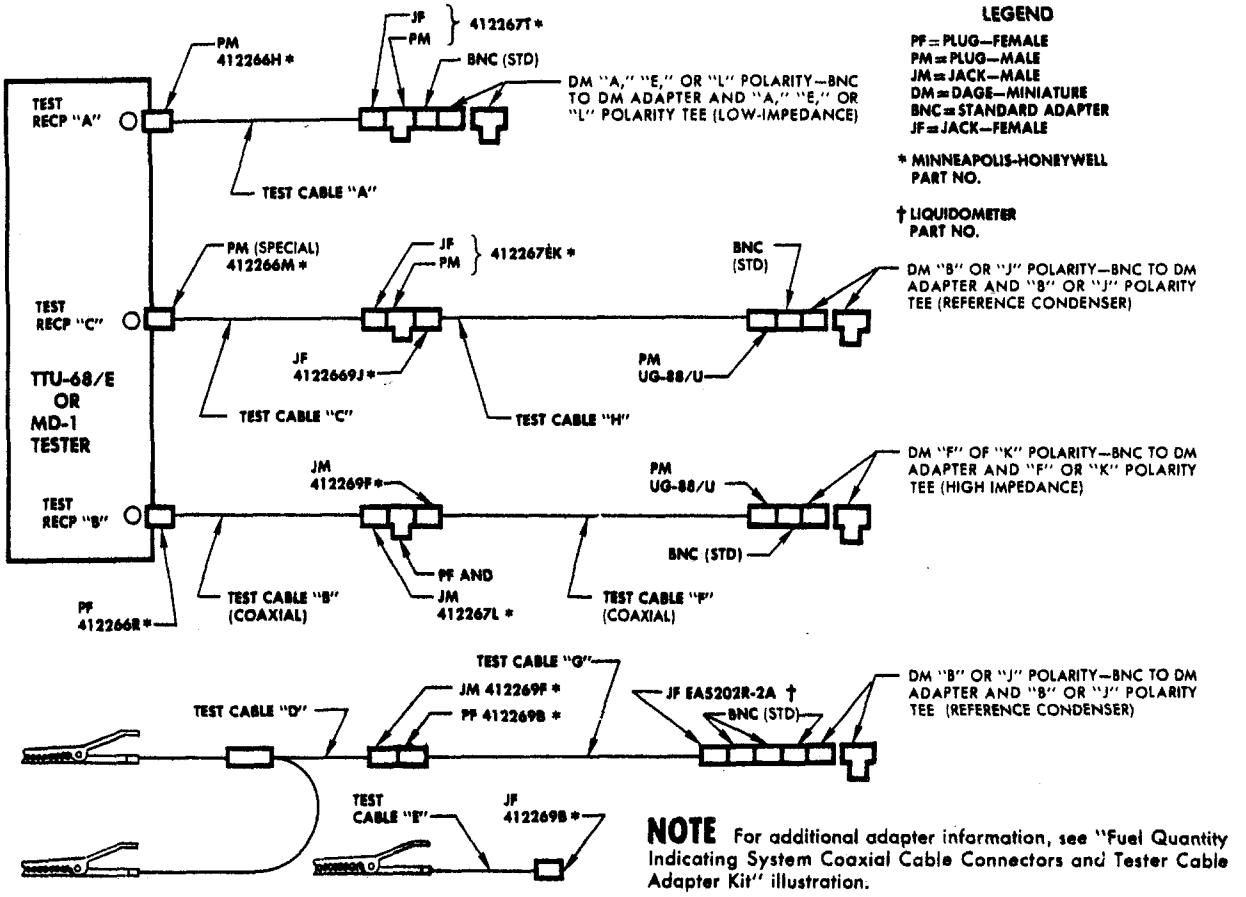

NOTE For additional adapter information, see "Fuel Quantity Indicating System Coaxial Cable Connectors and Tester Cable Adapter Kit" illustration.

CABLE, PLUG, AND TEE ADAPTER PARTS LIST

DESCRIPTION		MFR	PART NO.	QUANTITY
CABLE ADAPTERS MATES BNC STANDARD PLUG—DM JACK	"B" POLARITY	DAGE	1-399-1	2
	"J"		5727-1	2
	"F"		5726-1	1
	"K"		5963-1	1
	"A"		5725-1	1
	"E"		5846-1	1
MATES BNC STANDARD PLUG—DM JACK	"L" POLARITY		5972-1	1
MATES BNC STANDARD JACK—BNC STANDARD JACK		DAGE	3426-1 (OR UG-491B/U)	1
TEE ADAPTERS (JACK-JACK-PLUG) DM	"B" POLARITY	DAGE	1-332-1	2
	"J"		1-965-1	2
	"F"		1-847-1	1
	"K"		1-968-1	1
	"A"		1-333-1	1
	"E"		1-846-1	1
(JACK-JACK-PLUG) DM	"L" POLARITY	DAGE	5235-1	1

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TTU-68/E or MD-1 Tester Cable and Adapter Assembly Schematic - for Testing Fuel Quantity Indicating System
Figure 28-40-503

(c) Disconnect airplane wire harness from fuel quantity indicator. Restore connection by installing tester wire harness (265-810084) between fuel quantity indicator and airplane wire harness as shown in figure 28-40-507.

(d) Connect test cable "A" to tester receptacle A. Connect test cable "B" to tester receptacle B. (See figures 28-40-504 and 28-40-507.) Do not connect tester reference condenser lead (test cable "C") to tester receptacle C.

(5) Adjust "C1" section of TTU-68/E or MD-1 tester until indicator reads zero. The capacitance value read on TTU-68/E or MD-1 tester should be 151.16 (+3.5, -2.0) uuf, indicating that all tank units were properly connected when making the empty adjustment.

(6) Adjust "C1" section of TTU-68/E or MD-1 tester until indicator reads about 500 pounds. Slowly decrease tester value until caution light comes on. Indicator should read 307 (±20) pounds.

B. Full Adjustment

(1) Connect "C1" section of TTU-68/E or MD-1 tester in parallel with tank units and "C3" in parallel with reference condenser as follows:

(a) Reconnect tank unit low-impedance coaxial cable.

(b) Connect test cable "C" to tester receptacle C.

NOTE: Under these conditions, the test setup is exactly as illustrated in figure 28-404-507.

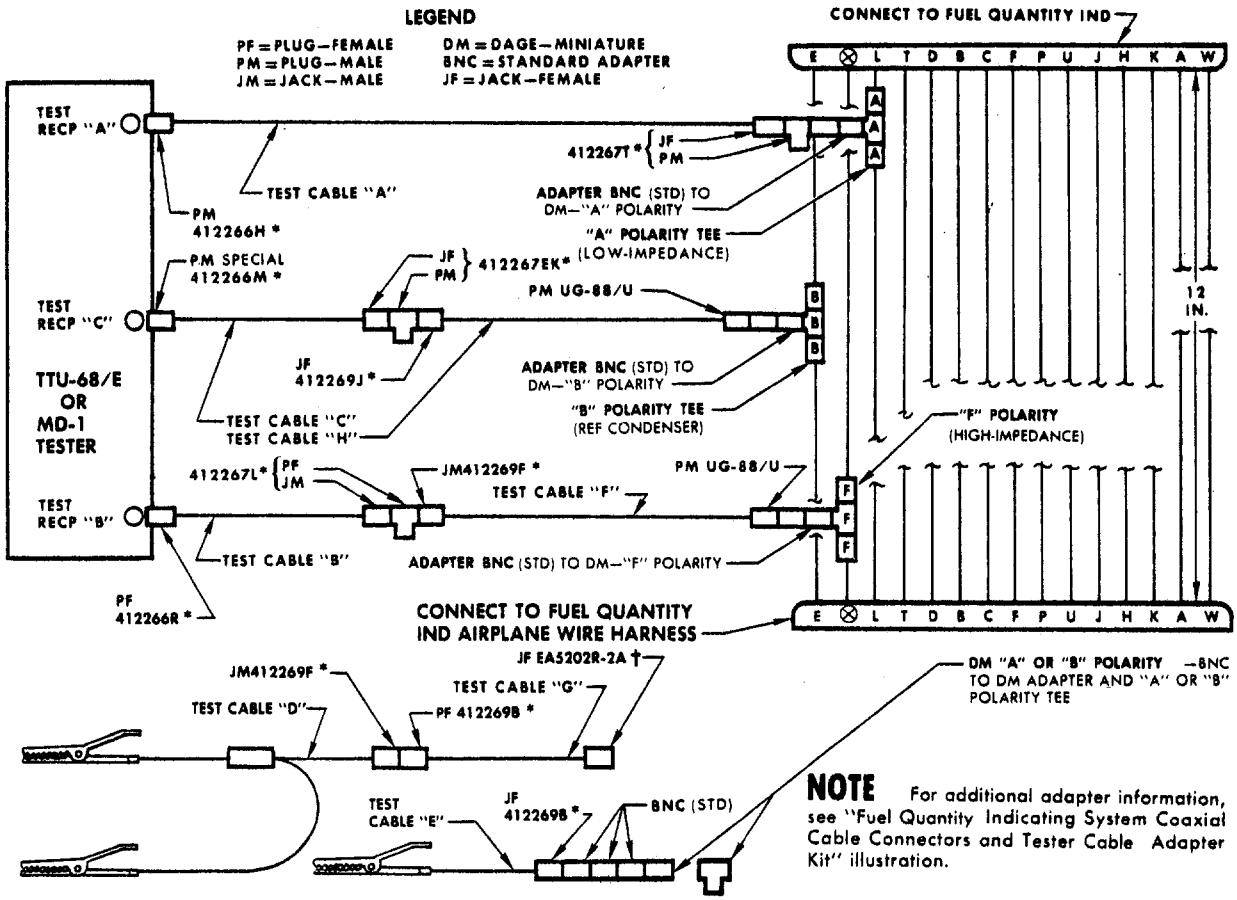
(c) Note that tank unit and tester high-impedance leads are connected in parallel.

(2) Set "C1" section of TTU-68/E or MD-1 tester to capacitance value of 176.59 uuf.

(3) Set "C4" section of TTU-68/E or "C3" section of MD-1 tester to capacitance value (reference condenser) of 31.13 uuf.

(4) Turn "F" adjustment (on back of fuel quantity indicator) until indicator pointer coincides with last scale mark on dial. Then momentarily actuate test switch to move pointer counterclockwise and check repeatability when pointer returns to last scale mark. Look directly into indicator to avoid error due to parallax.

(5) Disconnect and ground coaxial cable leading from tank unit reference condenser.



NOTE For additional adapter information, see "Fuel Quantity Indicating System Coaxial Cable Connectors and Tester Cable Adapter Kit" illustration.

CABLE ADAPTER AND WIRE HARNESS PARTS LIST

DESCRIPTION	MFR	PART NO.	QUANTITY
WIRE HARNESS—265-810084			
CONNECTOR PLUG	AMPHENOL	67-01C18-62P	1
CONNECTOR SOCKET	AMPHENOL	67-06C18-62S	1
DM CABLE PLUG "B" POLARITY	DAGE	1-630-1	2
DM CABLE PLUG "F" POLARITY		1-718-1	2
DM CABLE PLUG "A" POLARITY		1-373-1	2
DM TEE ADAPTER "B" POLARITY (JACK—JACK—JACK)		1-709-1	1
DM TEE ADAPTER "F" POLARITY (JACK—JACK—JACK)		1-711-1	1
DM TEE ADAPTER "A" POLARITY (JACK—JACK—JACK)	DAGE	1-708-1	1
COAX CABLE—SUPRENANT		12219	
WIRE, INSULATED		AN-20	
WIRE, INSULATED WITH COVERED SHIELD		AN-22	
CABLE ADAPTERS			
MATES BNC STANDARD PLUG—DM JACK "B" POLARITY	DAGE	1-399-1	2
MATES BNC STANDARD PLUG—DM JACK "F" POLARITY		5726-1	1
MATES BNC STANDARD PLUG—DM JACK "A" POLARITY		5725-1	2
MATES BNC STANDARD JACK—BNC STANDARD JACK		3426-1	1
MATES DM TEE ADAPTER "B" POLARITY (JACK—JACK—PLUG)		1-332-1	1
MATES DM TEE ADAPTER "A" POLARITY (JACK—JACK—PLUG)	DAGE	1-333-1	1

* MINNEAPOLIS HONEYWELL PART NO. † LIQUIDOMETER PART NO.

5370-2-48-34

TTU-68/E or MD-1 Tester Cable and Adapter Assembly Schematic - for Calibrating Fuel Quantity Indicating System Indicators
Figure 28-40-504

(6) Adjust "C4" section of TTU-68/E or "C3" section of MD-1 tester until indicator pointer coincides with last scale mark on indicator dial. Capacitance value read on tester should be 56.63 (±1.0) uuf, indicating that reference condenser was properly connected in circuit when making the full adjustment.

(7) Disconnect TTU-68/E or MD-1 tester leads and reconnect airplane reference condenser coaxial cable. Make sure tank unit coaxial cables are correctly reconnected.

(8) Check that indicator reads zero. Repeat entire adjustment procedure if "E" setting is incorrect.

(9) Safety "E" and "F" adjustment covers with lockwire and install indicator in panel.

NOTE: The following adjustments are to be made after all the system coaxial leads are reconnected.

(10) Place fuel quantity test switch at RH IND, RH WING, LH IND, AUX and gain access to "L WING" and "R AUX" adjustment cap covers on relay control unit.

NOTE: On some airplanes, two holes (with plug type buttons installed) are provided in the cabin console side panel for access to the fuel quantity unit adjustment screws.

(11) Turn "L WING" adjustment until left fuel quantity indicator reads zero, and turn "R AUX" adjustment until right fuel quantity indicator reads zero.

(12) Place fuel quantity test switch at TOTAL.

2. Testing Fuel Tank Unit Capacitance and Coaxial Cable Insulation Resistance, Using MD-2A Tester

NOTE: A tank unit that appears defective may not actually be defective; there may be a drop of water in the connector. Water in a fuel cell may have bridged the tip of the tank unit, or test leads may be faulty.

The insulation resistance test may be made with the wire and coaxial cable assemblies installed if disconnected from any system component.

CAUTION: AN OHMMETER (0 TO 200 MEGOHMS) MUST BE USED EXCLUSIVELY IN CONDUCTING THE INSULATION RESISTANCE TESTS IN FUEL TANK INTERIORS. DO NOT USE 500-VOLT MEGGER TESTER.

- FAILURE TO OBSERVE POLARITY WHEN CONNECTING CABLE TO POWER SOURCE WILL RESULT IN A SHOCK HAZARD, BLOWING A FUSE, OR OVERLOADING OF THE 115-VOLT AC POWER SOURCE.

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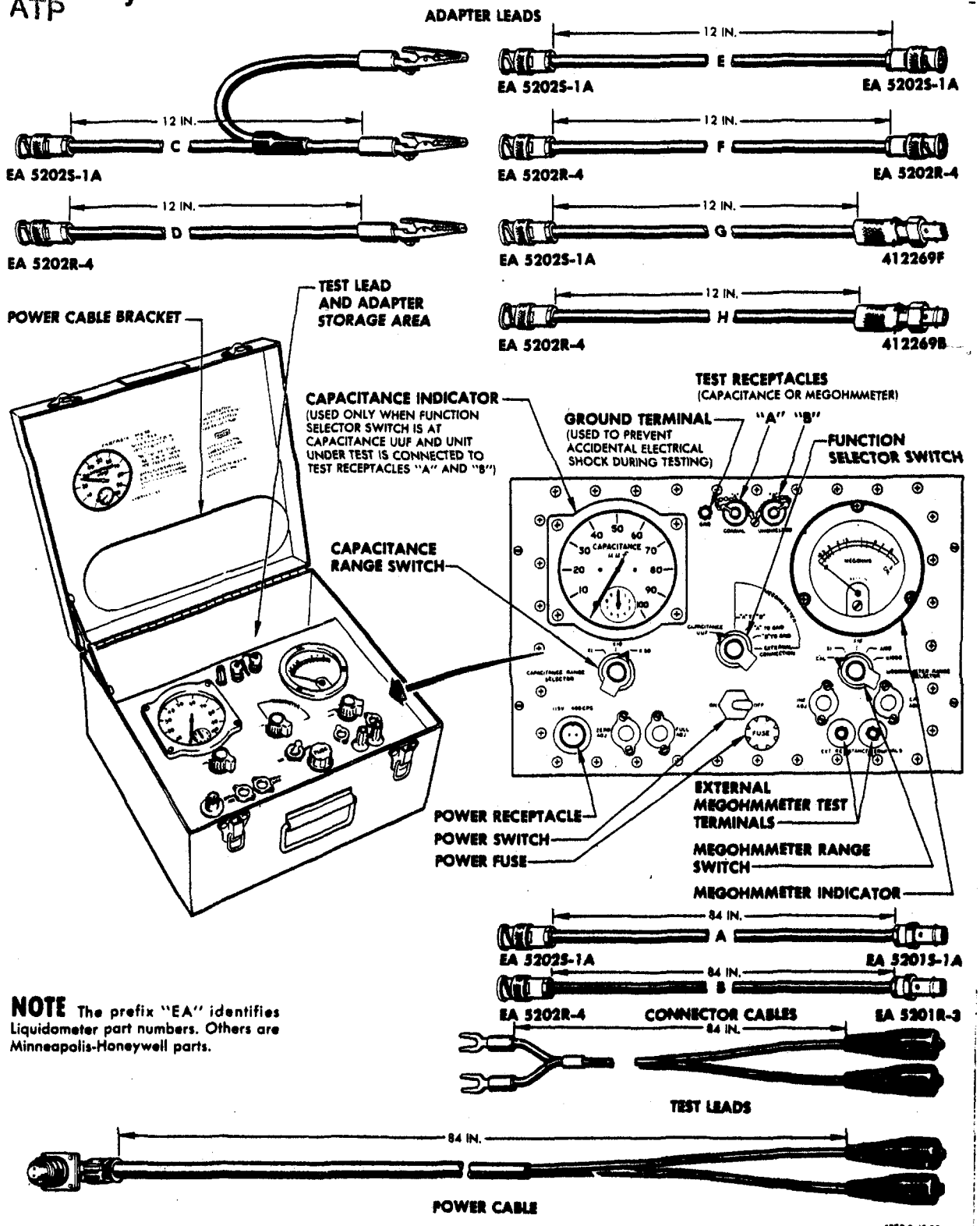

NOTE The prefix "EA" identifies Liquidometer part numbers. Others are Minneapolis-Honeywell parts.

MD-2A Fuel Quantity Capacitance and Megohm Tester
Figure 28-40-505

- TO OBTAIN CORRECT READING, DISCONNECT ALL UNITS AND LEADS NOT BEING MEASURED FROM UNIT UNDER TEST.

- THE TESTER MUST BE EARTH-GROUNDED DURING CALIBRATION AND TESTING OF COMPONENT. FOR AIRPLANE USE, THE AIRPLANE AND TESTER MUST BE EARTH-GROUNDED AS WELL AS GROUNDED TO EACH OTHER.

NOTE: When measuring insulation leakage of an individual circuit unit, the following is to be used as a guide. Between either terminal and ground or between both terminals of a tank unit or the reference condenser, the minimum reading should not be less than 500 megohms; between the coaxial and ground, the reading should not be less than 100 megohms.

- The MD-2A fuel quantity capacitance and megohm tester (figures 28-40-505, 28-40-506, and 28-40-507) is used to test capacitance of tank units.

- The MD-2A tester requires 115 (±10) volts ac, 400 (±40) cps, and 38-volt amperes of power. When connecting power lead, always connect black clip to low side of power (airframe) and red clip to high side of power.

- Allow a minimum of 5 minutes of warm-up time for MD-2A to stabilize. When MD-2A tester is stabilized, the megohmmeter range switch is positioned at CAL. The indicator should read ".5 CAL" (±1/8 inch). When the capacitance range switch is set at "X10," the indicator should read "0" (±3.0) micromicrofarads (uuf). Calibration of MD-2A tester is required if readings exceed tolerances given. MD-2A tester tolerances must be added to capacitance values obtained from table.

A. Perform test as follows:

(1) Connect test lead between tester ground terminal and airplane structure. (See figure 28-40-507.)

(2) Disconnect all other circuit units from unit being tested and connect applicable adapter lead (figure 28-40-506 or 28-40-507).

(3) Measure capacitance of each tank unit. Then read value from appropriate indicator on MD-2A. Capacitance values can be obtained from the following table of tank unit.

TANK UNIT CAPACITANCE (DRY VALUES)

TANK	TANK UNIT NO.	TANK UNIT PART NO.	CAPACITANCE		
			MINIMUM	AVERAGE	MAXIMUM
LH and RH	1	B347-2063 or 010-003-001	20.08 20.33	20.58 20.83	21.08 21.33
LH and RH	2	B347-2064 or 010-003-002	27.22 27.49	27.72 27.99	28.22 28.49
LH and RH	3	B347-2065 or 010-003-003	18.56 18.80	19.06 19.30	19.56 19.80
LH and RH	4	B347-2066 or 010-003-004	31.78 32.28	32.28 32.78	32.78 33.28
LH and RH	5	EA772-2067 or 020-006-001	11.20 11.20	11.70 11.70	12.20 12.20
LH and RH	6	EA772-2068 or 020-006-002	6.91 6.91	7.41 7.41	7.91 7.91
LH and RH	7	EA818-2069 reference condenser or 021-002-001 reference condenser	2.13 25.00 2.53 25.50	2.63 25.50 3.03 26.00	3.13 26.00 3.53 26.50
Auxiliary fuselage	8	034-001-002 (each circuit)	29.16	29.66	30.16

3. Adjusting Fuel Quantity Indicating System, Using TF20-1 Tester

The following procedure should be performed on each indicating system. The tank units must be dry when the adjustment is being made; therefore, make sure tanks are drained. Make empty and full adjustments (accessible on back of indicator) by sliding indicator out of panel or removing panel from frame, taking care not to damage coaxial and wire leads. Disconnect connectors from indicator not being adjusted.

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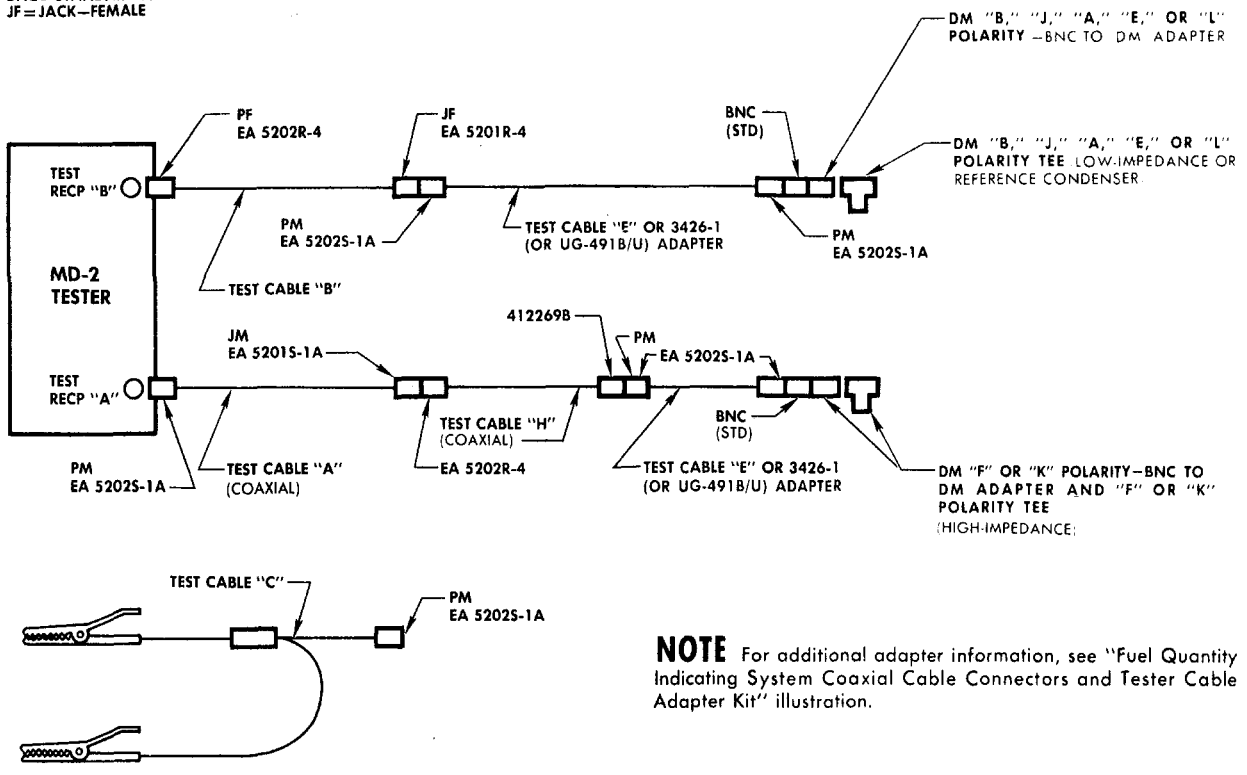
LEGEND
PF = PLUG—FEMALE
PM = PLUG—MALE
JM = JACK—MALE
DM = DAGE—MINIATURE
BNC = STANDARD ADAPTER
JF = JACK—FEMALE
DM "B," "J," "A," "E," OR "L" POLARITY —BNC TO DM ADAPTER
PF EA 5202R-4
JF EA 5201R-4
BNC (STD)
DM "B," "J," "A," "E," OR "L" POLARITY TEE LOW-IMPEDANCE OR REFERENCE CONDENSER
TEST RECP "B"
PM EA 5202S-1A
TEST CABLE "E" OR 3426-1 (OR UG-491B/U) ADAPTER
PM EA 5202S-1A
MD-2 TESTER
TEST CABLE "B"
412269B
PM EA 5202S-1A
JM EA 5201S-1A
TEST RECP "A"
TEST CABLE "H" (COAXIAL)
BNC (STD)
PM EA 5202S-1A
TEST CABLE "A" (COAXIAL)
EA 5202R-4
TEST CABLE "E" OR 3426-1 (OR UG-491B/U) ADAPTER
DM "F" OR "K" POLARITY—BNC TO DM ADAPTER AND "F" OR "K" POLARITY TEE (HIGH-IMPEDANCE)
TEST CABLE "C"
PM EA 5202S-1A

NOTE For additional adapter information, see "Fuel Quantity Indicating System Coaxial Cable Connectors and Tester Cable Adapter Kit" illustration.

CABLE AND TEE ADAPTER PARTS LIST

DESCRIPTION		MFR	PART NO.	QUANTITY
CABLE ADAPTERS				
MATES BNC STANDARD PLUG—DM JACK	"B" POLARITY	DAGE	1-399-1	1
	"J"		5727-1	1
	"F"		5726-1	1
	"K"		5963-1	1
	"A"		5725-1	1
	"E"		5846-1	1
MATES BNC STANDARD PLUG—DM JACK	"L" POLARITY	DAGE	5972-1	1
MATES BNC STANDARD JACK—BNC STANDARD JACK		DAGE	3426-1 (OR UG-491B/U)	2
TEE ADAPTERS				
(JACK-JACK-PLUG) DM TEE ADAPTER	"B" POLARITY	DAGE	1-332-1	1
	"J"		1-965-1	1
	"F"		1-847-1	1
	"K"		1-968-1	1
	"A"		1-333-1	1
	"E"		1-846-1	1
(JACK-JACK-PLUG) DM TEE ADAPTER	"L" POLARITY	DAGE	5235-1	1

S370-2-48-35

Figure 28-40-506 — MD-2A Tester Cable and Adapter Assembly Schematic — for Testing Fuel Quantity Indicating System

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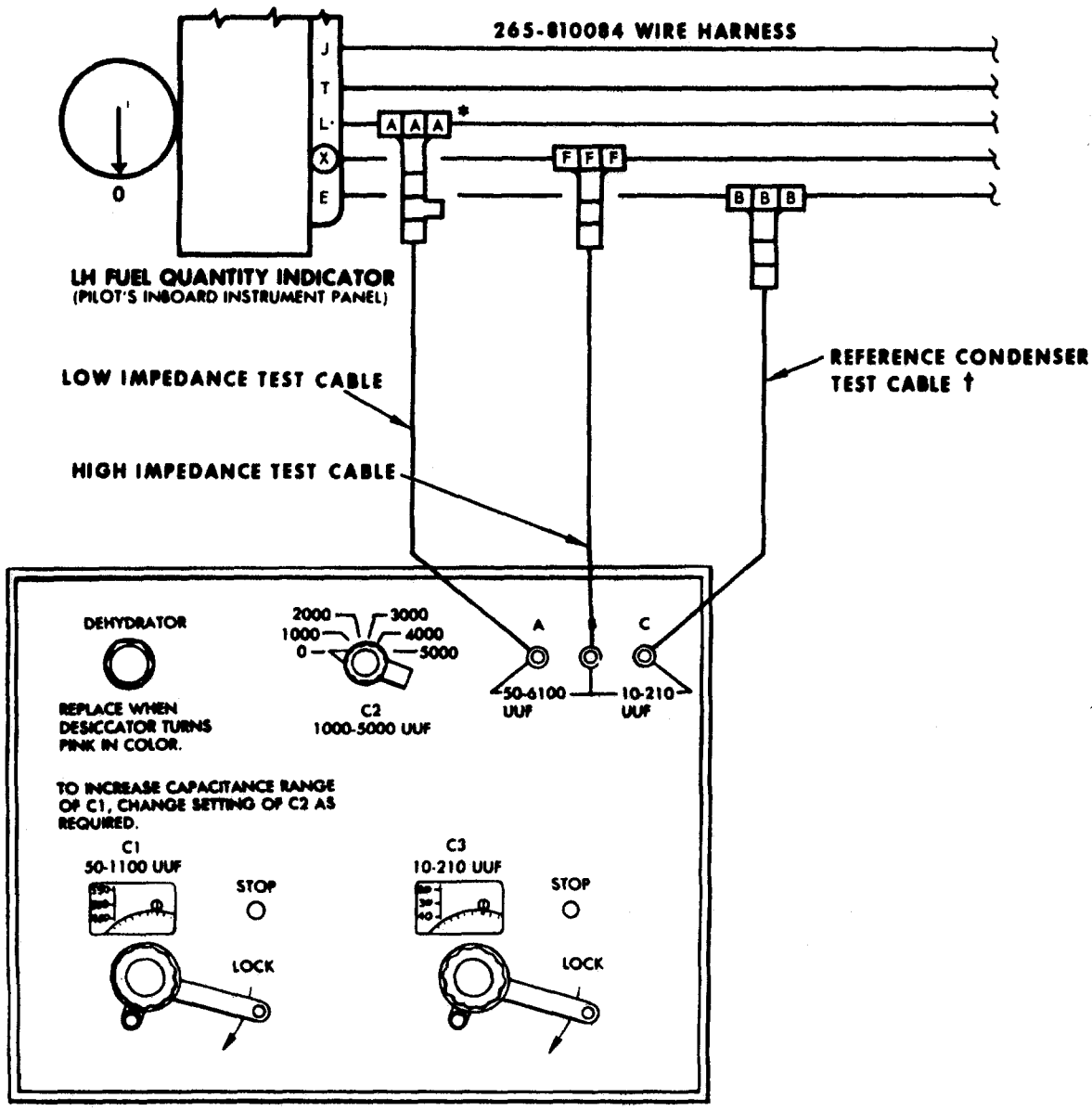

* DISCONNECTED AND GROUNDED WHEN PERFORMING "EMPTY" CHECK

† NOT CONNECTED WHEN PERFORMING "EMPTY" CHECK

S265-2-48-29A

Figure 28-40-507 — Fuel Quantity Indicating System and Test Setup (Sheet 1 of 2)

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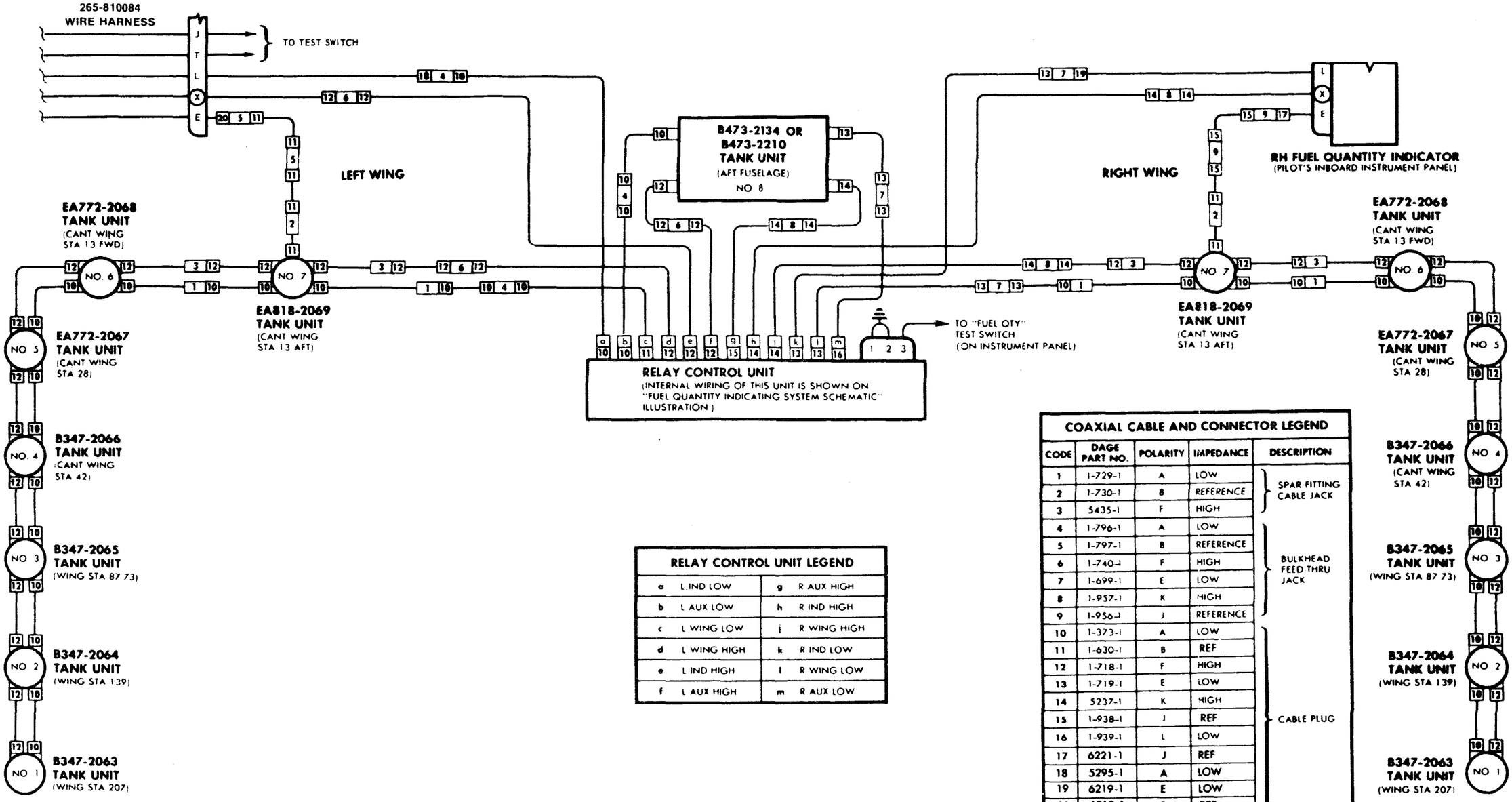

RELAY CONTROL UNIT LEGEND			
a	L IND LOW	g	R AUX HIGH
b	L AUX LOW	h	R IND HIGH
c	L WING LOW	i	R WING HIGH
d	L WING HIGH	k	R IND LOW
e	L IND HIGH	l	R WING LOW
f	L AUX HIGH	m	R AUX LOW

COAXIAL CABLE AND CONNECTOR LEGEND				
CODE	DAGE PART NO.	POLARITY	IMPEDANCE	DESCRIPTION
1	1-729-1	A	LOW	SPAR FITTING CABLE JACK
2	1-730-1	B	REFERENCE	
3	5435-1	F	HIGH	
4	1-796-1	A	LOW	BULKHEAD FEED THRU JACK
5	1-797-1	B	REFERENCE	
6	1-740-1	F	HIGH	
7	1-699-1	E	LOW	
8	1-957-1	K	HIGH	
9	1-956-1	J	REFERENCE	
10	1-373-1	A	LOW	CABLE PLUG
11	1-630-1	B	REF	
12	1-718-1	F	HIGH	
13	1-719-1	E	LOW	
14	5237-1	K	HIGH	
15	1-938-1	J	REF	
16	1-939-1	L	LOW	
17	6221-1	J	REF	
18	5295-1	A	LOW	
19	6219-1	E	LOW	
20	6218-1	B	REF	

S265-2-48-29A

Figure 28-40-507 — Fuel Quantity Indicating System and Test Setup (Sheet 2 of 2)

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NOTE: See figure 28-40-2 for fuel quantity test switch selections.

A. Empty Adjustment

(1) Defuel airplane.

(2) Connect 28-volt dc external power to airplane, and push in the following circuit breakers:

- "INVERTERS - CONTROL - NO. 2 & INST"
- "INVERTERS - PWR TRIP - NO. 2"
- "FUEL QTY - LH"
- "FUEL QTY - RH"
- "FUEL WARNING - LEVEL"

Place "INVERTER" switch at NO. 2. Allow 2 minutes for fuel quantity indicating system to stabilize before making adjustments with fuel quantity test switch at center TOTAL position.

(3) Turn "E" adjustment (on back of fuel quantity indicator) until indicator reads zero. Momentarily actuate fuel quantity test switch to move pointer off set position and check repeatability when pointer returns to zero. Look directly into indicator to avoid error due to parallax.

(4) Connect TF20-1 test set to 115-volt ac 400-cycle power source. Ground test set (to airplane, if power is taken from this source). Allow 5 minutes for test set to warm up.

(5) Disconnect airplane wire harness from fuel quantity indicator. Restore connection by installing test wire harness 265-810084, part of adapter kit 265-810081, between fuel quantity indicator and airplane wire harness.

NOTE: Assemble tester wire harness and cable adapters as shown in figure 28-40-504.

- For empty calibration, the cable hookup should be as shown in figure 28-40-509, detail A.
- Figures in parentheses refer to control identification numbers in figure 28-40-508.

(6) Place "CAP - RES CHECK" switch (1) at CAP, "RANGE SELECTOR" switch (2) at X3, and "FUNCTION SELECTOR" switch (3) at TEST. Loosen c clamp (4) and rotate "PROBE 25-250 MMF" knob (5) until fuel quantity indicator in airplane reads zero.

(7) Place "FUNCTION SELECTOR" switch (3) at PROBE SET. The "CAPACITANCE INDICATOR" should read 151.16 (+3.5, -2.0), indicating that the tank units were properly connected when the empty adjustment was made.

(8) Place "FUNCTION SELECTOR" switch (3) at PROBE SET and rotate "PROBE 25-250 MMF" knob (5) until "CAPACITANCE INDICATOR" reads 57.70 uuf (57.70 x 3 equals 173.10 uuf). Tolerance is ±2.00 uuf.

(9) Place "FUNCTION SELECTOR" switch (3) at TEST. The fuel quantity indicator should read 500 (±20) pounds. With "PROBE 25-250 MMF" knob, decrease tester value until fuel level caution light comes on. The fuel quantity indicator should read 307 (±20) pounds.

B. Full Adjustment

(1) Connect TF20-1 tester as shown in figure 28-40-509, detail B.

(2) Place "FUNCTION SELECTOR" switch (3) at PROBE SET. Rotate "PROBE 25-250 MMF" knob (5) until "CAPACITANCE INDICATOR" reads 58.86 uuf (58.86 x 3 equals 176.58 uuf). Place "FUNCTION SELECTOR" switch (3) at COMP SET and "RANGE SELECTOR" switch (2) at X1; then loosen clamp (6) and adjust "COMP 25-250 MMF" knob (7) until "CAPACITANCE INDICATOR" reads 31.13 uuf. Lock clamp (6). Return "RANGE SELECTOR" switch (2) to X3. Place "FUNCTION SELECTOR" switch (3) at TEST. Fuel quantity indicator should indicate approximately FULL. Adjust fuel quantity indicator full adjustment until indicator pointer coincides with last scale mark on dial. Look directly into indicator to avoid error due to parallax.

(3) Disconnect and ground reference condenser lead at point "A" as shown in figure 28-40-509, detail B.

(4) Place "FUNCTION SELECTOR" switch (3) at TEST and "RANGE SELECTOR" switch (2) at X3; then loosen clamp (6) and rotate "COMP 25-250 MMF" knob until fuel quantity indicator pointer is on last scale mark. Place "FUNCTION SELECTOR" switch (3) at COMP SET and "RANGE SELECTOR" switch at X1. The "CAPACITANCE INDICATOR" should read 56.63 (±1.00) uuf, indicating that the reference condenser was properly connected in the circuit when the full adjustment was made.

(5) Disconnect tester leads, and unground and reconnect airplane reference condenser. The fuel quantity indicator should read zero. Repeat entire adjustment procedure if "E" setting is incorrect.

(6) Safety "E" and "F" adjustment covers with lockwire and install indicator in panel.

NOTE: The following adjustments are to be made after all the system coaxial leads are reconnected.

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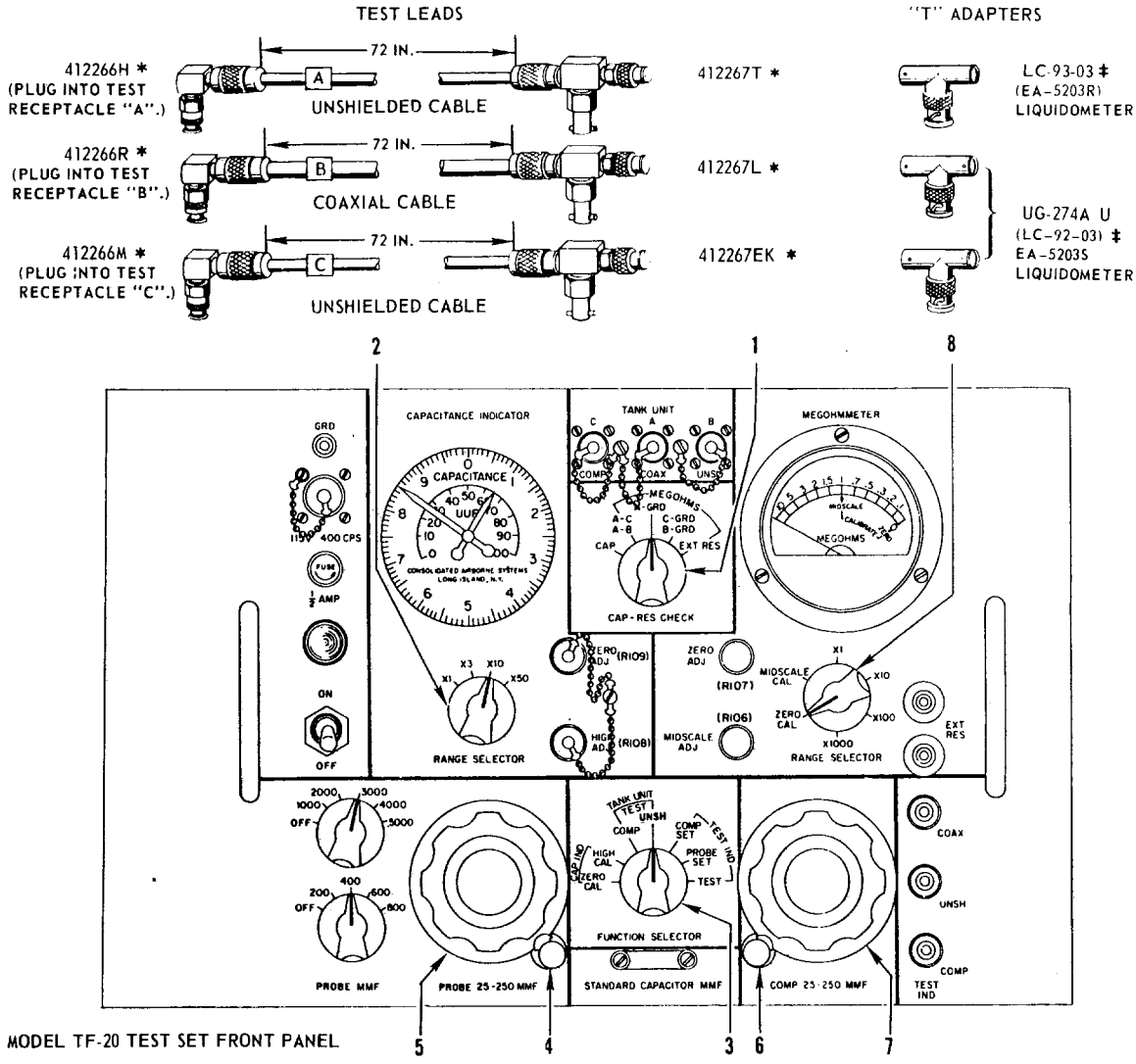
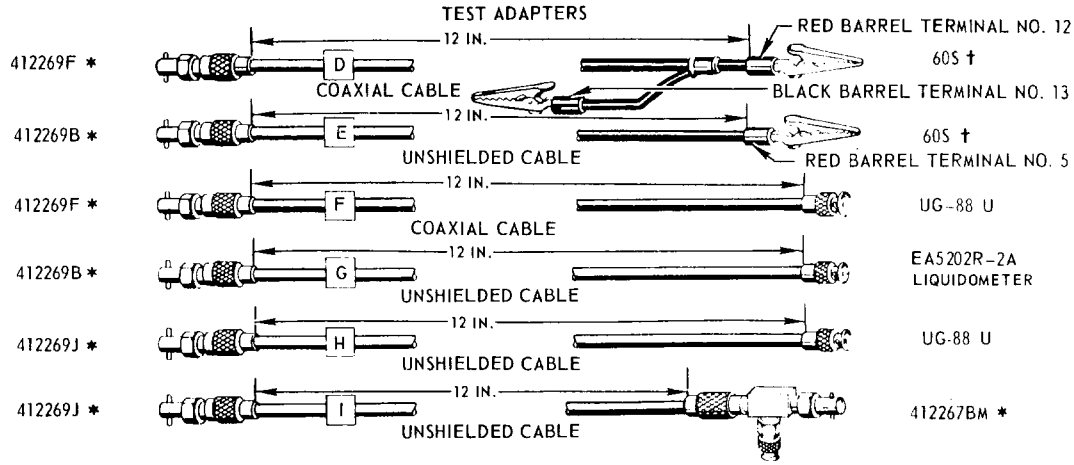


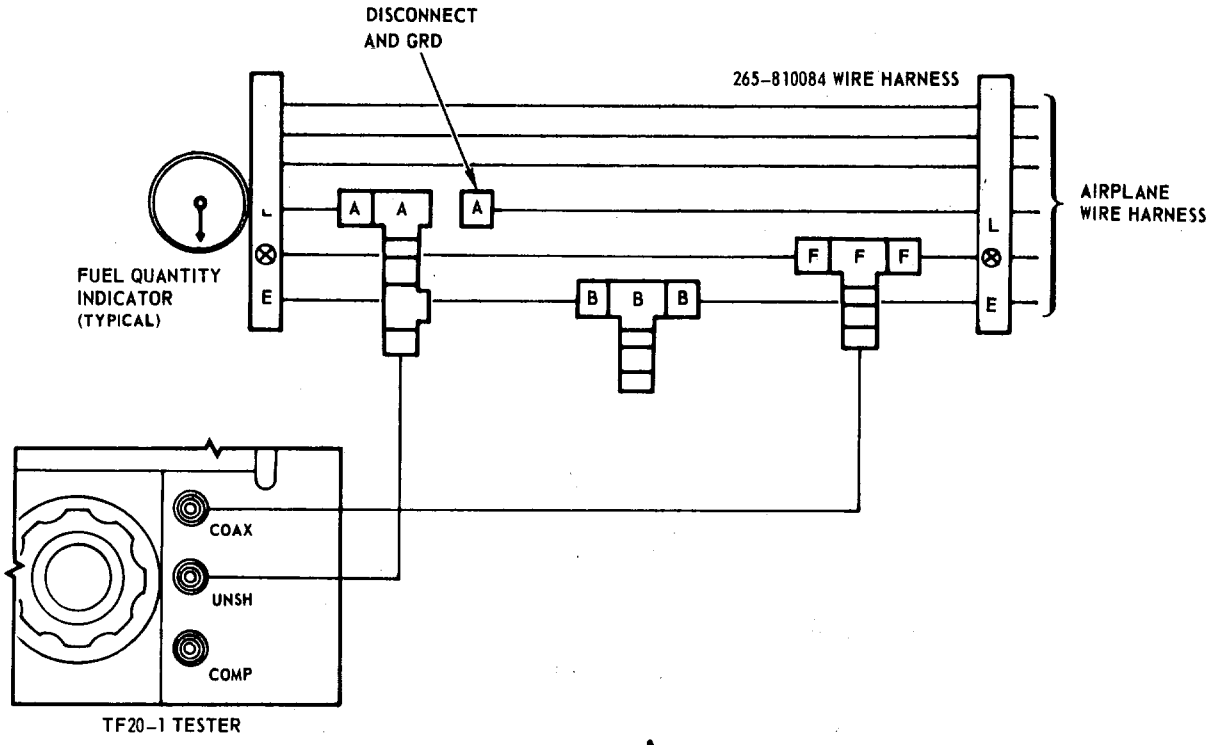
MODEL TF-20 TEST SET FRONT PANEL



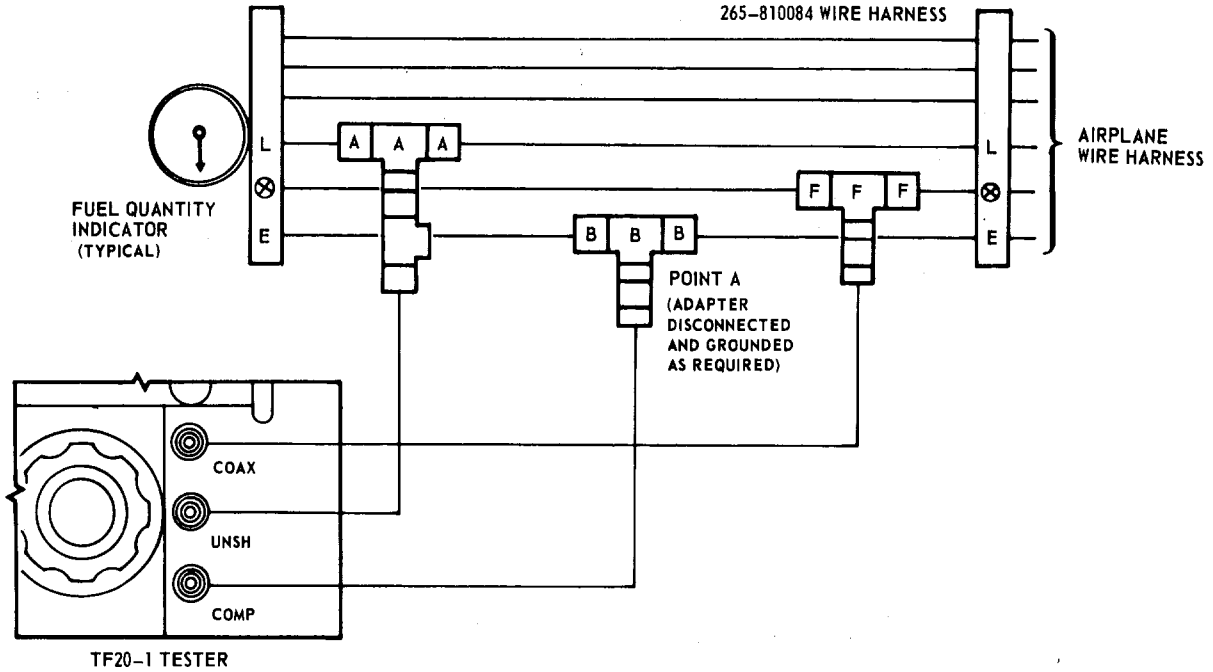
* MINNEAPOLIS-HONEYWELL † MULLER ELECTRIC CO ‡ KINGS ELECTRONICS CO

5370-2-48-39

TF20-1 Fuel Quantity Variable-capacitance Tester
Figure 28-40-508



DETAIL A



DETAIL B

S370-2-48-3B

TF20-1 Tester Cable and Adapter Assembly Schematic
Figure 28-40-509

(7) Place fuel quantity test switch at RH IND, RH WING, LH IND, and AUX and remove "L WING" and "R AUX" adjustment cap covers from relay control unit.

(8) Turn "L WING" adjustment until fuel quantity indicator reads zero; and turn "R AUX" adjustment until right fuel quantity indicator reads zero.

(9) Install adjustment cap covers on relay control unit.

(10) Place fuel quantity test switch at TOTAL.

4. Testing Tank Unit Probe Capacitance, Using TF20-1 Tester

The TF20-1 tester can be used to check capacitance of individual probes, or total probe capacitance or segments thereof.

NOTE: To accomplish these tests, do not have power applied to the fuel quantity indicating system.

- A tank unit that appears defective may not actually be defective; there may be a drop of water in the connector. Water in a fuel cell may have bridged the tip of the tank unit, or test leads may be faulty.

- Figures in parentheses refer to control identification number in figure 28-40-508.

CAUTION: THE TESTER MUST BE EARTH-GROUNDED DURING CALIBRATION AND TESTING OF COMPONENTS. FOR AIRPLANE USE, THE AIRPLANE AND TESTING MUST BE EARTH-GROUNDED AS WELL AS GROUNDED TO EACH OTHER.

A. Check Individual Probes

(1) Connect coaxial lead to "TANK UNIT" terminal "A" and the unshielded lead to terminal "B."

(2) Place "CAP - RES CHECK" switch (1) at CAP, "FUNCTION SELECTOR" switch (3) at UNSH, and "RANGE SELECTOR" switch (2) at X1.

(3) Read capacitance directly on "CAPACITANCE INDICATOR." The reading should be as shown in the following table of tank unit capacitance (dry values).

TANK UNIT CAPACITANCE (DRY VALUES)

TANK	TANK UNIT NO.	TANK UNIT PART NO.	CAPACITANCE MINIMUM	CAPACITANCE AVERAGE	CAPACITANCE MAXIMUM
LH and RH	1	B347-2063 or 010-003-001	20.08 20.33	20.58 20.83	21.08 21.33
LH and RH	2	B347-2064 or 010-003-002	27.22 27.49	27.72 27.99	28.22 28.49
LH and RH	3	B347-2065 or 010-003-003	18.56 18.80	19.06 19.30	19.56 19.80
LH and RH	4	B347-2066 or 010-003-004	31.78 32.28	32.28 32.78	32.78 33.28
LH and RH	5	EA772-2067 or 020-006-001 or 020-006-003	11.20 11.20 11.20	11.70 11.70 11.70	12.20 12.20 12.20
LH and RH	6	EA772-2068 or 020-006-002 or 020-006-004	6.91 6.91 6.91	7.41 7.41 7.41	7.91 7.91 7.91
LH and RH	7	EA818-2069 reference condenser or 021-002-001 or -002 reference condenser	2.13 25.00 2.53 25.50	2.63 25.50 3.03 26.00	3.13 26.00 3.53 26.50
Auxiliary Fuselage	8	034-001-002 (each circuit)	29.16	29.66	30.16

B. Checking Total Capacitance

(1) Connect coaxial cable to "TANK UNIT" terminal "A," unshielded cable to terminal "B," and compensator cable to terminal "C."

(2) Disconnect fuel quantity indicator.

(3) Place "CAP - RES CHECK" switch (1) at CAP, "FUNCTION SELECTOR" switch (3) at UNSH, and "RANGE SELECTOR" switch (2) at X3. "CAPACITANCE INDICATOR" should read 50.34 uuf (50.34 x 3 equals 151.02 uuf). Tolerance is +3.5, -2.0 uuf.

C. Checking Capacitance of Compensator

(1) Connect coaxial cable to "TANK UNIT" terminal "A," unshielded cable to terminal "B," and compensator cable to terminal "C."

(2) Place "CAP - RES CHECK" switch (1) at CAP, "FUNCTION SELECTOR" switch (3) at COMP, and "RANGE SELECTOR" switch (2) at X1. "CAPACITANCE INDICATOR" should read from 25.50 uuf to 26.50 uuf.

NOTE: Any segment may be checked for capacitance by following the procedure in steps (1) and (2) of paragraph A. However, if the total capacitance of the segment is more than 100 uuf, the "RANGE SELECTOR" switch should be placed at X3. The indicated value is then multiplied by 3.

5. Testing Probe and Insulation Resistance, Using TF20-1 Tester

NOTE: Calibrate "MEGOHMMETER" at zero and midscale.

- Test set should be connected as in step (1) of paragraph 3. B.
- Figures in parentheses refer to control identification numbers in figure 28-40-508.

A. Measuring Resistance Between "TANK UNIT" Terminals "A" and "B"

(1) Place "CAP - RES CHECK" switch (1) at A-C A-B and "FUNCTION SELECTOR" switch (3) at UNSH. Adjust "RANGE SELECTOR" switch (8) to indicate closest to mid-scale of "MEGOHMMETER."

NOTE: When insulation leakage of an individual unit circuit is being measured, the following is to be used as a guide. Between either terminal and ground or between both terminals of a tank unit or the reference condenser, the minimum reading should be not less than 500 megohms. The reading should be not less than 100 megohms between the coaxial cable connector and ground.

B. Measuring Resistance Between "TANK UNIT" Terminals "A" and "C"

(1) Place "CAP - RES CHECK" switch (1) at A-C A-B and "FUNCTION SELECTOR" switch at COMP. Adjust "RANGE SELECTOR" switch (8) to indicate closest to mid-scale of "MEGOHMMETER." [Refer to NOTE following step (1) of paragraph A.]

C. Measuring Resistance Between "TANK UNIT" Terminal "A" and Ground

(1) Place "CAP - RES CHECK" switch (1) at A-GRD and "FUNCTION SELECTOR" switch (3) at COMP. Adjust "RANGE SELECTOR" switch (8) to indicate closest to mid-scale of "MEGOHMMETER." [Refer to NOTE following step (1) of paragraph A.]

D. Measuring Resistance Between "TANK UNIT" Terminal "B" and Ground

(1) Place "CAP - RES CHECK switch (1) at C-GRD B-GRD and FUNCTION SELECTOR" switch (3) at UNSH. Adjust "RANGE SELECTOR" switch (8) to indicate closest to mid-scale of "MEGOHMMETER." Refer to NOTE following step (1) of paragraph A.

6. Circuit Polarization

The coaxial cable assembly connectors have been chosen so that it is impossible to misconnect the cable assemblies at the tank units and at the interconnect points in the airplane. Following is a list of the Dage connector polarities and their applications.

POLARITY	APPLICATION
"F"	All tank unit high-impedance leads except "K" polarity applications listed

POLARITY	APPLICATION
"A"	All tank unit low-impedance leads except "E" polarity applications listed
"B"	All reference condenser leads except "J" polarity applications listed
"K"	Auxiliary tank unit high-impedance leads to indicator and passengers' compartment seal junction in right system only
"E"	Auxiliary tank unit low-impedance leads to indicator and passengers' compartment seal junction in right system only
"J"	Airplane reference condenser lead between passengers' compartment seal and indicator in right system only
"B"	Left wing system low-impedance lead at relay control unit (figure 28-40-2)
"J"	Auxiliary tank unit right system high-impedance lead at relay control unit (figure 28-40-2)
"L"	Auxiliary tank unit right system low-impedance lead at relay control unit (figure 28-40-2)

7. Testing Fuel Level Caution System

A. Perform test as follows:

(1) Connect 28-volt dc external power supply to airplane.

(2) Check that "FUEL WARNING - IEVEL" circuit breaker is open.

NOTE: This procedure checks out the caution light bulbs, the caution system wiring from the indicators, and the circuit breaker.

(3) Remove plug from left and right fuel quantity indicator.

(4) Connect a jumper wire across pins "J" and "K" of plug removed from left indicator.

(5) Close "FUEL WARNING - LEVEL" circuit breaker. Caution light should come on. Open circuit breaker.

(6) Repeat steps (4) and (5) for right fuel quantity indicator.

1. Fuel Quantity Indicating System Check-out Procedure

A. Special tools and/or equipment needed to check the fuel quantity indicating system are as follows:

(1) Multimeter (similar to Model 260, Simpson Electric Co) Test set, indicator (Type TTU-68/E, General Radio Co) or

(2) Tester, liquid quantity gage (Type MD-1, General Radio Co)

(3) Tester, liquid quantity system (Type MD-2A, Consolidated Airborne Systems)

(4) Ohmmeter (0 to 200 megohms)

(5) Cable adapter kit (265-810081)

B. Perform operational check-out of fuel quantity indicating system as follows:

(1) Connect 28-volt dc and place inverter switch at NO. 1.

(2) Push in "FUEL QTY - LH," "FUEL QTY - RH," and "FUEL WARNING - LEVEL" circuit breakers. (See figure 28-40-2.)

(3) Allow about 2 minutes for power unit to warm up and stabilize. Place fuel quantity test switch at TOTAL. position and note position of fuel quantity indicator pointers. Record pointer positions.

(4) Place fuel quantity test switch at TEST. (See figure 28-40-2.) The left and right fuel quantity indicator pointers should rotate counterclockwise. With the Gull indicator installed, the indicator pointers will stop below the zero position when the test switch is at TEST.

(5) Return fuel quantity test switch to TOTAL. The indicator pointers should return to their former positions (recorded in step 3).

(6) Place fuel quantity test switch at RH IND, RH WING, LH IND, AUX and check left fuel quantity indicator for auxiliary tank fuel quantity and right indicator for right wing tank fuel quantity. Then return fuel quantity test switch to TOTAL.

2. Fuel Low-pressure Warning System Check-out Procedure

A. Checking Fuel Pressure Switches

(1) Make sure there is enough fuel in wings for an engine-run operation.

(2) Start right-hand engine.

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(3) Maintain engine at idle during this check. Disengage "BOOST PUMP - RH FUEL" circuit breaker. Check whether "RH FUEL PRESS LOW" caution light, on caution-warning light panel, comes on, indicating fuel pressure switch is operating properly when boost pump fuel pressure is 5 (±1) psi or less.

(4) Engage "BOOST PUMP - RH FUEL" circuit breaker. The "RH FUEL PRESS LOW" caution light should go out, indicating fuel pressure switch is operating properly when boost pump pressure is more than 5 (±1) psi.

NOTE: When the engine is running, "RH FUEL PRESS LOW" caution light is out, and steps (2) and (3) are complied with, the fuel pressure differential switch monitoring the engine-driven fuel pump is operating properly.

(5) Shut down right-hand engine. As engine revolutions decrease, check that "RH FUEL PRESS LOW" caution light comes on, indicating that fuel pressure differential switch monitoring engine-driven fuel pump pressure, is operating properly when differential pressure is less than 6 (±1) psi.

(6) Check "LH FUEL PRESS LOW" caution light, using the same procedure as in steps (1) through (4); however, check that "BOOST PUMP - LH FUEL" instead of "BOOST PUMP - RH FUEL" circuit breaker is disengaged and re-engaged in steps (3) and (4).

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CHAPTER 29

HYDRAULIC POWER

CHAPTER 29

HYDRAULIC POWER

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CHAPTER 29

HYDRAULIC POWER

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*The asterisk indicates pages revised, added, or deleted by the current revision.

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HYDRAULIC POWER - DESCRIPTION

1. General

The hydraulic power is provided by a 3000 (±100) psi pressure (closed-center) type system, containing the necessary units to provide, maintain, and control hydraulic pressure to the subsystems. The hydraulic power system includes a normal pressure system, an auxiliary pressure system, a pump electrical control system, a reservoir air pressurization system, and a pressure-indicating system.

The normal system supplies hydraulic pressure for the operation of the following subsystems: landing gear, wheel brakes, nose wheel steering, speed brake, and on airplanes serial numbers NA-370-2 through -6, -8 and -9, the thrust reverser doors (targets). The auxiliary system, controlled by a switch in the cockpit, is for emergency or alternate use. In case of normal power system failure, the auxiliary system, through separate pressure lines, provides sufficient fluid and pressure to operate the following subsystems for a limited number of operations: wheel brakes, nose wheel steering, and speed brake. On airplanes NA-370-1 and -7, NA-380-1 and later, and airplanes NA-370-2 thru -6, -8 and -9 that are changed by Sabreliner Optional Kit No. 11; the auxiliary system will assist the normal system in retracting the landing gear whenever the normal hydraulic system pressure drops below 2000 (±100) psi. The pump electrical control system controls electrical power to a single dc-motor-powered hydraulic pump. The reservoir air pressurization system provides a constant pressure to the reservoir, regardless of altitude, to prevent pump cavitation. The hydraulic pump obtains its fluid supply from the reservoir, which is pressurized to 35 psi by bleed air from each engine compressor section. The pressure indicating system consists of two transmitters which sense normal and auxiliary hydraulic system pressures, and electrically transmit pressure indications to two cockpit hydraulic pressure indicators. System relief valves protect the hydraulic system against excessive pressures which might build up in the system. Fluid filters in the hydraulic power system protect the subsystems from foreign matter that may enter the power system. A ground test stand may be connected to a panel on the lower side of the fuselage rear section for ground testing and operation of all the hydraulic systems in the airplane. On NA-370-9 and NA-380-1 and later airplanes, a hydraulic pressure warning system is installed to provide an audible warning whenever the hydraulic system pressure decreases below 2000, or the hydraulic pump control circuit breaker is disengaged (pulled out) or if the hydraulic pump control switch is positioned at OFF.

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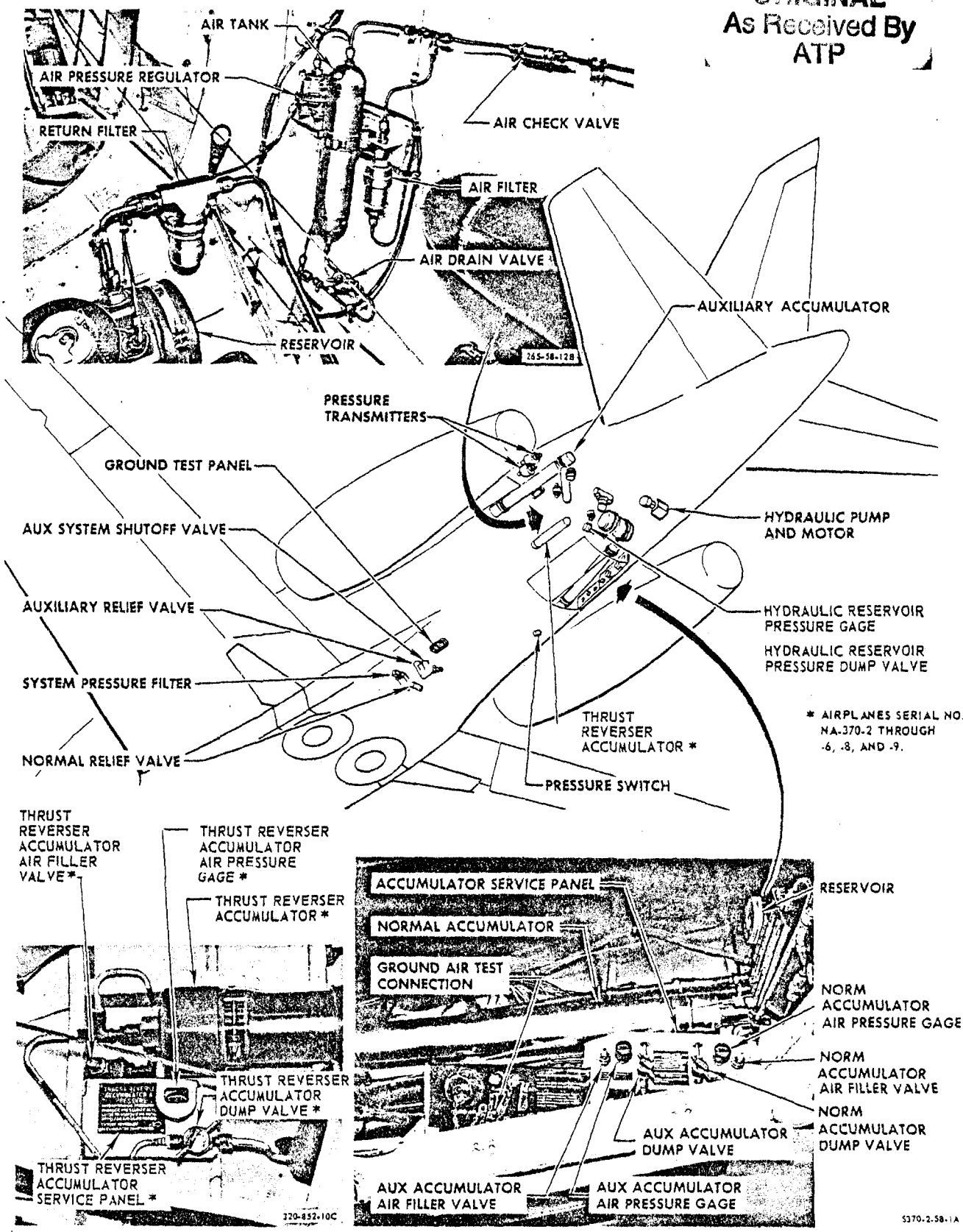
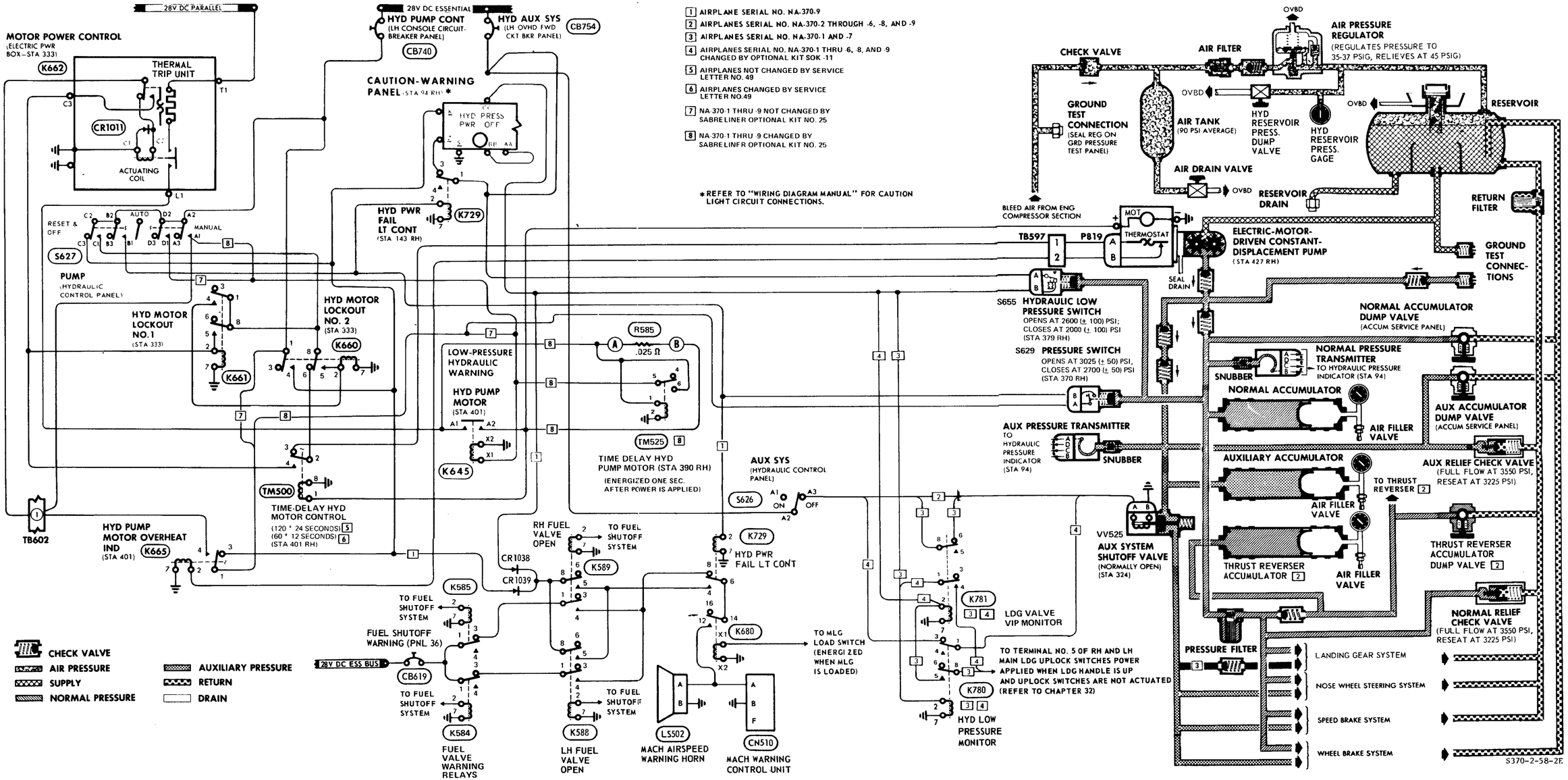
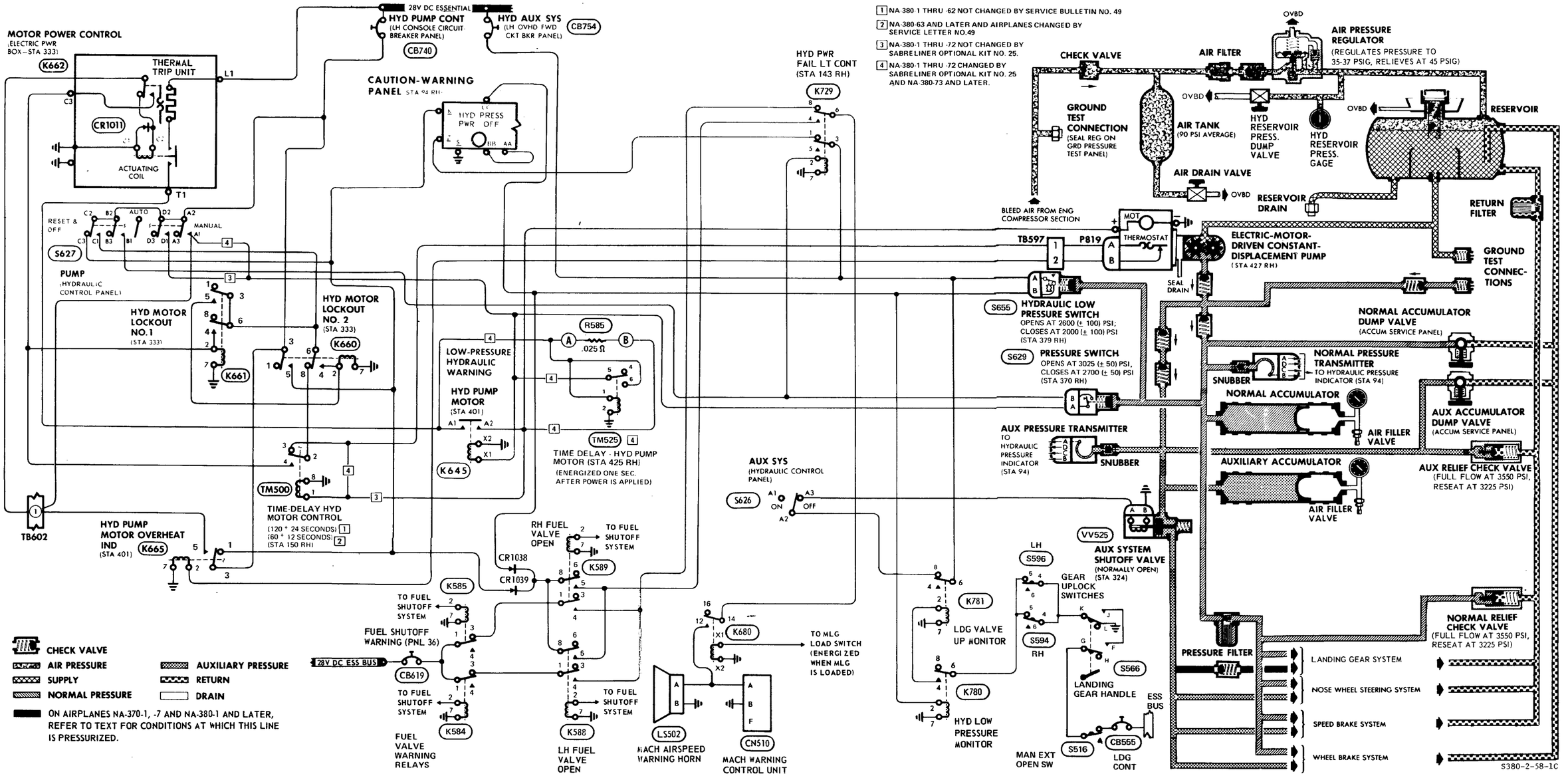


Hydraulic Power System Component Location
Figure 29-00-1


28V DC PARALLEL
28V DC ESSENTIAL
MOTOR POWER CONTROL
(ELECTRIC PWR BOX–STA 333)
K662
THERMAL TRIP UNIT
ACTUATING COIL
CR1011
HYD PUMP CONT
(LH CONSOLE CIRCUIT-BREAKER PANEL)
CB740
HYD AUX SYS
(LH OVHD FWD CKT BKR PANEL)
CB754
CAUTION-WARNING PANEL (STA 94 RH) *
HYD PRESS PWR OFF
HYD PWR FAIL LT CONT (STA 143 RH)
K729
RESET & OFF
AUTO
MANUAL
S627
PUMP (HYDRAULIC CONTROL PANEL)
HYD MOTOR LOCKOUT NO. 1 (STA 333)
K661
HYD MOTOR LOCKOUT NO. 2 (STA 333)
K660
LOW-PRESSURE HYDRAULIC WARNING
HYD PUMP MOTOR (STA 401)
K645
R585
.025 Ω
TM525
TIME DELAY HYD PUMP MOTOR (STA 390 RH)
(ENERGIZED ONE SEC. AFTER POWER IS APPLIED)
TM500
TIME-DELAY HYD MOTOR CONTROL
(120 ± 24 SECONDS) 5
(60 ± 12 SECONDS) 6
(STA 401 RH)
TB602
HYD PUMP MOTOR OVERHEAT IND (STA 401)
K665
AUX SYS (HYDRAULIC CONTROL PANEL)
S626
ON
OFF
RH FUEL VALVE OPEN
TO FUEL SHUTOFF SYSTEM
K589
CR1038
CR1039
K729
HYD PWR FAIL LT CONT
K585
TO FUEL SHUTOFF SYSTEM
FUEL SHUTOFF WARNING (PNL 36)
28V DC ESS BUS
CB619
TO FUEL SHUTOFF SYSTEM
K584
FUEL VALVE WARNING RELAYS
K588
LH FUEL VALVE OPEN
K680
TO MLG LOAD SWITCH (ENERGIZED WHEN MLG IS LOADED)
LS502
MACH AIRSPEED WARNING HORN
CN510
MACH WARNING CONTROL UNIT
K781
LDG VALVE VIP MONITOR
TO TERMINAL NO. 5 OF RH AND LH MAIN LDG UPLOCK SWITCHES POWER APPLIED WHEN LDG HANDLE IS UP AND UPLOCK SWITCHES ARE NOT ACTUATED (REFER TO CHAPTER 32)
K780
HYD LOW PRESSURE MONITOR
VV525
AUX SYSTEM SHUTOFF VALVE (NORMALLY OPEN) (STA 324)
CHECK VALVE
AIR PRESSURE
SUPPLY
NORMAL PRESSURE
AUXILIARY PRESSURE
RETURN
DRAIN
1 AIRPLANE SERIAL NO. NA-370-9
2 AIRPLANES SERIAL NO. NA-370-2 THROUGH -6, -8, AND -9
3 AIRPLANES SERIAL NO. NA-370-1 AND -7
4 AIRPLANES SERIAL NO. NA-370-1 THRU -6, -8, AND -9 CHANGED BY OPTIONAL KIT SOK -11
5 AIRPLANES NOT CHANGED BY SERVICE LETTER NO. 49
6 AIRPLANES CHANGED BY SERVICE LETTER NO.49
7 NA-370-1 THRU -9 NOT CHANGED BY SABRELINER OPTIONAL KIT NO. 25
8 NA-370-1 THRU -9 CHANGED BY SABRELINER OPTIONAL KIT NO. 25
*REFER TO "WIRING DIAGRAM MANUAL" FOR CAUTION LIGHT CIRCUIT CONNECTIONS.
OVBD
AIR PRESSURE REGULATOR (REGULATES PRESSURE TO 35-37 PSIG, RELIEVES AT 45 PSIG)
CHECK VALVE
AIR FILTER
GROUND TEST CONNECTION (SEAL REG ON GRD PRESSURE TEST PANEL)
AIR TANK (90 PSI AVERAGE)
HYD RESERVOIR PRESS. DUMP VALVE
HYD RESERVOIR PRESS. GAGE
RESERVOIR
AIR DRAIN VALVE
RESERVOIR DRAIN
RETURN FILTER
BLEED AIR FROM ENG COMPRESSOR SECTION
MOT
THERMOSTAT
TB597
P819
ELECTRIC-MOTOR-DRIVEN CONSTANT-DISPLACEMENT PUMP (STA 427 RH)
SEAL DRAIN
GROUND TEST CONNECTIONS
S655 HYDRAULIC LOW PRESSURE SWITCH
OPENS AT 2600 (± 100) PSI; CLOSES AT 2000 (± 100) PSI (STA 379 RH)
S629 PRESSURE SWITCH
OPENS AT 3025 (± 50) PSI, CLOSES AT 2700 (± 50) PSI (STA 370 RH)
NORMAL ACCUMULATOR DUMP VALVE (ACCUM SERVICE PANEL)
NORMAL PRESSURE TRANSMITTER TO HYDRAULIC PRESSURE INDICATOR (STA 94)
SNUBBER
NORMAL ACCUMULATOR
AIR FILLER VALVE
AUX ACCUMULATOR DUMP VALVE (ACCUM SERVICE PANEL)
AUX PRESSURE TRANSMITTER TO HYDRAULIC PRESSURE INDICATOR (STA 94)
SNUBBER
AUXILIARY ACCUMULATOR
AUX RELIEF CHECK VALVE (FULL FLOW AT 3550 PSI, RESEAT AT 3225 PSI)
TO THRUST REVERSER 2
AIR FILLER VALVE
THRUST REVERSER ACCUMULATOR 2
AIR FILLER VALVE
THRUST REVERSER ACCUMULATOR DUMP VALVE 2
NORMAL RELIEF CHECK VALVE (FULL FLOW AT 3550 PSI, RESEAT AT 3225 PSI)
PRESSURE FILTER
LANDING GEAR SYSTEM
NOSE WHEEL STEERING SYSTEM
SPEED BRAKE SYSTEM
WHEEL BRAKE SYSTEM
S370-2-58-2F

Hydraulic Power System Schematic
NA-370 Airplanes
Figure 29-00-2



Hydraulic Power System Schematic
NA-380 Airplanes
Figure 29-00-2A

HYDRAULIC POWER MAIN - DESCRIPTION AND OPERATION

1. Normal Hydraulic Pressure System

The hydraulic power system reservoir holds fluid supply at 35 psi for the electric-motor-driven hydraulic pump. When the airplane bus system is energized, and the "PUMP" switch is in ON position, fluid is drawn from the reservoir into the inlet of the pump. The pump pressurizes the fluid and routes it to the pressure-sensing switch, normal pressure transmitter, normal accumulator, normal accumulator dump valve, and normal relief valve. The pressure transmitter gages the available hydraulic pressure and electrically registers it as pounds per square inch on the cockpit pressure gage. The accumulator stores hydraulic pressure at 3000 psi. It ensures rapid application of pressure when the system demands. It may also be used as a limited source of pressure in case of pump failure or when the pump is not operating. The pressure relief valve relieves excessive pressure by bleeding it into the hydraulic power system return line and back to the reservoir. The fluid is then filtered before it enters the subsystems. Return fluid from the subsystems is routed through a common return line and the system return filter before entering the reservoir. The wheel brake system uses an independent return line for fluid to the reservoir.

2. Pump Electrical Control System

Operation of the electric-motor-driven hydraulic pump is controlled by a pressure-sensing switch in the hydraulic power system. When the system pressure is below 2700 (±50) psi, contacts in the pressure-sensing switch close and energize the electric motor, which drives the constant-displacement hydraulic pump. The pump continues to operate until a pressure of 3025 (±50) psi is sensed by the pressure-sensing switch. At this time, the switch contacts open and the electric motor is de-energized. The hydraulic pump operates in accord with the demands made on the hydraulic power system. The pump will operate if the "PUMP" switch is positioned at AUTO and hydraulic system pressure is below 2700 (±50) psi, or if the "PUMP" switch is positioned at MANUAL, regardless of hydraulic system pressure. The "MANUAL" position of the switch bypasses the pressure-sensing switch, providing manual control of the pump in case of failure in the automatic control system.

The hydraulic pump motor is protected from overtemperature by two methods. If the pump runs continuously for over 2 minutes (±24 seconds)*, a time-delay energizes and causes the hydraulic pump motor overheat indicator and motor control relays to de-energize, removing power from the pump motor. The motor control relay may be reset by placing the "PUMP" switch at OFF & RESET, and then back to ON or AUTO. If the pump motor develops an overtemperature condition, the hydraulic pump motor overheat relay will shut off the pump. The relay is controlled by a normally closed contact overheat thermoswitch in the hydraulic pump motor.

*1 minute ±12 seconds on airplanes NA380-63 and Subs and airplanes changed by SSL #49.

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If the hydraulic motor draws excessive current (approximately 185 amperes), the remote circuit breaker opens the switch in the motor power control relay. This interrupts power to the hydraulic pump motor, shutting off the pump. At the same time, hydraulic motor lockout relays No. 1 and 2 are energized. This, in turn, energizes the master caution relay, and the hydraulic power fail light comes on. Moving the "PUMP" switch to OFF & RESET resets the motor power control relay. (See figure 29-00-2 or 29-00-2A.)

3. Reservoir Air Pressurization System

The reservoir air pressurization system provides a constant pressure to the reservoir, regardless of altitude, to prevent pump cavitation. The hydraulic pump obtains its fluid supply from the reservoir, which is pressurized to 35 psi by bleed air from each engine compressor section. Bleed air from the engines is stored under pressure (about 90 psi) in an air tank to supplement pressure lost at high altitude. (For flights originating at, or continuing from, fields above 2500-foot elevation, the air storage tank is precharged to 100-120 psi.) Air is then delivered through a filter and an air pressure regulator to the reservoir. A check valve, in the bleed line to the air tank, prevents loss of stored air pressure and ensures proper supply to the reservoir at all altitudes. A drain valve under the air tank manually dump air tank pressure. A "HYD RESERVOIR PRESSURE" indicator and a manual "HYD RESERVOIR PRESS DUMP" valve are installed in the hydraulic system. The indicator provides a positive means of checking that proper reservoir pressure exists. The dump valve provides a means of releasing pressure from the reservoir for servicing or maintenance.

4. Fluid Reservoir

The fluid reservoir, mounted in the fuselage rear section, supplies fluid to the hydraulic pump. Total capacity of the reservoir is 2 US gallons. A fluid-level gage, attached to the forward side of the reservoir, determines fluid quantity. The reservoir also has a pressure-relief type filler cap, a scupper drain, and a strainer. It has ports for pump suction, system return, brake return, pressurizing, and draining.

5. Hydraulic Pump

An electric-motor-driven hydraulic constant displacement pump supplies the hydraulic pressure for operation of the hydraulic subsystems. The pump and motor combination is mounted in the lower fuselage rear section at station 427. The pump has a output of 3000 psi. The fluid supply to the pump is from the hydraulic power system reservoir. The pump electric motor operates on electrical power from the 28-volt dc parallel bus. Operation of the pump is entirely automatic, being controlled by the pressure-sensing switch in the hydraulic power system. The pressure switch is electrically connected into the pump motor control circuit powered by the 28-volt dc essential bus. (See figure 29-00-2 or 29-00-2A.)

6. Air Pressure Regulator

The air pressure regulator is mounted on the left side of the fuselage rear section. It maintains a constant 35 to 37 psig pressure in the reservoir.

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This ensures that fluid is supplied to the hydraulic pump at a positive pressure at all times. It relieves air pressure from the reservoir at 45 psig maximum. If pressure at the vent port becomes 1.0 psig greater than outlet pressure to the reservoir, the regulator will route the vent pressure through the outlet port to the reservoir.

7. Hydraulic Filters

Hydraulic filters are used in the hydraulic power system pressure and return lines. These filters contain paper (cellulose) elements which are to be replaced at a specified inspection period. These filters consist primarily of a case, the element, and a head with inlet and outlet ports and containing a bypass relief valve. System pressure forces the fluid through the pores of the paper element. In passing through the element, solid particles exceeding 10 microns in size are retained on the outer side of the element. Filtered fluid on the inside of the element flows up into the filter head, then continues on through the filter outlet port. The bypass relief valve opens at a differential pressure of about 50 psi. This allows fluid to bypass the element if the element becomes clogged. The filter will not retain foreign matter in the fluid that bypasses the element through the bypass relief valve. For this reason, it is essential that filter elements be replaced at prescribed intervals.

8. Air Filter

A line-type air filter is used in the reservoir pressurization system to prevent foreign matter from entering the air pressure regulator and reservoir. It is a full-flow unit with a 10-micron degree of filtration. The filter is in the upper left side of the fuselage rear section, forward of the air tank. The filter contains a removable element, which normally is replaced at 200-hour inspections.

9. Accumulator Service Panels

The accumulator service panels are reached through access doors on the underside of the fuselage rear section. The units required to service the normal and auxiliary hydraulic accumulators are on the right side of the fuselage, and on NA-370-2 -6, -8, and -9 airplanes, the units required to service the thrust reverser hydraulic accumulators are on the left side of the fuselage. The panels each consist of an air filler valve, an air pressure gage, and manual pressure dump valves. (See figure 29-00-1.)

10. External Test Connection Panel

The external test connection panel is used when the hydraulic power system is tested with a hydraulic test stand. The test connection panel is on the lower left side of the fuselage rear section, below the engine pod. The panel contains two quick-disconnect fittings, one a pressure connection and the other a suction connection.

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11. Hydraulic Pressure Audible Warning System - Airplanes Serial No. NA-370-9 and NA-380-1 and Later

The hydraulic pressure audible warning system utilizes the airspeed warning horn (LS502) to give an audible warning when the pressure in the normal hydraulic system drops below 2000 ±100 psi, or the PUMP (OFF/AUTO/MANUAL) switch (S627) is in the OFF position, or the HYD PUMP CONT circuit breaker (CB740) is disengaged (pulled out), or the hydraulic motor is overheated, or the hydraulic motor control system is in a lockout condition (caused by a hydraulic motor overload trip); provided the airplane is on the ground (main landing gear loaded) and either engine master switch is positioned at ON.

If the PUMP switch (S627) is positioned at either AUTO or MANUAL and the normal hydraulic system falls below 2000 ±100 psi, electrical power is routed from HYD AUX SYS circuit breaker (CB754) through hydraulic low pressure switch (S655) to the HYD PRESS PWR OFF caution-warning light and to isolation diode CR1038. From the diode power is routed through either of the fuel valve open relays to the normally open contact of hydraulic power fail light control relay K729. Since K729 relay is energized (HYD PUMP CONT circuit breaker (CB740) engaged and PUMP switch S627 positioned at either AUTO or MANUAL), power will be routed to the ground electrical control No. 2 relay (K680) (energized when main landing gear are loaded) and to the coil of the mach airspeed warning horn (LS502). The horn will sound until the hydraulic pressure rises above 2600 ±100 psi, or both ENGINE MASTER switches are positioned at OFF, or both main landing gears become unloaded, or the HYD AUX SYS circuit breaker (CB754) is disengaged (pulled out). (Moving PUMP switch (S627) to the OFF position or pulling HYD PUMP CONT circuit breaker (CB740) will not stop the horn while hydraulic pressure is low, since electrical power from FUEL SHUTOFF WARNING circuit breaker (CB619) is then routed through the fuel valve warning and fuel valve open relays, and the normally closed contacts of the de-energized relay K729, to relay K680 and the horn.)

If the PUMP switch (S627) is positioned at either AUTO or MANUAL and either a pump motor overheat of an overload trip condition occurs, electrical power from CB740 is routed through either the hydraulic pump overheat indicator relay K665 or the hydraulic motor lockout relay No. 2 (K660), through diode CR1039, either of the fuel valve open relays, and relays K729 and K680 to the warning horn LS502. For either the overheat or overload condition, moving the PUMP switch (S627) to the OFF position will stop the horn and permit the overheat or overload circuit to reset.

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12. Reduction of Electrical System Voltage Depression During Hydraulic Pump Initial Start - NA-370 and NA-380-1 Through -72 Airplanes Changed by Optional Kit No. 25 and NA-380-73 and Later

On NA-370 and NA-380-1 thru -72 airplanes changed by Optional Kit No. 25 and on NA-380-73 and later airplanes, a voltage depression on the electrical bus system is reduced during initial start of the hydraulic pump. (See figures 29-00-2 or 2A). The reduction of the voltage depression is provided by the use of a .025 ohm, 250 watt resistor and a time delay relay which energizes the hydraulic pump motor relay (K645) after one second of initial application of electrical power to the hydraulic pump.

With the hydraulic system pressure below 2600 psi and the HYD PUMP control switch at AUTO, electrical power is routed from the hydraulic pump control circuit breaker to the coil of the hydraulic pump motor overheat indicator relay (K665). Relay K665 is then energized, closing contacts 1 and 4 on NA-370 airplanes and contacts 3 and 5 on NA-380 airplanes. Electrical power is then routed from the hydraulic pump control switch (S627), contacts B1 and B2, through the closed system pressure switch (S629), through K665 relay contacts 1 and 4 on NA-370 airplanes or K665 relay contacts 3 and 5 on NA-380 airplanes, to the hydraulic pump motor power control relay (K662). Relay K662 is then energized which in turn routes electrical power from the parallel bus through the .025 ohm resistor (R585) to the positive terminal of the hydraulic pump motor. Subsequent to initial application of electrical power to the hydraulic pump motor, the hydraulic pump motor time delay relay TM525 energizes after approximately one second. When relay TM525 energizes, electrical power from relay K662 is routed through contacts 6 and 5 of relay TM525 to the hydraulic pump motor relay (K645). Relay K645 is then energized, which in turn shunts the current limiting resistor (R585) allowing the hydraulic pump motor to draw its rated current directly from the parallel bus. The one second delay allows the hydraulic pump motor to start rotation with limited current which reduces the voltage depression on the electrical bus system.

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HYDRAULIC POWER MAIN - TROUBLE SHOOTING

1. Trouble Shooting Hydraulic Power System

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. SYSTEM WILL NOT BUILD UP PRESSURE.		
(1) Too little or loss of hydraulic fluid.	Check fluid level on reservoir sight glass.	Fill and bleed hydraulic system. CAUTION: IF AIRPLANE HAS RETURNED FROM FLIGHT WITH HYDRAULIC RESERVOIR EMPTY, HYDRAULIC PUMP MAY BE DAMAGED.
(2) Faulty hydraulic pump.	Check reservoir fluid level content. If reservoir is full, start hydraulic pump and attempt to operate speed brake. If pressure does not start to build up and speed brake does not operate, hydraulic pump is faulty.	Replace hydraulic pump.
(3) Faulty pump electrical system.	Perform continuity check.	Repair electrical system or replace faulty component.
B. SYSTEM WILL NOT BUILD UP ENOUGH PRESSURE.		
(1) Faulty relief valve.	With hydraulic power system operating, feel relief valve for hot spots or vibration. Listen for hissing or squealing. If there are noises, valve is relieving below required pressure.	Replace relief valve.
(2) Faulty pressure switch.	If pump stops operating before normal pressure (3025 (±50) psi) is reached, pressure switch is faulty.	Replace pressure switch.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
C. LOSS OF FLUID IN SYSTEM.		
(1) Leakage of subsystem hydraulic components.	Check reservoir fluid level. If reservoir has enough fluid, connect hydraulic test stand, and cycle subsystems. Visually check all hydraulic units for leakage.	Repair or replace leaking hydraulic unit.
D. EXCESSIVE PRESSURE DROP WHEN SUBSYSTEMS ARE OPERATED.		
(1) Normal accumulator improperly serviced.	Check accumulator air pressure.	Service accumulator.
(2) Faulty normal accumulator.	When system is operated, accumulator air gage should show 3000 (±100) psi. When subsystems are operated, gage indication should fluctuate. If gage does not fluctuate, accumulator piston may be sticking. Shut off test stand. Discharge hydraulic pressure from accumulator, then release air from accumulator. If fluid is forced out of air chamber, accumulator is faulty.	Replace faulty accumulator.

1. Servicing - Manual Filling

A. Servicing Hydraulic System - General

The hydraulic power system is filled through the fluid reservoir. The hydraulic power system, including all of the subsystem components, holds about 4.6 US gallons. The reservoir contains 2 US gallons when serviced to the "FULL" mark. Before the system is filled, both accumulators must be serviced to the correct precharge. Fill reservoir as follows:

(1) Through fuselage rear section access, slowly discharge hydraulic pressure from normal and auxiliary hydraulic accumulators.

CAUTION: TOO RAPID DUMPING OF THE ACCUMULATORS MAY EXCEED THE RESERVOIR PRESSURE RELIEF VALVE RATE AND CAUSE POSSIBLE DAMAGE.

(2) Check that accumulators are properly serviced. (Refer to paragraph 3.)

B. Servicing Hydraulic Reservoir

(1) Dump reservoir air pressure as follows:

(a) Discharge air pressure from storage tank by operating drain valve under tank.

(b) Discharge air pressure from reservoir by operating reservoir pressure dump valve until air discharge stops.

(2) Remove reservoir filler cap and pour hydraulic fluid from a clean container into reservoir. Fill reservoir to "FULL" mark on reservoir sight glass.

(3) Replace filler cap on reservoir and position locking tab down.

(4) Precharge of the hydraulic reservoir may be accomplished by either of the following procedures:

(a) Before operating hydraulic pump, connect a source of dry air or nitrogen (120 psi maximum) to "SEAL REG" fitting on air conditioning ground test panel. On NA380-1 thru -38 airplanes not changed by Optional Kit No. 7, remove APU sense line from "T" fitting on back of ground service panel and cap "T" fitting. Apply pressure until reservoir pressure gage indicates at least 35 psi. Disconnect air source and cap fitting. Remove cap from "T" fitting, reinstall APU sense line, and tighten sense line "B" nut.

(b) Before operating hydraulic pump, precharge reservoir by operating one engine at 70% (NA-370-2 thru -6, -8 and -9 airplanes) or 85 95% (NA-370-1 and -7 and NA-380 airplanes) rpm (cabin pressuriza off) for 1/2 minute to supply reservoir air pressure.

NOTE: Use emergency brakes or adequately chock wheels for this procedure.

(5) If hydraulic system has been drained, make operational check-out of hydraulic power system to be sure that all hydraulic lines are filled and that air is bled from system.

2. Servicing - Test Stand Filling

A. The system reservoir can be filled or replenished from a hydraulic test stand connected to the external test connection panel. The test panel is in the lower left side of the fuselage rear section, below the engine pod. Before the system is filled, the normal and auxiliary accumulators must be serviced to the correct precharge.

(1) Through fuselage rear section access, slowly discharge hydraulic pressure from normal and auxiliary hydraulic accumulators by operating dump valves on accumulator service panels.

CAUTION: TOO RAPID DUMPING OF THE ACCUMULATORS MAY EXCEED THE RESERVOIR PRESSURE RELIEF VALVE RATE AND CAUSE POSSIBLE DAMAGE.

(2) Check that accumulators are properly serviced. (Refer to paragraph 3.)

(3) Discharge air pressure from air tank by operating drain valve under tank.

(4) Open hydraulic reservoir pressure dump valve. Allow pressure to bleed off; then remove cap.

(5) With test stand operating at a low flow rate, press system fill button on stand while holding normal accumulator dump valve open.

(6) Watch reservoir sight glass.

(7) When fluid reaches "FULL" mark on sight glass, release test stand fill button and close accumulator dump valve.

NOTE: On initial filling of a dry system, the reservoir may be overfilled. Later filling should be to the normal fill point only, so as not to overfill the reservoir when the hydraulic system is not pressurized.

(8) Replace filler cap on reservoir.

(9) Check that air tank drain and hydraulic reservoir pressure dump valves are closed.

(10) Precharge hydraulic reservoir by using either of the following procedures:

(a) Before operating hydraulic pump, connect a source of dry air or nitrogen (45 psi minimum to 120 psi maximum) to "SEAL REG" fitting on air conditioning ground test panel. On NA380-1 thru -38 airplanes not changed by Optional Kit No. 7, remove APU sense line from "T" fitting on back of ground service panel and cap "T" fitting. Apply pressure until reservoir pressure gage indicates at least 35 psi. Disconnect air source and cap fitting. Remove cap from "T" fitting, reinstall APU sense line, and tighten sense line "B" nut.

(b) Before operating hydraulic pump, precharge reservoir by operating one engine at 70% rpm (NA-370-2 thru -6, -8 and -9 airplanes) or 85 to 95% rpm (NA-370-1, -7 and NA-380 airplanes) for 1/2 minute to supply reservoir air pressure.

NOTE: Use emergency brake or adequately chock wheels for this procedure.

3. Servicing Hydraulic Power System Accumulators

A. The hydraulic power system accumulators and servicing panels are in the aft equipment compartment. The accumulators are serviced through the hinged access doors on the underside of the fuselage rear section. Check that speed brakes are retracted.

(1) With reservoir filler cap closed and locking tab down, hold accumulator dump valves partially open, one at a time, and slowly bleed off all hydraulic pressure.

CAUTION: TOO RAPID DUMPING OF THE ACCUMULATORS MAY EXCEED THE RESERVOIR PRESSURE RELIEF VALVE RATE AND CAUSE POSSIBLE DAMAGE.

(2) Check accumulator air pressure gages. Pressure should be 1500 (±50) psi for normal and auxiliary system accumulators, and 900 (±50) psi for thrust reverser accumulator.

(3) If air servicing is required, attach nitrogen or dry-air supply source chuck to filler valve stem and loosen small hexnut on filler valve of affected accumulator not more than two turns.

(4) Precharge normal and auxiliary system accumulator to 1500 (±50) psi and thrust reverser accumulator to 900 (±50) psi with dry air or nitrogen.

(5) Tighten filler valve hexnut, and open accumulator dump valve. Recheck precharge, and service as required.

(6) Check filler valve hexnut for tightness, and disconnect air supply source.

4. Bleeding Hydraulic Power System

A. The extent of bleeding necessary to return the hydraulic power system to a serviceable condition depends upon the extent to which maintenance has been performed on the hydraulic power system or any of the subsystems. Normally, only the subsystem in which hydraulic units or lines have been replaced or disconnected need be bled. This may be done by operating the subsystem at a normal rated hydraulic flow and pressure for a minimum of five complete cycles. If the hydraulic power system has been completely drained, or hydraulic lines disconnected at several points, the complete system should be bled as follows:

(1) Operate each subsystem at normal rated flow and pressure for a minimum of five complete cycles to purge systems of air. (Air trapped in the subsystems will be returned to the reservoir by the circulating fluid flow.)

(2) Check reservoir glass for correct fluid level.

5. Draining Hydraulic Power System

A. To drain hydraulic power system, proceed as follows:

(1) With reservoir filler cap closed and locking tab down, hold accumulator dump valves partially open, one at a time, and slowly bleed off all hydraulic pressure.

CAUTION: TOO RAPID DUMPING OF THE ACCUMULATORS MAY EXCEED THE RESERVOIR PRESSURE RELIEF VALVE RATE AND CAUSE POSSIBLE DAMAGE.

(2) Discharge air pressure from air storage tank by operating drain valve under tank.

(3) Open hydraulic reservoir pressure dump valve. Allow pressure to bleed off; then remove cap.

(4) Remove cap assembly at end of drain line and allow reservoir to drain. The hydraulic power system can be drained by disconnecting the hydraulic lines at several low points in the system.

6. Flushing Hydraulic Power System

A. Conditions that warrant flushing are as follows:

(1) After replacement of a hydraulic component that has failed or has been replaced at other than recommended overhaul hours.

(2) Indication of a filter bypass as evidenced by filter clogging element.

(3) When a 4-ounce sample of hydraulic fluid taken from a low point (not a filter) shows any visible sediment or shows water exceeding 0.5 percent at the bottom of a transparent vial or tube approximately one inch in diameter after setting for 2 minutes.

(4) If contamination from other sources is found or suspected in the hydraulic system.

NOTE: Before flushing any part of hydraulic system, install clean filter elements in pressure, return, and nose wheel steering system filters.

B. To flush power system after a hydraulic pump change, proceed as follows:

(1) Drain hydraulic reservoir.

(2) Disconnect line from return filter to reservoir and drain the line.

(3) Disconnect line from reservoir to hydraulic pump and drain the line.

(4) Remove reservoir from airplane. Remove strainer assembly and cap from reservoir; then proceed as follows:

(a) Clean strainer assembly and cap.

(b) Pour 2 quarts of solvent in reservoir. Slosh and drain. Check reservoir for cleanness and repeat sloshing procedure as necessary.

(c) Pour one quart of clean hydraulic fluid in reservoir. Slosh and drain.

(d) Reinstall strainer assembly in reservoir.

(5) Reinstall reservoir in airplane.

(6) Install new filter elements.

(7) Connect lines removed in steps (2) and (3), and refill reservoir.

(8) Service hydraulic power system.

(9) Pressurize reservoir; run pump to charge accumulators.

(10) De-energize pump and operate speed brake to deplete both accumulators.

(11) Repeat steps (9) and (10) three times.

(12) Inspect filter elements and repeat flushing procedure if foreign objects are found in filters.

(13) Install new filter elements after completion of flushing.

C. To flush any subsystem after replacement of a component in that system, proceed as follows:

(1) Connect a hydraulic test stand having 10-micron filtration to ground test connections.

(2) Operate subsystem a minimum of 10 full cycles.

(3) Install new filter elements after completion of flushing.

D. To flush a contaminated system, proceed as follows:

(1) Drain hydraulic power system according to paragraph 5.

(2) Remove reservoir from airplane. Remove strainer assembly and cap from reservoir; then proceed as follows:

(a) Clean strainer assembly and cap.

(b) Pour 2 quarts of solvent in reservoir. Slosh and drain. Check reservoir for cleanness and repeat sloshing procedure as necessary.

(c) Pour one quart of clean hydraulic fluid in reservoir. Slosh and drain.

(d) Reinstall strainer assembly in reservoir.

(3) Reinstall reservoir in airplane.

(4) Install new filter elements.

(5) Service hydraulic power system.

(6) Connect a hydraulic test stand having 10-micron filtration to ground test connections.

(7) Operate subsystem a minimum of 10 full cycles.

(8) Check filter elements for evidence of hydraulic fluid contamination. If contamination is still present, repeat steps (1) through (7).

(9) Check all fittings and lines for evidence of leakage.

(10) Remove hydraulic test stand and close all access panels.

MAINTENANCE PRACTICES (Continued)

7. REPLACEMENT — HYDRAULIC RESERVOIR PRESSURE REGULATOR AIR FILTER ELEMENT

NOTE

As part of the procedure to replace the hydraulic reservoir pressure regulator air filter element, it will also be necessary to inspect the reservoir air tank for presence of hydraulic fluid.

Procedure

A. Open aft fuselage equipment doors 1 and 51 (See chapter 6).

B. Hold air tank drain valve open (See Figure 29-10-00) to relieve pressure and observe drain outlet for presence of hydraulic fluid.

C. If hydraulic fluid is present:

1. Purge air tank to remove all traces of hydraulic fluid.
2. Remove two check valves (see Figure 29-10-00) and check for proper operation. Replace as required.

D. Disconnect air lines and remove filter housing from airplane.

1. Cut safety wire and disassemble air filter halves.
2. Remove and discard air filter element and O-ring.
3. Inspect filter housing halves for contamination. Clean as required.

E. Install new filter element and O-ring in filter housing.

F. Assemble filter housing halves and safety-wire.

G. Install filter housing in airplane and connect air lines.

H. Reinstall two check valves, if removed. Ensure that flow arrows are in the proper direction.

J. Check complete installation for proper safeties and security.

K. Service hydraulic system as required.

L. Pressurize air tank to 60 psi (regulated dry air or nitrogen) and check system for air leaks.

NOTE

Air gauge reads reservoir air pressure only.

M. Perform operational check-out of hydraulic power system.

N. Close aft fuselage equipment doors 1 and 51. (See chapter 6.)

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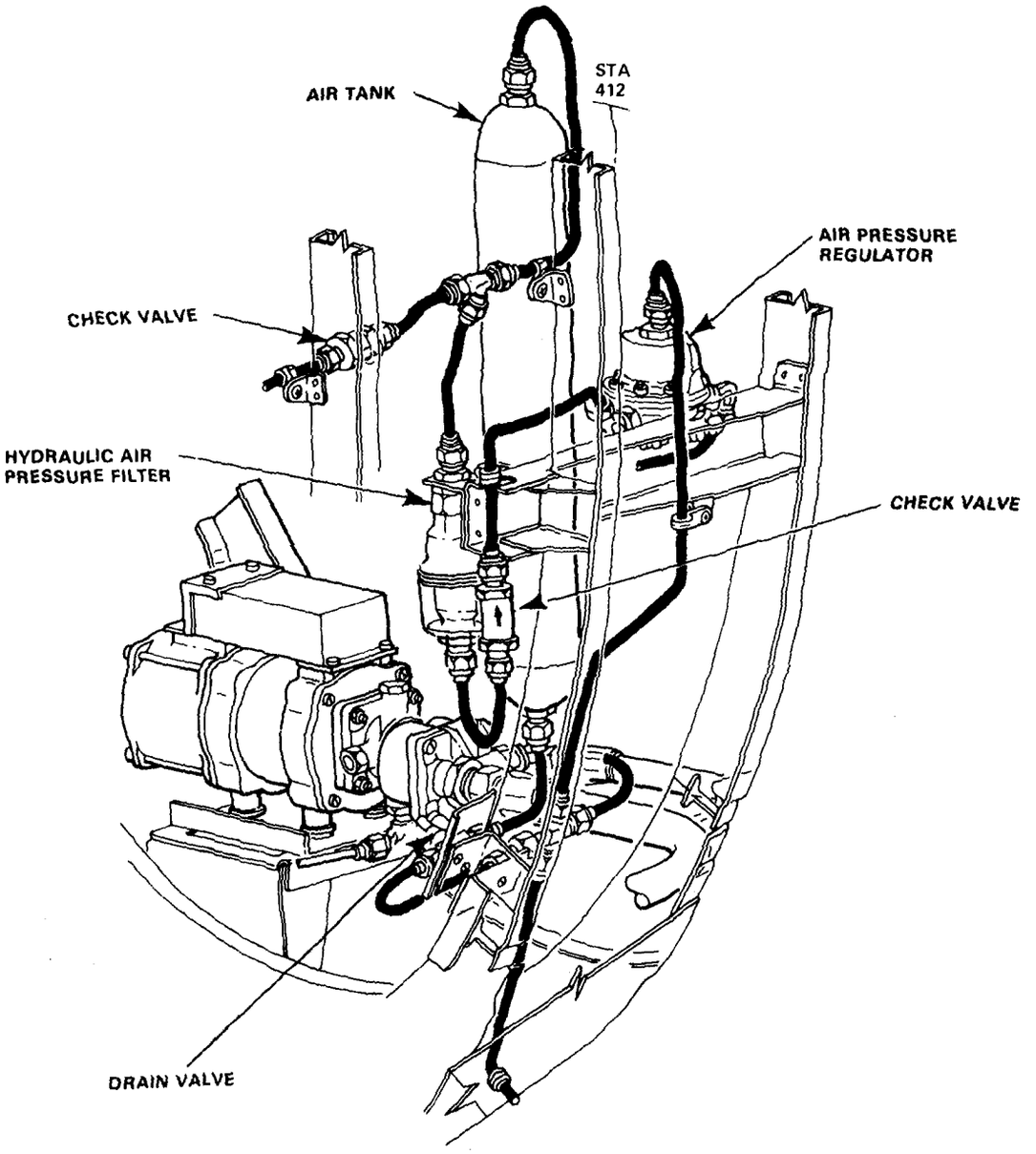

Figure 29-10-00 — Hydraulic Air Reservoir, Air Regulator, Filter and Related Equipment

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1. Removing Hydraulic Filter Elements

NOTES:
- It is strongly recommended when using filter element (P/N AC6091F-8Y82, three micron) that the filter element (P/N AN6235-3A, 10 micron) not be interchanged or used simultaneously.
- Use care in preventing undue exposure of the hyd system to contaminants when adding fluid.
- Removal of the pressure and return hyd filter elements are identical.

A. Open fuselage rear section access when replacing hydraulic return filter element. When replacing hydraulic pressure filter element, open access door 45. (Refer to Chapter 6, figure 6-00-3.)

B. Discharge air pressure from air tank by operating drain valve under tank.

C. Discharge hydraulic pressure from normal and auxiliary hydraulic accumulators.

D. Unscrew and remove filter case.

E. Remove filter elements.

2. Installing Hydraulic Filter Elements

A. Before installing filter element, inspect filter head O-ring and backup rings for evidence of damage. Replace them if necessary. Clean filter case and filter head.

B. Install filter element and filter case. Torque case to 25-35 ft-lbs. Lock using 0.032 safetywire.

C. Clean area of spilled hydraulic fluid.

D. Check fluid level of hydraulic power system reservoir. Service reservoir if needed.

E. Connect hydraulic test stand to airplane hydraulic test panel. Regulate test stand to 4 gpm at 3000 psi.

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G. Operate a hydraulic subsystem to bleed air from hydraulic power system. Check filter for evidence of hydraulic leaks.

H. Shut down and disconnect hydraulic test stand.

I. Check reservoir fluid level. Service reservoir if needed.

J. Install all access doors.

3. Removing and Installing Hydraulic Pump Motor

A. To remove and install hydraulic pump motor, proceed as follows:

(1) Open access doors 1 and 51.

(2) Turn all electrical power OFF.

(3) Disconnect hydraulic pump motor electrical leads.

(4) Remove four nuts securing hydraulic pump to motor and separate pump from motor.

(5) Remove four bolts securing motor mounting bracket to fuselage structure and remove complete assembly.

(6) Remove four bolts securing motor to bracket and remove motor.

(7) Record serial numbers.

(8) Install new motor on mounting bracket with four bolts, washers, bushings, and bonding cable. Lock bolts with safetywire.

(9) Install motor and bracket assembly on fuselage support and secure bonding cable.

(10) Install hydraulic pump on motor and secure with four nuts.

(11) Connect hydraulic pump motor electrical wiring.

(12) Inspect hydraulic pump motor grounding point. (Refer to paragraph 4.)

(13) Check complete installation for security and safeties.

(14) Perform operational check-out of hydraulic power system.

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4. Inspecting Hydraulic Pump Motor Grounding Point

NOTES: • Faulty electrical bonding between hydraulic pump ground strap and fuselage structure can result in failure of hydraulic pump to operate properly or cause damage to fuselage structure due to arcing.

• Improper hydraulic pump operation can result in loss of hydraulic pressure to operate wheel brakes, speed brake, nose wheel steering, and engine thrust reversers on S/N 370-2 thru -6, -8, and -9.

A. Check electrical bonding as follows:

NOTE: Do not use low voltage scales on voltmeter for initial check as a faulty electrical bond may cause a high (up to 28 vdc) voltage between pump ground terminal and fuselage.

(1) Connect positive lead of voltmeter to negative (ground) terminal of hydraulic pump.

(2) Connect negative lead of voltmeter to fuselage structure other than hydraulic pump ground point.

(3) Turn on airplane electrical power and, while observing voltmeter, cycle hydraulic pump. If voltmeter reading exceeds 0.05 vdc, electrical bonding between hydraulic pump ground point and fuselage structure is faulty and must be repaired.

B. Repair hydraulic pump ground point electrical bonding as follows:

(1) Remove hydraulic pump ground strap.

(2) Remove doubler from fuselage structure at hydraulic pump ground point.

CAUTION: Do not use ferrous type abrasives such as emory cloth, crocus cloth, or steel wool to clean bonding surfaces. Imbedded particles from these abrasives may cause subsequent corrosion of electrical bonding area.

(3) Burnish both sides of doubler and mating surface of airframe.

(4) Treat bare metal with alodyne.

(5) Reinstall doubler on airframe using MS20426AD-3 rivets.

(6) Using a clean 0.25 inch diameter steel bolt and aluminum washer (bolt head on opposite side of fuselage structure from doubler) install hydraulic pump ground strap between doubler and hydraulic pump.

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(7) Refinish bonding area with blue-tinted clear lacquer.

(8) Repeat electrical bonding check per step A.

5. Removing and Installing Hydraulic Reservoir Air Filter Element

A. To remove and install air filter element, proceed as follows:

(1) Open access doors. (Refer to chapter 6.)

(2) Hold air tank drain valve open and relieve reservoir air pressure.

(3) Disconnect air lines and remove air filter assembly from airplane.

(4) Cut safetywire and disassemble filter halves.

(5) Remove and replace air filter element (AN6237-1) and O-ring packing (MS28775-222).

(6) Reassemble filter. Secure halves together with safetywire.

(7) Install air filter in airplane and reconnect air lines.

(8) Service hydraulic system as necessary. (Refer to Servicing Hydraulic Reservoir.)

(9) Close access doors.

6. Replacement of Hydraulic Pressure Switches

A. Remove either or both hydraulic pressure switches as follows:

(1) Remove all external electrical power and position BATTERY switches to OFF; or, pull HYD PUMP CONT (left hand console circuit breaker panel) and HYD AUX SYS (left hand overhead fwd circuit breaker panel) circuit breakers and set PUMP switch (hydraulic control panel) to OFF.

(2) Open rear access doors.

CAUTION: Too rapid dumping of accumulator may exceed reservoir pressure relief valve capacity and could cause damage to valve.

(3) With reservoir filler cap closed and locking tab down, discharge normal accumulator pressure by holding NORMAL ACCUMULATOR DUMP VALVE partially open to bleed off all normal system hydraulic fluid pressure.

(4) Cut safetywire and remove elctrical connector from pressure switch being removed. Install dust caps.

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(5) Using two wrenches, disconnect hydraulic line from pressure switch being removed. Install protective caps and plugs.

(6) Loosen clamp and remove pressure switch from airplane.

B. Install either or both hydraulic switches as follows:

(1) Position switch in clamp and, using two wrenches, connect switch to hydraulic line. (The low pressure switch connects to the upper hydraulic line at FS 379 RH. The pump pressure control switch connects to the lower line at FS 370 RH.)

(2) Secure switch in place with clamp and install electriccal connector(s) and safetywire low pressure switch only. (Do not safetywire pump pressure control switch at this time.)

(3) Check fluid level of hydraulic power system reservoir. Service reservoir if needed.

(4) Connect hydraulic test stand to hydraulic test panel. Regulate test stand to 4 gpm at 3000 psi. (If test stand is not available the airplane pump may be used to provide pressure as required.)

(5) Connect external electrical power and push HYD PUMP CONT and HYD AUX SYS circuit breakers (as applicable). If airplane pump is used, set PUMP switch to AUTO.

CAUTION: EXCESSIVE OPERATION MAY CAUSE PUMP TO OVERHEAT. DO NOT OPERATE PUMP FOR MORE THAN TWO MINUTES AT A TIME. ALLOW AT LEAST FIVE MINUTES BETWEEN OPERATIONS FOR PUMP TO COOL.

(6) Operate a hydraulic subsystem repeatedly to bleed air from hydraulic power system. Check pressure switch connections for any evidence of hydraulic leak.

(7) Disconnect hydraulic test stand, if used, or set hydraulic PUMP switch to OFF.

(8) Temporarily disconnect electrical connector from pump pressure control switch. [This prevents pump from operating during step (9).]

(9) Set hydraulic PUMP switch to AUTO, then reduce hydraulic pressure (by operating subsystem such as brakes). Check that HYD PRESS PWR OFF caution light is on at 2000 (±100) psi.

(10) Set hydraulic pump switch to OFF, then install electrical connector on pump pressure switch and safetywire.

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(11) Set hydraulic PUMP switch to AUTO. Check that HYD PRESS PWR OFF caution light goes off at 2600 (±100) psi.

(12) Check that pump operates to build up system pressure and cuts out when pressure reaches 3025 (±50) psi.

(13) Operate a hydraulic subsystem as needed to reduce system pressure. Check that pump cuts in when pressure falls to 2700 (±50) psi.

(14) Set all circuit breakers and switches in their normal or OFF positions and disconnect external electrical power (as applicable).

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1. Operational Check-out of Hydraulic Power System

A. Before an operational check-out of the hydraulic power system, the system must be properly serviced. If system hydraulic units have been replaced or hydraulic lines disconnected, a utility subsystem should be operated a minimum of five cycles to bleed air from the hydraulic power system.

NOTE: An operational check-out of the hydraulic power system must be made if a unit has been replaced, if hydraulic lines have been disconnected, or if an adjustment has been made on any unit. The check-out or checking procedure must be thorough enough to ensure that the system is free of hydraulic leaks and that no malfunction could occur because of improper maintenance.

- Because of allowable indication error in the hydraulic pressure indicating system, the pressure on the gage may not be the actual system pressure. For this reason, the indication error of the hydraulic pressure gage and pressure transmitter should be known, if these instruments are used to determine the serviceability of hydraulic units.

- System pressures specified in this procedure do not include the allowable indication error for the hydraulic pressure indicating system.

B. To perform an operational check-out of hydraulic power system, proceed as follows:

(1) Through fuselage rear section access, discharge hydraulic pressure from hydraulic accumulators by operating dump valves on accumulator service panels.

(2) Check that accumulators are properly serviced.

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(3) Energize hydraulic pump electric motor as follows:

(a) Push in "HYD PUMP CONT" circuit breaker.

(b) Connect external power to airplane.

(c) On airplanes NA-370-9 and NA-380-1 and later, place LH ENGINE-ENGINE MASTER switch at ON. Check that warning horn sounds. Position LH ENGINE-MASTER switch at OFF. Warning horn operation should cease.

(d) Move hydraulic pump switch to AUTO.

(4) Using a stop watch, check pump operating time from start through accumulator pressure build-up to pump shutoff. The time should be approximately 12-1/2 seconds for one accumulator or 25 seconds for all accumulators. The pump should shut off when pressure increases to 3025 (±50) psi.

CAUTION: DO NOT MAKE CONSECUTIVE SUBSYSTEM OPERATIONAL CHECK-OUTS WITH ELECTRIC MOTOR HYDRAULIC PUMP WITHOUT AT LEAST A 5-MINUTE DELAY BETWEEN CHECKS TO COOL MOTOR.

(5) Check that normal accumulator does not bleed down to pump cutin pressure in less than 5 minutes because of system leakage. Pump cutin pressure is 2700 (±50) psi.

(6) Momentarily hold "PUMP" switch at MANUAL. The hydraulic pump motor should run with system pressure above pump cutin pressure (about 2700 psi). Release switch.

(7) Check that auxiliary accumulator has no more than 60 psi drop in 30 minutes. (When checking the auxiliary accumulator, place the "AUX SYS" switch at OFF and turn off the hydraulic pump.)

(8) Turn off hydraulic pump and place all switches at OFF.

NOTE: The following procedure checks the operation of the time delay relays.

(9) Open HYD PUMP CONT circuit breaker.

(10) Disconnect positive electrical lead from hydraulic pump motor and secure from structure.

(11) Close HYD PUMP CONT circuit breaker. Energize the AC inverter system.

(12) Decrease normal hydraulic system pressure to 2400 psi.

(13) Position HYDRAULIC-PUMP switch to AUTO, and check the following indicator lights:

(a) MASTER CAUTION - illuminates in 2 minutes ±24 seconds*

(b) HYD PRESS PWR OFF - illuminates in 2 minutes ±24 seconds*

(14) Position HYDRAULIC-PUMP switch to OFF and return to AUTO.

(15) Slowly decrease normal hydraulic system pressure by applying and releasing brake pedals and check the following indicator lights:

(a) HYD PRESS PWR OFF - illuminates at 2000 ±100 psi.

(b) MASTER CAUTION - illuminates at end of 2 minute(±24 seconds)* time delay period.

(16) Set HYDRAULIC-PUMP switch to OFF. Open HYD PUMP CONT circuit breaker.

(17) Reconnect positive electrical lead to hydraulic pump motor.

(18) Close HYD PUMP CONT circuit breaker.

(19) Position all circuit breakers and switches in their normal or OFF position.

*1 minute ±12 seconds on airplanes NA380-63 and Subs and airplanes changed by SSL #49.

2. Operational Check of Hydraulic Pump Time Delay Relay - NA-370 and NA-380-1 Through -72 Airplanes Changed by Optional Kit No. 25 and NA-380-73 and Later

A. To perform an operational check of the time delay relay, proceed as follows:

(1) Relieve hydraulic pressure from normal and auxiliary systems (and thrust reverser system on NA-370 airplanes) by operating accumulator dump valves.

CAUTION: TO RAPID DUMPING OF ACCUMULATORS MAY EXCEED THE RESERVOIR RELIEF VALVE RATE AND CAUSE POSSIBLE DAMAGE TO THE RESERVOIR COMPONENTS.

(2) Ensure that the hydraulic system is properly serviced. (Refer to Chapter 12).

(3) Close the hydraulic pump control and hydraulic auxiliary system circuit breakers.

(4) Apply external 28 vdc electrical power to the airplane.

(5) Gain access to relay K645, then connect the positive lead of a volt-ohm-meter (Simpson 260 or equivalent) to terminal X1 of hydraulic pump motor relay (K645) and connect negative lead to ground. Set volt-ohm-meter to at least 30 volts dc.

(6) Place hydraulic pump control switch to AUTO or ON position. After a one-second delay verify that volt-ohm-meter indicates 28 vdc.

NOTE: Subsequent to placing the hydraulic pump control switch to AUTO, the hydraulic pump motor relay (K645) will not actuate until the one-second (±10%) time delay has terminated.

(7) If the time limits are exceeded, it may be necessary to adjust the time delay relay (TM525) for a 1.0 (±10%) second delay. To adjust use a small slotted screwdriver to turn the adjustment screw on the end of the relay. Turn adjustment screw clockwise to increase time, or counterclockwise to decrease time.

(8) Check that hydraulic pump motor shuts off at 3050 (±50) psi.

(9) Decrease the normal hydraulic system pressure by repeated application of the wheel brake pedals. Verify that hydraulic pump motor operates when system pressure decreases to 2700 (±50) psi.

(10) Remove volt-ohm-meter test leads from relay and ground, then place the hydraulic pump control switch to the OFF AND RESET position.

(11) Remove electrical power from airplane and install access panel.

3. Inspect and Clean Thrust Reverser Hydraulic Filter

A. Inspect and clean thrust reverser hydraulic filter as follows:

NOTE: Sonic cleaning is the preferred method for cleaning the thrust reverser hydraulic filter. If this method is not available, the filter may be cleaned by dipping and air drying.

(1) Remove all electrical power from the airplane.

(2) Gain access to thrust reverser hydraulic lines through aft fuselage equipment bay doors.

WARNING: To prevent injury to personnel, do not perform maintenance on thrust reverser system unless ground safety locks are installed. Make sure all pressure is discharged.

(3) Install thrust reverser ground safety locks.

(4) Discharge all hydraulic pressure from normal, auxiliary and thrust reverser accumulators.

(5) Dump air pressure on hydraulic reservoir.

(6) Disconnect filter from hydraulic line and check valve. Remove from airplane.

CAUTION: Perform steps (7) thru (11) only in a well ventilated area and wear protective clothing.

(7) Disassemble filter per steps (a) thru (c). (See figure 29-10-1.)

(a) Remove safetywire and filter head from filter body.

(b) Remove filter element from filter body.

(c) Remove O-rings.

(8) Sonic clean per steps (a) thru (e). If sonic cleaning is used omit step 9 (alternate cleaning method).

(a) Place filter element in beaker of trichloroethylene, (technical grade). Locate filter element with open end up. Level of cleaning fluid shall be above open end of filter.

(b) Place beaker in sonic cleaner and operate for five minutes. Cleaning fluid will be dirty.

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(c) Discard dirty cleaning fluid and rinse filter element and beaker with new cleaning fluid using plastic squeeze bottle.

(d) Refill beaker with new cleaning fluid and operate sonic cleaner from an additional five minutes.

(e) Repeat steps (c) and (d) until cleaning fluid remains clean after five minute operation.

(9) Dip clean filter using the following procedure (alternate method):

(a) Dip and agitate filter in solution of trichloroethylene. Repeat process until cleaning fluid remains clean.

CAUTION: Do not dry filter with compressed air or nitrogen as high pressure can damage fine mesh screen.

(10) Air dry filter element.

NOTE: Do not disassemble or use cloth on by-pass assembly in filter.

(11) Clean filter head and filter body cavity with soft, lint free cloth saturated with trichloroethylene and allow to air dry.

(12) Reassemble filter per steps (a) thru (d). (See figure 29-10-1.)

NOTE: Coat all parts with clean hydraulic fluid before assembly.

(a) Install new O-ring on filter element.

(b) Install filter element in filter body.

(c) Install new O-ring and screw filter head on filter body.

(d) Safetywire filter head.

(13) Install hydraulic filter in airplane per steps (a) and (b).

NOTE: Make sure flow arrow on filter is pointing toward check valve.

(a) Install new O-ring and connect filter to check valve.

(b) Connect hydraulic line to filter.

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(14) Service hydraulic system.

(15) Remove thrust reverser ground safety locks.

(16) Perform operational check of thrust reverser. (See chapter 78.)

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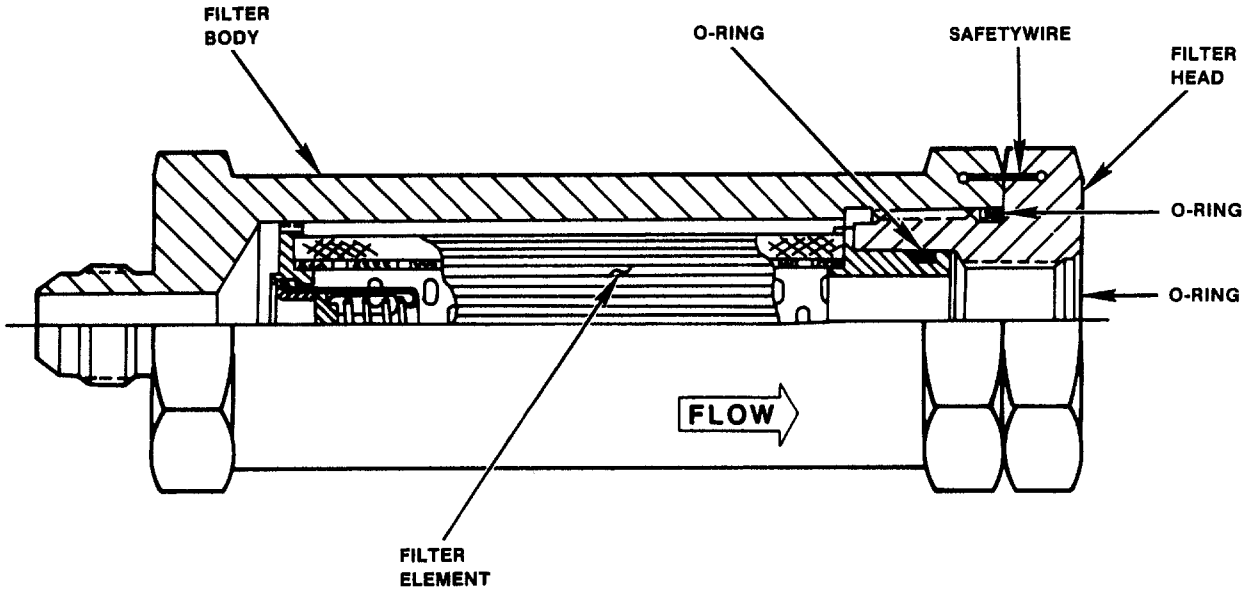

NEW

Thrust Reverser System Filter
Figure 29-10-1

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HYDRAULIC POWER AUXILIARY - DESCRIPTION AND OPERATION

1. Auxiliary Hydraulic Pressure System

Hydraulic pressure is supplied to the auxiliary hydraulic pressure system from the hydraulic power system pump. Fluid under pressure is routed through two check valves to the auxiliary pressure transmitter, auxiliary accumulator, auxiliary accumulator dump valve, and auxiliary relief valve. The fluid pressure, stored at 3000 psi in the auxiliary accumulator, is immediately available for emergency or alternate use. Placing the "AUX SYSTEM" switch, on the hydraulic control panel, in the ON position opens the auxiliary system shutoff valve and releases auxiliary system pressure.

In case of a hydraulic pump failure, both the normal and auxiliary accumulators can operate the essential hydraulic functions. Assuming the normal system accumulator is fully charged, a total fluid volume of about 190 cubic inches would be available.

In case of a normal power system failure, the auxiliary hydraulic pressure system would provide enough fluid under pressure to extend the speed brake and land the airplane with normal braking and nose wheel steering.

2. Auxiliary System Shutoff Valve

The auxiliary system shutoff valve is reached through access (refer to Chapter 6, figure 6-00-3) at the bottom of the fuselage rear section. The shutoff valve is spring-loaded to open when de-energized, and is electrically operated to close when the "AUX SYSTEM" switch, on the hydraulic control panel, is placed in OFF position. This provides a fail-safe feature which releases auxiliary system pressure if the airplane electrical system should fail. When closed, the valve shuts off auxiliary system pressure to the subsystems. Placing the switch in the ON position opens the valve and releases auxiliary system pressure.

On airplanes NA-370-1, -7 and NA-380-1 and later, and airplanes NA-370-2 thru -6, -8 and -9 that are changed by Sabreliner Optional Kit No. 11, the auxiliary system shutoff valve is automatically opened (power removed from the valve) in the event of a hydraulic pump failure at take-off during landing gear retraction (see figure 29-00-2 or 2A). In the event of hydraulic pump failure during take-off and the landing gear system in the retraction cycle when the normal system hydraulic pressure drops to 2000 (±100) psi, electrical power is routed through the hydraulic low pressure switch to the coil of the hydraulic low pressure monitor relay (K780), which is then energized. At the same time, electrical power is routed through the RH and LH landing gear uplock switches (S596 and S594), through the energized closed contacts of relay K780 to the coil of the landing gear valve (up) monitor relay K781. Relay K781 energizes which in turn removes electrical power from the auxiliary system shutoff valve allowing the valve to open which releases the hydraulic pressure stored in the auxiliary accumulator through a check valve (see figure 29-00-2 or 29-00-2A) into the normal landing gear hydraulic system thereby completely retracting the main landing gear. As soon as the RH and LH landing gear uplocks are actuated, electrical power is removed from relay K781, which then de-energizes and applies electrical power to the auxiliary system shutoff valve. The shutoff valve is then closed and traps the remaining auxiliary system pressure in the auxiliary system accumulator.

HYDRAULIC POWER AUXILIARY - TROUBLE SHOOTING

1. Trouble Shooting Auxiliary Hydraulic Power System

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. PUMP OFF - NORMAL ACCUMULATOR DISCHARGED - AUXILIARY SYSTEM WILL NOT OPERATE SUBSYSTEMS.		
(1) Auxiliary accumulator improperly serviced.	Check accumulator air pressure.	Service accumulator.
(2) Faulty auxiliary accumulator.	Place "AUX SYSTEM" switch on hydraulic control panel at ON. When system is operated, accumulator air gage should show 3000 (±100) psi. When subsystems are operated, gage indication should fluctuate. If gage does not fluctuate, accumulator piston may be sticking. Shut off test stand. Discharge hydraulic pressure from accumulator; then release air from accumulator. If fluid is forced out of air chamber, accumulator is faulty.	Replace faulty accumulator.
(3) Faulty auxiliary system shutoff valve.	Pressurize hydraulic power system. Place "AUX SYSTEM" switch at OFF. Shut off test stand or hydraulic pump. Dump normal accumulator pressure. Check that auxiliary pressure maintains 3000 psi. Operate speed brake. Speed brake should not operate, and pressure should not drop. A pressure drop indicates a faulty shutoff valve. Place "AUX SYSTEM" switch at ON. Operate speed brake. Speed brake should operate, and pressure should drop. If speed brake does not operate, shutoff valve is faulty.	Replace faulty shutoff valve.
(4) Faulty auxiliary electrical system.	Perform continuity check.	Repair electrical system.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
B. WHEN NORMAL ACCUMULATOR PRESSURE IS DUMPED, AUXILIARY ACCUMULATOR PRESSURE IS ALSO DUMPED.		
(1) Faulty relief check valve.	Check each relief check valve (speed brake system, wheel brake system, nose wheel steering system) for an open or stuck condition.	Replace faulty relief check valve.

HYDRAULIC POWER AUXILIARY - MAINTENANCE PRACTICES

1. For maintenance practices of the auxiliary hydraulic power system components, refer to 29-10-2.

HYDRAULIC POWER INDICATING - DESCRIPTION AND OPERATION

1. General

The hydraulic power indicating system consists of two hydraulic pressure gages, two hydraulic pressure transmitters, and three hydraulic accumulator air pressure gages. One of the hydraulic pressure gages monitors the output of the electric-motor-driven hydraulic pump; the other gage indicates the pressure in the auxiliary hydraulic power system.

2. Hydraulic Pressure Gages

The hydraulic pressure gages are mounted on the center instrument panel. The gages are mounted on the panel with a clamp fastened to the back of the panel in line with the instrument cutout. The instruments may be installed or removed from the front (face) of the panel. The indicators are helium-filled and hermetically sealed. Each gage is an autosyn-type instrument, electrically connected to the hydraulic pressure transmitter. Each gage provides a visual indication of the pressure being measured by its transmitter. The range of instrument indication is from 0 to 4000 psi hydraulic pressure. The indicating system requires 26-volt, 400-cycle, single-phase alternating current which is received from the instruments power bus. Each instrument is integrally lighted, and lamp requiring 5-volt ac. This current is regulated by a dimming control.

3. Hydraulic Pressure Transmitters

The hydraulic pressure transmitters are shock-mounted in the fuselage aft equipment bay at the top of the fuselage, between stations 390 and 401, and are accessible through the fuselage rear section access doors. Each transmitter is a gas-filled, hermetically sealed unit. It consists of an autosyn unit coupled by mechanical linkage to two Bourdon tubes. One Bourdon tube is connected to the hydraulic pressure system; the other tube is vented to the atmosphere. The pressure differential between the Bourdon tubes is transferred through the mechanical linkage to the rotor shaft of the autosyn unit. Signal voltage induced in the autosyn unit is transmitted to the gage, which shows the pressure in the hydraulic system.

4. Accumulator Air Pressure Gages

The accumulator air pressure gages, in the aft equipment bay are accessible through the fuselage rear section access doors. Each gage is in a Bakelite case which is open to the atmosphere. Each gage mechanism consists of a Bourdon tube and a multiplying mechanism. The multiplying mechanism amplifies the motion of the Bourdon tube and transfers motion to the pointer. The position of the pointer on the dial shows the pressure to which the Bourdon tube (or gage) is subjected.

HYDRAULIC POWER INDICATING - TROUBLE SHOOTING

1. When checking for troubles in external wiring of hydraulic pressure indicating system, pay particular attention to connector plugs at point of soldering. Corrosion, mishandling, and metal fatigue may result in failure at this point. It is best to use an ohmmeter when making continuity checks. A continuity tester does not show high-resistance soldered connections. An ohmmeter, however, shows a higher resistance than is normal for the circuit. The hydraulic pressure indicating system has 2 hydraulic pressure measuring systems and two indicators. Troubles given here could apply to any one system.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
HYDRAULIC PRESSURE INDICATING SYSTEM		
A. HYDRAULIC PRESSURE GAGE DOES NOT INDICATE CORRECT PRESSURE.		
(1) Malfunction in hydraulic pressure indicating system.	Check reservoir fluid level. If reservoir is full, connect a pressure gage of known accuracy to a hydraulic line and quick-disconnect adapter. Open test stand access and connect gage line to hydraulic pressure quick-disconnect. With hydraulic system operating, loosen "B" nut of line to pressure gage to bleed air from gage. Tighten "B" nut. Hydraulic pressure at disconnect panel should be 3000 (±100) psi. If pressure at gage is not within tolerance, trouble is in hydraulic power system. If pressure is within tolerance, trouble can be assumed to be in hydraulic pressure indicating system.	
B. POINTER SLUGGISH.		
(1) No Power on one rotor.	Make continuity check of gage and transmitter "A" and "B" wires.	Replace defective wires.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
(2) Low voltage.	Check voltage output of instrument transformer and instrument inverter.	Supply correct voltage.
(3) Instrument panel vibrator not operating.	Check instrument panel for vibrating with vibrator energized.	Replace vibrator.
C. GAGE POINTER 180 DEGREES IN ERROR.		
(1) No power on one rotor.	Make continuity check of gage and transmitter "A" and "B" wires.	Replace defective wire.
D. GAGE POINTER SWINGS BACK AND FORTH ACROSS DIAL.		
(1) Open transmitter ground lead.	Make continuity check of "A" wire.	Replace defective wire.
(2) Reversed power wires.	Check "A" and "B" wires at gage and transmitter.	Connect wires correctly.
(3) Open gage ground wire.	Make continuity check of "A" wire.	Replace defective wire.
(4) Open stator wire.	Make continuity check of "A," "C," and "D" wires.	Replace defective wire.
E. POINTER SWINGS AND INSTRUMENT SQUEALS.		
(1) Power wire shorted to stator wire.	Check for short between "B," "C," and "D" wires.	Replace defective wires.
(2) Power and stator wires reversed.	Check connections of "B," "C," and "D" wires.	Connect wires correctly.
F. GAGE HOWLS OR SQUEALS.		
(1) Defective gage.	Audible check.	Replace gage.
G. POINTER ROTATES IN REVERSE.		
(1) Reversed stator wires.	Check connection of "A," "C," and "D" wires.	Connect wire correctly.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
H. SLIGHT OR NO MOVEMENT OF POINTER.		
(1) Stator wires shorted.	Check connection of "A," "C," and "D" wires.	Replace defective wires.
(2) No power.	Check fuse and power source.	Repair as required.
I. POINTER SPINS.		
(1) Power and stator wires shorted or reversed.	Check connection and check for shorting of "A," "B," and "D" wires.	Connect wires correctly, or replace defective wire.
(2) Intermittent contact.	Make continuity check.	Replace defective wire.
(3) Defective gage.	Check indicator by checking pressure against test stand. If indicator indicates incorrectly, indicator should be replaced.	Replace gage.
ACCUMULATOR AIR PRESSURE GAGE		
J. GAGE DOES NOT INDICATE.		
(1) Defective gage.	With pressure on gage, make visual inspection of gage. If no reading, gage is defective.	Replace gage.
K. POINTER DOES NOT RETURN TO ZERO WHEN NO PRESSURE IS APPLIED.		
(1) Foreign matter in gage.	Visual inspection.	Replace gage.
(2) Defective gage.	Visual inspection.	Replace gage.
L. GAGE INDICATES INCORRECTLY.		
(1) Defective gage.	Check against allowable tolerance.	Replace gage.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
M. POINTER STICKS.		
(1) Worn or bent movement.	Visual inspection.	Replace gage.
(2) Foreign matter in instrument.	Visual inspection.	Replace gage.
(3) Pointer bent.	Visual inspection.	Replace gage.

HYDRAULIC POWER INDICATING - MAINTENANCE PRACTICES

1. Removing and Installing Hydraulic Pressure Gages

   A. For information concerning removal and installation of hydraulic pressure gages, refer to Chapter 31.

2. Removing Hydraulic Pressure Transmitters

   A. The hydraulic pressure transmitters are in the aft fuselage equipment bay at the top of the fuselage, between stations 390 and 401, and are accessible through the fuselage rear section access doors. The removal procedures for both transmitters are the same. To remove hydraulic pressure transmitters, proceed as follows:

      (1) Discharge hydraulic pressure from normal and auxiliary hydraulic accumulators.

      (2) Disconnect and cap hydraulic line and fitting.

      (3) Cut lockwire and disconnect electrical plug.

      (4) Remove bolts securing transmitter and shock mount assembly to frame and remove transmitter assembly.

      (5) Cut lockwire and loosen and remove shock mount clamp assembly from transmitter.

      (6) Remove pressure snubber from transmitter, and cap.

3. Installing Hydraulic Pressure Transmitters

   A. To install hydraulic pressure transmitters, proceed as follows:

      (1) Install pressure snubber to transmitter, using a new "O" ring. Arrow on snubber should point toward transmitter.

      (2) Install shock mount clamp assembly on transmitter. Tighten clamp assembly and safety with lockwire.

      (3) Place transmitter assembly in position and install attach bolts.

      (4) Connect electrical disconnect plug and safety with lockwire.

      (5) Uncap and connect hydraulic line.

      (6) Connect external utility electrical power to airplane.

      (7) Apply hydraulic pressure to hydraulic power system.

(8) Crack hydraulic line connection at transmitter and bleed air from transmitter. Tighten hydraulic line connection and check transmitter for hydraulic leaks.

(9) Check hydraulic pressure indication on respective hydraulic gage.

(10) Shut down hydraulic pressure, return all switches to off position, and install or close all access doors.

1. Operational Check-out and Inspection of Hydraulic Power Indicating System

A. For operational check-out and inspection of hydraulic pressure instruments, a hydraulic test stand capable of 3000 psi pressure and a flow of 2 gpm is required.

B. Checking Hydraulic Pressure Indicating System

(1) To check hydraulic pressure indicating system, proceed as follows:

NOTE: When any instrument on instrument panel is being checked, the instrument panel vibrators must be operating.

(a) Check gages for security of mounting, cleanness of glass, condition of dial markings, and tightness of electrical plugs or terminals.

(b) Check transmitters for external damage, loose mountings, loose electrical plugs, and deterioration and chafing of interconnecting leads.

(c) Connect external 28-volt dc to airplane and energized inverter system.

(d) Connect external 28-volt dc and 115-volt ac power to airplane.

(e) Regulate output of hydraulic test stand to 650, 1500, 2500, and 3000 psi. The hydraulic pressure gages should read the same as the test stand (±100 psi).

(f) Disconnect hydraulic test stand from airplane.

C. Checking Accumulator Air Pressure Gage

(1) To check accumulator air pressure gage, proceed as follows:

(a) Check gage for security of mounting, cleanness of glass, and condition of dial markings.

(b) Check gage for proper indication of pressure by connecting a test gage of known accuracy to the relative accumulator air filler valve. After connection is made, open accumulator air filler valve and pressurize hydraulic system. Compare gage readings. Tolerance is ±50 psi.

NOTE: This check should be made with the accumulator serviced with air to its normal precharge pressure level and with all accumulator hydraulic pressure discharged.

CHAPTER

30

ICE AND RAIN PROTECTION

CHAPTER 30

ICE AND RAIN PROTECTION

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ICE AND RAIN PROTECTION - DESCRIPTION

1. General

The ice and rain protection systems (figure 30-00-1) consist of electric heating elements, pneumatic deicer boots, installed in the necessary areas. On NA-370-1 and -7 and NA-380-1 and later airplanes, heated bleed air from the engine is used to provide anti-icing air for the fixed engine air intake, removable engine access doors and engine aft fairing as follows:

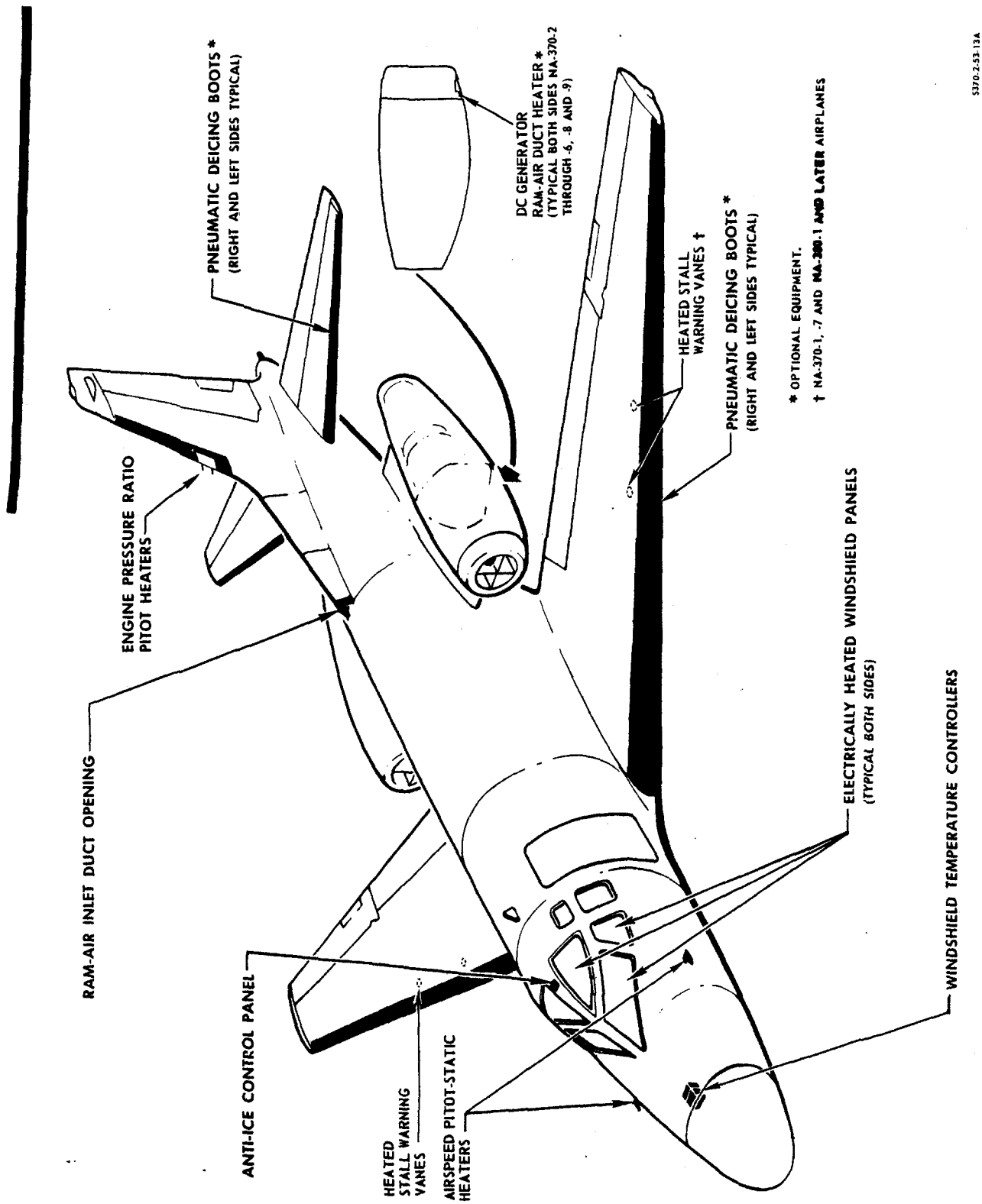
- Pneumatically operated deicer boots for the leading edges of the right and left wings and the vertical and horizontal stabilizers (optional equipment)
- Electrically heated stall warning lift transducer vanes
- Electrically heated environmental system ram-air inlet duct opening
- Electrically heated ram-air duct openings for dc generators (optional equipment on NA-370-2 through -6, -8 and -9)
- Electrically heated windshields
- Battery vent heaters (installed with deicer boots on NA-380 airplanes)

System components and their locations are as follows:

COMPONENT	NO. PER AIRPLANE	LOCATION
Pneumatic deicing system distributor or ejector/distributor valve	2	One at bottom of fuselage, right side, station 195; one at right side of fuselage, near centerline, station 427
Pneumatic pressure switches	3	One at right side of fuselage, near centerline, station 427; two at bottom of fuselage, right side, station 195
Pressure switch (vacuum) (NA-370-2 thru -6, -8 and -9)	1	Bottom of fuselage, right side, station 200
Pneumatic timer	1	Passengers' cabin, right side, station 173
Pneumatic pressure regulator	1	Fuselage rear section, top, centerline, station 372
Pneumatic pressure relief valve	1	Fuselage rear section, top, centerline, station 375
Ejector (NA-370-2 thru -6, -8 and -9)	1	Fuselage rear section, right side, centerline, station 420

COMPONENT	NO. PER AIRPLANE	LOCATION
Vacuum relief valve (NA-370-2 thru -6, -8 and -9)	1	Fuselage rear section, right side, center line, station 412
Windshield temperature controllers	2	Fuselage, forward of cockpit, centerline, station 25
Windshield wiper motor	1	Cockpit, forward of instrument panel
Windshield wiper converter assembly	1	Cockpit, forward of instrument panel
Stall warning lift transducers	2	Underside of right wing (NA-370-2 thru -6, -8 & -9). Both wings on NA-370-1, -7, NA-380-1 and later.

On NA-370-1 and -7, and NA-380-1 and later airplanes, bleed air is supplied to the engine air intake duct, and to the aft fairing, providing anti-icing protection for these areas. Bleed air is also supplied to the three engine access doors through male and female type connectors located on the lower centerline of the engine at the aft ends of the lower doors; and through rubber sealed connectors at approximately the 5 and 7 o'clock positions at the forward ends of the doors. Anti-icing air for the upper door is provided through a male and female connector located at the aft inboard edge of the upper door.



Ice and Rain Protection Systems
Figure 30-00-1

AIRFOIL DEICING PROVISIONS (OPTIONAL EQUIPMENT) - DESCRIPTION AND OPERATION

1. Pneumatic Deicing System

The pneumatic deicing system, an optional equipment item, is used to remove any accumulation of ice on the airfoil surfaces. Pneumatic deicer boots are installed on the wing slats, underslat leading edges, and leading edges of the wing roots and vertical and horizontal stabilizers. Air pressure [18 (±1) psig nominal] inflates the boots in a controlled pressure cycle. When the boots are not pressurized, a suction pressure is applied to keep the airplane airfoil clean and prevent automatic inflation due to the aerodynamic effect on the airfoil sections. (See figure 30-10-1.)

Air to inflate the boots or operate the ejector* or ejector/distributor† valve (providing the necessary vacuum source) is supplied by engine bleed air. The air is extracted from the bleed-air system ducting of both engines, upstream of the engine bleed-air shutoff valve, supplying air for the pneumatic deicing system and the pressurized inflatable seal system. Air is routed from the engine pylons through check valves, and then joined to become a common pressure source. On 380 airplanes, two water separators are installed upstream of each ejector/distributor valve. The separators have a drain line and a disposable filter element.

A fixed flow limiting venturi in the common pressure source controls the flow rate for the deicer system. Airflow from the venturi is directed through a pressure regulator and regulated to 18 (±1) psig. On NA-370 airplanes, a pressure relief valve, downstream from the regulator, relieves at 22.5(±1) psig and reseats at 19.5 (±0.5) psig, to protect the system from overpressurization. On NA-380 airplanes, the pressure relief valve is integral with the regulator and relieves at 20 (±1) psig.

Regulated airflow is divided between two distributing* or ejector/distributor† valves (one for the wing and wing slat, and one for the empennage). The valves are controlled by the timer, allowing a selection of two different timing cycles. Both timing cycles are on (boots pressurized) for three consecutive 5-second intervals (15-second total), the difference being in the time the cycle is off (boots depressurized). The light icing cycle is off for 165 seconds; the heavy icing cycle is off for 45 seconds.

The suction pressure created by the ejector is between 6* and 10* in.(4.5 in.†) Hg at sea level and is dependent on the engine setting. On NA-370 airplanes, vacuum relief valve, a check valve, and a pressure switch are installed downstream from the ejector. The vacuum relief valve is set to relieve at 10 in Hg. The check valve prevents reverse positive pressure to the vacuum system in case of a blocked ejector air outlet. The pressure switch in the vacuum supply ducting will illuminate a light on the caution light panel 25 seconds

*NA-370 Airplanes
†NA-380 Airplanes

after the vacuum drops to 1.5 in. Hg, provided the throttle setting exceeds 95 percent.

To operate deicer boots, place "PNEUMATIC BOOTS" switch to LT or HVY. The pneumatic timer will send signals to the distributor* or ejector/distributor† valves, starting the inflating cycle. The forward valve vacuum port will close and the pressure port will open, pressurizing each outlet port for consecutive 5-second intervals. Then the aft valve vacuum port will close and the pressure inlet port will open, pressurizing the outlet port for 5 seconds. When each valve has completed its inflating cycle, the pressure inlet port is closed, and the air pressure bleeds off through a relieving port in the distributor.

*NA-370 Airplanes
†NA-380 Airplanes

Pressure switches, downstream from the distributor valve outlet ports, are actuated closed when air pressure from a distributor valve inflates its particular section of the deicer boots. These pressure switches, when actuated (15.5 psig), cause the "PNEUMATIC BOOTS OPERATING" lights ("WING," "SLATS," and "EMP") to alternately come on as its particular section is inflated. Each light will go out at the end of its 5-second inflating period, when the pressure drops to 10 psig and deactuates the pressure switch.

When the inflating cycle is completed, air pressure bleeds from the distributor valve through a relieving port. When the pressure has dropped to one psi, the vacuum port opens and completes the deflation of the deicer boots. At the end of 45 or 165 seconds (depending on which icing cycle is selected), the pressurizing cycle starts again.

The deicer boots may be operated manually in case of a failure in the timer. (See figure 30-10-2.) The "PNEUMATIC BOOTS OPERATING" lights contain switches which control manual operation of the system. When one of the lights is pressed, power is routed to the timer lockout relay and to the appropriate distributor valve. The timer lockout relay opens the circuit from the "PNEUMATIC BOOTS" switch to the timer, to isolate the timer from the system. To operate the pneumatic boots manually, each of the "PNEUMATIC BOOTS OPERATING" lights must be depressed consecutively. Timing of the system is controlled manually by the pilot; otherwise, system operation is normal.

2. Distributor* or Ejector/Distributor† Valve

There are two distributor valves, one forward and one aft. Each valve is a solenoid-signaled, pressure-actuated, two-part valve. The valves determine whether the deicer boots will be pressure-inflated or suction-evacuated. Each valve is capable of handling two sets of deicer boots, such as the forward valve, or else one portion of the two-part valve can be blocked to allow the valve to handle just one deicer boot, such as the aft distributor valve.

The valves are controlled by an electrical timer. When the valve solenoids are de-energized (timer off), the valve causes suction to be applied to the deicer boots it controls. When the valve solenoids are energized in sequence by the timer, pressure is applied in the identical timing sequence to the deicer boots. When the inflation period is completed, the air is dumped through an exhaust port in the valve. When the pressure drops to one psig, the valve automatically reapplies suction to the deicer boots.

3. Pneumatic Pressure Regulator

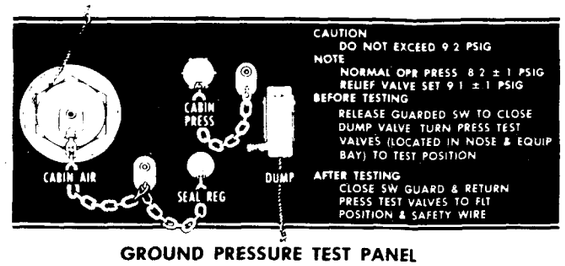
The pneumatic pressure regulator valve is a diaphragm-actuated, differential-pressure type, with a pressure differential of 18 (±1) psig above ambient air. It provides a regulated air pressure supply for the pneumatic deicing system. On NA-380 airplanes, the relief valve function opens at 20 (±1) psig.

4. Pneumatic Timer

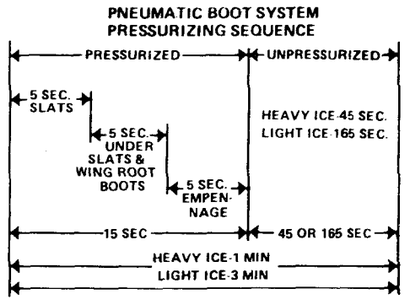
The pneumatic timer electrically controls the timing sequence of the deicing system. It is controlled by the three-positioned (OFF, LT, and HVY) "PNEUMATIC BOOTS" switch. When the switch is positioned at LT or HVY, the sequencing periods of inflation and evacuation for the three sets of deicer boots are automatic. The inflation signals from the timer to the distributor valves

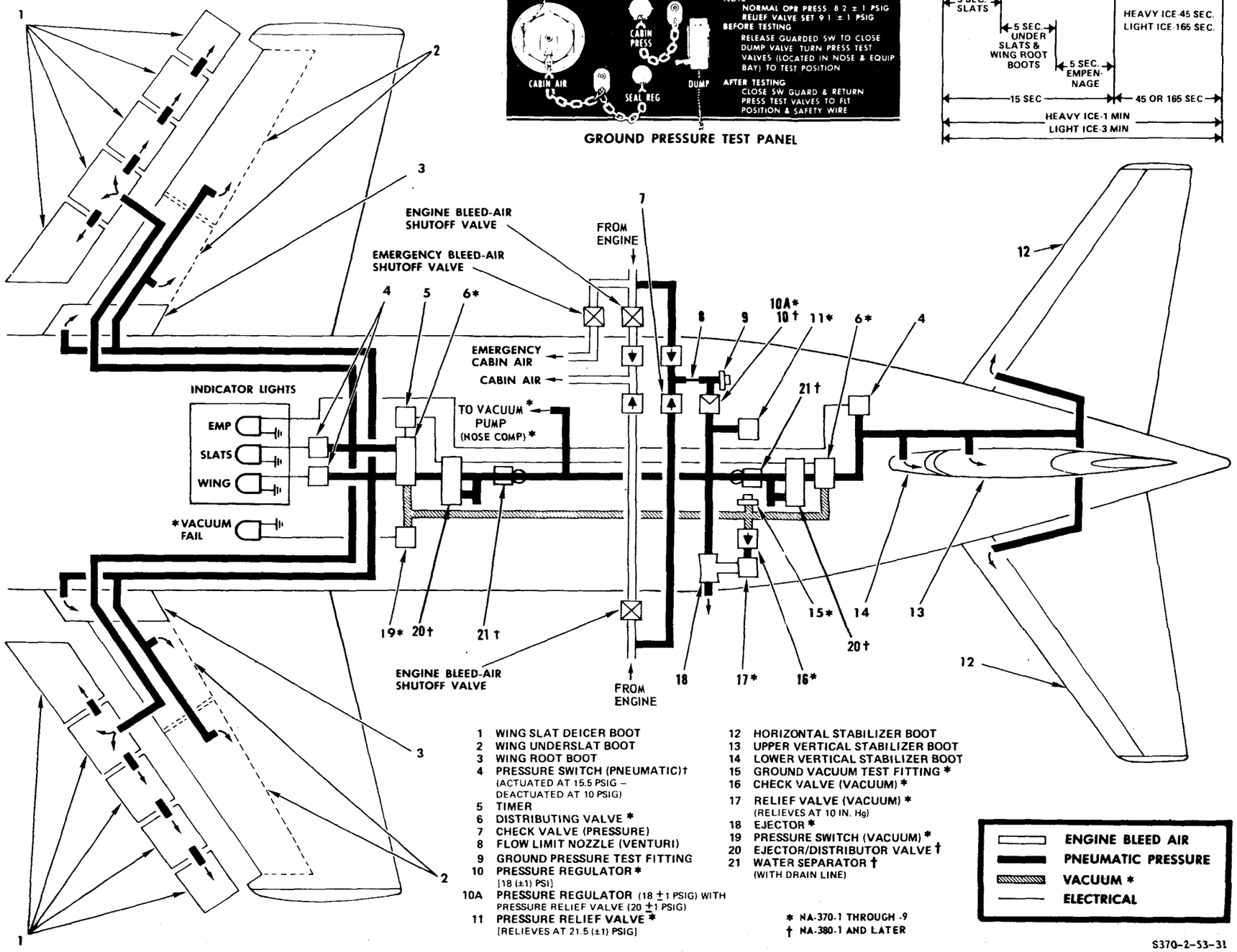
*NA-370 airplanes
†NA-380 airplanes

ORIGINAL
As Received By
ATP


CAUTION
DO NOT EXCEED 9.2 PSIG
NOTE
NORMAL OPR PRESS 8.2 ± 1 PSIG
RELIEF VALVE SET 9.1 ± 1 PSIG
BEFORE TESTING
RELEASE GUARDED SW TO CLOSE DUMP VALVE. TURN PRESS TEST VALVES (LOCATED IN NOSE & EQUIP BAY) TO TEST POSITION
AFTER TESTING
CLOSE SW GUARD & RETURN PRESS TEST VALVES TO FLT POSITION & SAFETY WIRE
CABIN PRESS
CABIN AIR
SEAL REG
DUMP

GROUND PRESSURE TEST PANEL





1 WING SLAT DEICER BOOT
2 WING UNDERSLAT BOOT
3 WING ROOT BOOT
4 PRESSURE SWITCH (PNEUMATIC)† (ACTUATED AT 15.5 PSIG – DEACTUATED AT 10 PSIG)
5 TIMER
6 DISTRIBUTING VALVE *
7 CHECK VALVE (PRESSURE)
8 FLOW LIMIT NOZZLE (VENTURI)
9 GROUND PRESSURE TEST FITTING
10 PRESSURE REGULATOR * [18 (±1) PSI]
10A PRESSURE REGULATOR (18 ±1 PSIG) WITH PRESSURE RELIEF VALVE (20 ±1 PSIG)
11 PRESSURE RELIEF VALVE * [RELIEVES AT 21.5 (±1) PSIG]
12 HORIZONTAL STABILIZER BOOT
13 UPPER VERTICAL STABILIZER BOOT
14 LOWER VERTICAL STABILIZER BOOT
15 GROUND VACUUM TEST FITTING *
16 CHECK VALVE (VACUUM) *
17 RELIEF VALVE (VACUUM) * (RELIEVES AT 10 IN. Hg)
18 EJECTOR *
19 PRESSURE SWITCH (VACUUM) *
20 EJECTOR/DISTRIBUTOR VALVE †
21 WATER SEPARATOR † (WITH DRAIN LINE)

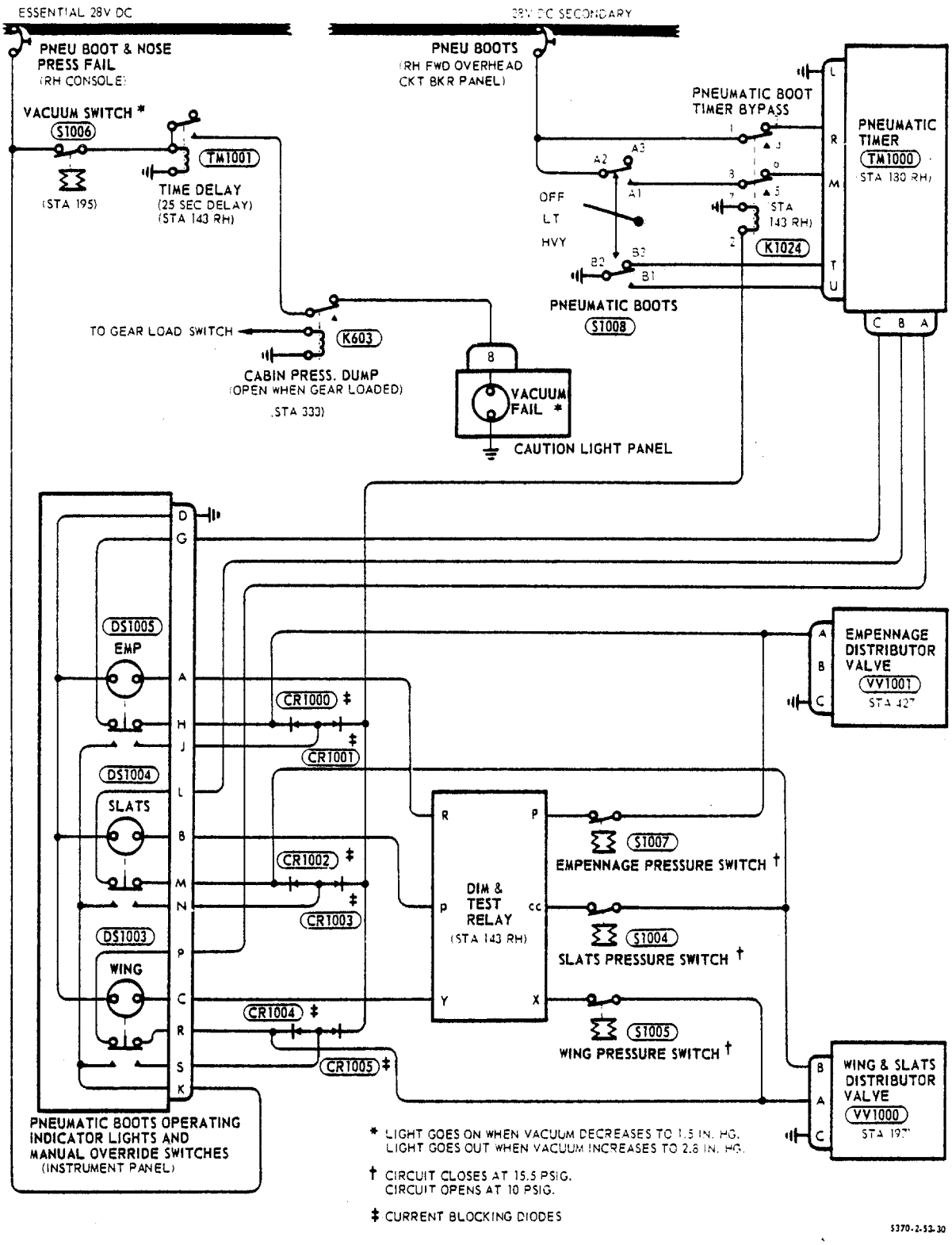
* NA-370-1 THROUGH -9
† NA-380-1 AND LATER

ENGINE BLEED AIR
PNEUMATIC PRESSURE
VACUUM *
ELECTRICAL

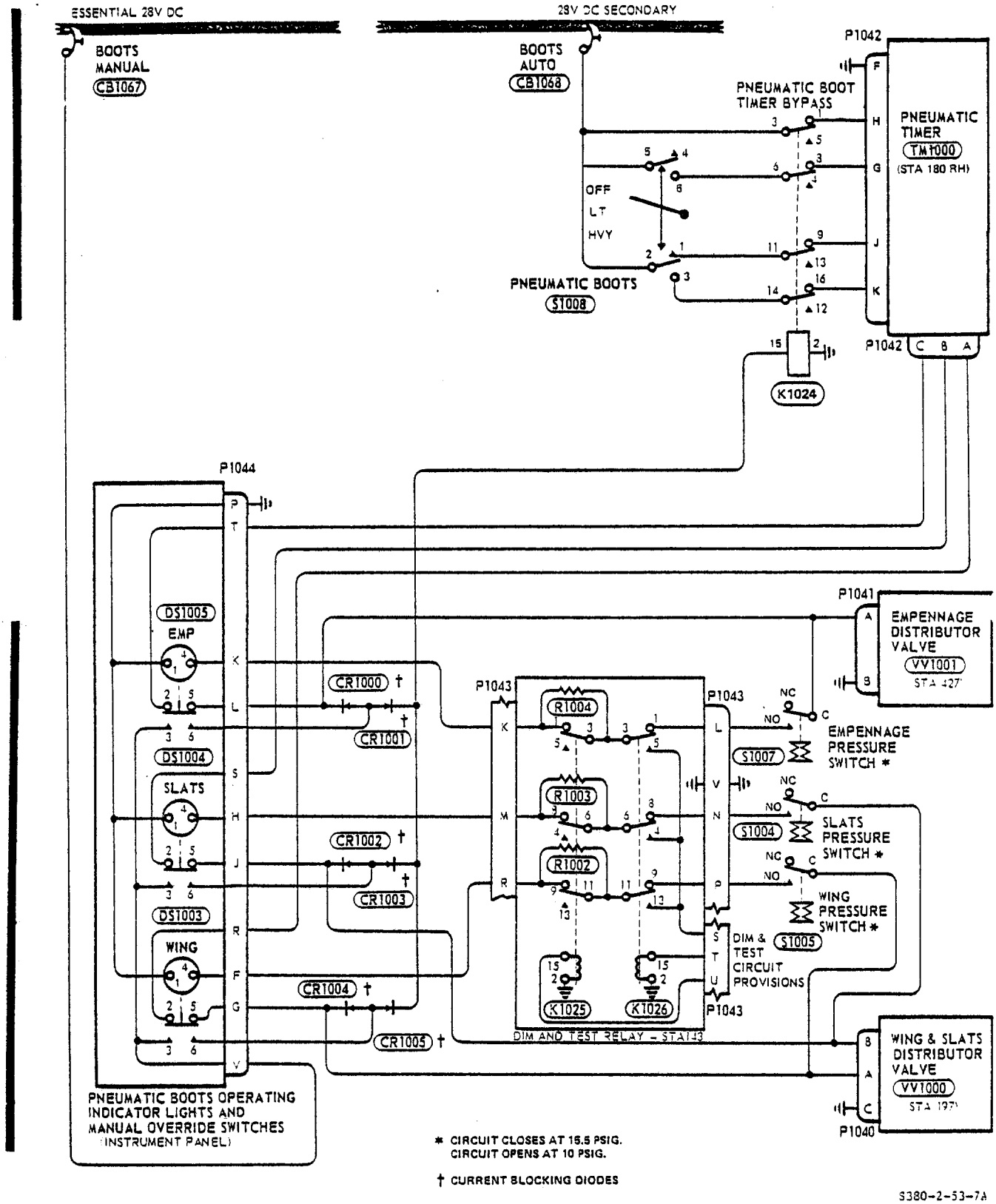
S370-2-53-31

ORIGINAL
As Received By
ATP

Pneumatic Deicing System Schematic
Figure 30-10-1



Deicing System Electrical Schematic -
Optional Equipment NA-370-2 through -6, -8 and -9
Figure 30-10-2



Deicing System Electrical Schematic -
Optional Equipment NA-380-1 and Later
Figure 30-10-3

are sequenced for 5-second intervals, causing the inflation of the deicer boots to be separate, but taking only 15 seconds total time. The LT and HVY positions on the "PNEUMATIC BOOTS" switch indicate the desired period of dwell time (evacuated deicer boots). The dwell time for the light icing cycle is 165 seconds; for the heavy icing cycle, it is 45 seconds. When the switch is actuated to OFF during deicing operations, the timer will automatically continue the cycle in progress and return to the starting position.

5. Pressure Relief Valve - NA-370 Airplanes

The pressure relief valve is actuated by a spring-controlled poppet and provides a secondary means of system pressure control. It functions when the pneumatic pressure regulator malfunctions. The valve relieves at 22.5 (±1) psig and will reseat at 19.5 (±0.5) psig.

6. Ejector - NA-370 Airplanes

The ejector is a jet pump, operating on the venturi principle. Pressurized air (18 psig) flows through the ejector to an overboard port. A port in the ejector is connected to each distributor valve, allowing the vacuum created by the ejector venturi to evacuate air from the deicer boots through ports in the distributor valves. There are no moving parts in the ejector.

7. Vacuum Relief Valve - NA-370 Airplanes

The vacuum relief valve is actuated by a spring-controlled poppet. It will maintain and regulate suction at a maximum value of 10 (±1) in. Hg.

8. Pressure Switch

There are three identical pressure switches in the pneumatic deicing system: one for the wing deicer boots, one for the wing slats deicer boots, and one for the empennage deicer boots. They are diaphragm-actuated and pressure-controlled, differentiating between ambient air and the air pressure within the plumbing to the deicer boots. The switches are actuated to a closed-circuit position when pressure reaches 15.5 psig. When pressure decreases to 10 psig, the switch is actuated to the open position, breaking the circuit.

9. Pressure Switch (Vacuum) - NA-370 Airplanes

The vacuum portion of the pneumatic deicing system contains a pressure switch to indicate a vacuum fail condition to the pilot. It is diaphragm-actuated and pressure-controlled, differentiating between ambient air and a negative pressure within the vacuum system plumbing. The switch is actuated to a closed-circuit condition and lights the "VACUUM FAIL" caution light when the vacuum drops to 1.5 in. Hg. A time-delay relay is used to prevent the light from coming on because of the normal temporary drop of vacuum. The fail light will not come on, on the ground, unless the throttle setting is above 95 percent. This circuit is tied through the landing gear load switches so that the light can come on in the air regardless of throttle position.

AIRFOIL DEICING PROVISIONS (OPTIONAL EQUIPMENT) - TROUBLE SHOOTING (OPTIONAL EQUIPMENT)

1. Trouble Shooting Pneumatic Deicing System

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. DEICER BOOTS DO NOT INFLATE PROPERLY		
(1) Plugged or broken ducting.	Run operational check on deicer boots.	Clear plugged line. Replace damaged duct.
(2) Leak in deicer boot.	Run operational check on deicer boots.	Repair damaged boot. (Refer to B.F. Goodrich Co Report No.70-04-014A or refer to maintenance practices). Replace damaged boot. (Refer to B.F. Goodrich Co Report No.70-04-014A or refer to maintenance practices).
(3) Pressure regulator not operating properly.	Disconnect ducting downstream from regulator and install a pressure gage. Start one engine and check pressure gage for 18 psig.	Repair or replace pressure regulator.
(4) Distributor valve not functioning properly.	Run operational check on deicer boots. (See figure 30-10-2 or 30-10-3.)	Repair or replace distributor valve.
(5) Pneumatic timer does not give proper signal to distributor valve.	Check wiring for open or shorted circuit. (See figure 30-10-2 or 30-10-3.)	Repair or replace damaged wiring. Repair or replace pneumatic timer.
B. "VACUUM FAIL" CAUTION LIGHT COMES ON - NA-370 Airplanes		
(1) Ejector venturi plugged.	Start one engine and check for airflow at outlet of ejector.	Clear or replace ejector.
(2) Vacuum relief valve relieves too low.	Run operational check on deicer boots, using portion of check that tests system vacuum.	Repair or replace vacuum relief valve.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
(3) Vacuum pressure switch set to high.	Run operational check on deicer boots, using portion of check that tests system vacuum.	Adjust or replace vacuum pressure switch.
(4) Check valve downstream of ejector does not open.	Remove check valve and check for proper operation. Make sure that check valve arrow is pointing in direction of flow, toward ejector.	Repair or replace check valve. Properly install check valve.
C. PNEUMATIC BOOTS DO NOT CYCLE PROPERLY.		
(1) Distributing valves do not receive proper electrical signal.	Check wiring for open or shorted circuit. (See figure 30-10-2 or 30-10-3.) Timer does not send proper signal to distributor valve.	Repair or replace damaged wiring. Repair or replace pneumatic timer.
D. ONE OR MORE "PNEUMATIC BOOTS OPERATING" LIGHTS DO NOT LIGHT.		
(1) Light burned out.		Replace light.
(2) Pressure siwtch inoperative.	See figure 30-10-2 or 30-10-3 for circuitry; then substitute pressure switch and put in place of one that will not operate.	Replace or repair defective pressure switch.
E. PNEUMATIC BOOT SYSTEM INOPERATIVE.		
(1) Pneumatic timer not functioning.	Check for shorts or open circuit in wiring. (See figure 30-10-2 or 30-10-3 .)	Repair or replace pneumatic timer.
(2) Timer lockout relay faulty.	Check for open circuit to timer.	Replace relay.

AIRFOIL DEICING PROVISIONS (OPTIONAL EQUIPMENT) - MAINTENANCE PRACTICES

1. Replacement of Deicer Boots-NA-380 Airplanes

The deicer boot removal and installation procedures are similiar on all surfaces. The major difference being the access to the boot connecting points to the aircraft system. Access plates are provided on the skin surfaces adjacent to these connections. On first time replacement of the horizontal stablizer boots, it will be required to remove the horizontal stablizer in accordance with exsisting procedures outlined in Chapter 27. Subsequent boot replacement will not require removal of the horizontal stablizer due to a longer hose (P/N B-118-1 obtained from B.F. Goodrich Company) to be installed during the first required boot replacement.

A. Removing Deicer Boots

WARNING: TO PREVENT A POSSIBLE EXPLOSION AND/OR SERIOUS INJURY TO PERSONNEL, THE AIRPLANE SHOULD BE POSITIONED IN A WELL VENTILATED AREA AWAY FROM ANY SPARKS OR FLAMES. THE CEMENT AND SOLVENTS USED FOR REPLACEMENT ARE FLAMMABLE AND TOXIC.

(1) Position airplane in a ventilated flame free area and remove all electrical power from airplane. Disconnect batteries and remove from airplane. Add an equivalent amount of ballast in place of batteries.

(2) If the tail surface boots are to be replaced, obtain a tail stand and the necessary work stands as required for access to deicer boots (remove horizontal stablizer on first time replacement, refer to Chapter 27). Position tail stand under fuselage.

(3) Mask the peripbery of the deicer boot to be replaced. Allow from 1/4 to 3/8" past the edges of the boot. The use of wax paper is recommended to prevent the solvent from contacting any painted surfaces. Mask painted surfaces as required.

NOTE: MEK or toluol may be used to loosen the adhesion of the boot to the skin surface. Toluol should not be used around any plastic material. A clean pump type oil can may be used to control the amount of solvent to be applied to the adhesion surface.

(4) Starting at one corner of the upper trailing edge of the boot, apply solvent sparingly to the seam line while tension is applied to peel back the corner of the boot.

(5) Separate the boot from the wing for a distance of 4 inches all along the upper trailing edge. On horizontal surfaces, this 4 inch area will act as a reservoir for the solvent, thus aiding in the removal process. Remove boot from surface.

NOTE: If the boot is to be scraped, and to aid in the removal procedure, it may be cut and removed in sections parallel to the inflation tubes. If this method is used, ensure that the skin surface is not scratched when cutting the boot.

B. Installing Deicer Boots

NOTE: Proper adhesion of the deicer boot is dependent on the preparation of the skin surface prior to installation; therefore, the following procedures should be followed. The ambient temperature should be between 40°and 110°F during the installation procedures.

WARNING: TO PREVENT A POSSIBLE EXPLOSION AND/OR SERIOUS INJURY TO PERSONNEL, THE AIRPLANE SHOULD BE POSITIONED IN A WELL VENTILATED AREA AWAY FROM ANY SPARKS OR FLAMES. THE CEMENT AND SOLVENTS USED FOR REPLACEMENT ARE FLAMMABLE AND TOXIC.

(1) Clean the dry cement from the skin surface using Kelite A-727-B (Kelite Products, 77 Industrial Rd., Berkeley Heights, N. J.) Ensure that the surfaces are free of paint and primer.

(2) Mask any area that may have come loose during the removal procedure.

(3) Clean the skin surface thoroughly with MEK or acetone at least two times. During final cleaning, remove the solvent film with a clean lint free cloth prior to drying of the solvent. Change cloth frequently to prevent re-contamination.

(4) Fill gaps of skin splices with sealing compound EC-801.

(5) Remove deicer from container and lay out on a protected surface. Using a clean lint free cloth moistened with MEK or acetone, clean the rough back surface of the boot at least two times. Change cloth frequently to avoid re-contamination of the surface.

(6) Throughly mix the 1300L* cement before application.

(7) Apply one even brush coat to the back surface of the boot and to the skin surface. Allow to air dry for a minimum of one hour. Apply a second coat of cement to both surfaces. Allow to air dry for a minimum of one hour.

*3 M Company
Federal Stock #8040-628-4199

NOTE: Longer drying time of the cement coats may be required as the humidity increases.

- The boot and leading edge may be cemented for a maximum of 48 hours before actual installation, providing the cemented surfaces are covered and kept clean. Avoid touching cemented surfaces with hands.

(8) Locate and mark (chalk mark or equivalent) the center line of the leading edge. This mark is used to align the boot during installation. Intensify the white reference line on the boot if required.

(9) Obtain sufficient personnel to hold boot adjacent to surface. Place connecting hoses in their respective openings (connect and tighten clamps if desired).

(10) Align the boot centerline to coincide with the leading edge centerline made in step (8).

(11) Using a clean lint-free cloth moistened with toluol, reactivate cement approximately three inches around connections in leading edge. When cement dries to a tacky state, press boot against surface; then use a rubber roller, roll boot to leading edge in the tacky areas.
CAUTION: AVIOD SOAKING OR RUBBING OF CEMENT TO PREVENT REMOVAL FROM SURFACE.

(12) Continue installation by reactivating the cement along the centerline leading edge surface and boot in spanwise strips approximately six inches wide. Roll boot (parallel to inflatable tubes) firmly against skin, being careful not to trap any air under boot.

NOTE: When installing a boot six feet or longer, the cement may not remain tacky to roll down the entire length of the six inch width, therefore, roll boot down part way leaving an open angle to permit access for reactivating cement.

- If boot becomes misaligned during installation, reactivate cement and re-align as required.

(13) Subsequent to boot installation, trim ends and trailing edges as required.

(14) Remove cement from trailing edges and boot edges. Ensure that no solvent enters the cemented areas.

(15) Mask the trimmed ends or gaps and fill with EC801 sealing compound. Feather edges as required.

(16) Remove tape from areas to be sealed with EC801 and clean with MEK.

(17) Mask off boot 1/4 inch forward of trailing edge around the periphery of the boot.

(18) Apply two brush coats (10 to 15 minutes between coats) of A-56-B to surface between the taped areas (step (17) and step (3) of removal procedures) and to the sealed seams. Ensure that conductive coating (A-56-B) is continuous from the boot to the painted surface. Remove all tape as soon as the second coat is applied (before coating dries).

(19) Connect and tighten all hose connections. Install access doors and clean areas of any excess cement as required. Touch up painted surfaces if required.

CAUTION: THE AIRPLANE SHOULD NOT BE FLOWN SUBSEQUENT TO BOOT INSTALLATION UNTIL A 12 HOUR PERIOD HAS ELASPED.
- DO NOT OPERATE BOOTS FOR 48 HOURS.

(20) Remove ballast and work stands and install batteries. Close access doors.

(21) Subsequent to 48 hours, perform an operational check-out of deicer boot system.

(22) Check complete installation for security of attachment and cleanliness.

B. Repair of Neoprene Dicer Boots

A cold patch repair kit (B. F. Goodrich Co. P/N 74-451C) is available for repairs on neoprene boots. Instructions within the kit are provided for installing the cold patches. To refurbish a neoprene deicer boot, proceed as follows:

WARNING: TO PREVENT A POSSIBLE EXPLOSION AND/OR SERIOUS INJURY TO PERSONNEL, THE AIRPLANE SHOULD BE POSITIONED IN A WELL VENTILATED AREA AWAY FROM ANY SPARKS OR FLAMES. THE CEMENT AND SOLVENTS USED FOR REFURBISHING ARE FLAMMABLE AND TOXIC.

(1) Position airplane in a suitable and warm atmosphere.

(2) Remove all electrical power from airplane and remove batteries. Add an equivalent amount of ballast in place of batteries.

(3) Position tail stand under airplane and work stand around empanage surfaces.

(4) Mask painted surfaces 1/4" to 3/8" aft of boot trailing edge.

(5) Clean boot surface with esopropyl acetate.

(6) Rough boot surface with 400 grit sandpaper. Sand only enough to rough surface without removing excess material.

(7) Clean boot surface with a clean lint free cloth moistened with MEK or acetone at least two times. Change cloth frequently to avoid re-contamination of boot surface.

(8) Apply one brush coat of A-56-B cement and allow to dry for at least one hour. Apply a second coat of A-56-B cement and allow 4 hours before operating boots.

(9) Remove tape from surfaces and clean painted surfaces as required. Remove tail and work stands as required.

(10) Remove ballast from battery trays and install batteries. Install access doors.

2. Replacement of Distributor* Valves

The forward distributor valve is located at RHFS198. The aft distributor valve is installed at FS412. Removal and installation procedures given are typical for both distributor valves.

A. Removing Distributor Valves

(1) Remove power from 28-volt dc essential bus.

(2) Remove or open access door as required.

(3) Loosen all clamps on both ends of hoses attached to valve. Slide hoses back on lines.

(4) Cut lockwire on the four attach bolts; then remove attach bolts.

(5) Pull valve clear until electrical plug is accessible.

(6) Cut lockwire on electrical plug; then disconnect plug. Cover plug and receptacle.

(7) Plug or seal all valve openings and line openings.

B. Installing Distributor Valves

(1) Remove all coverings or plugs from valve openings, line openings, electrical plug, and electrical receptacle.

(2) Install electrical plug in valve receptacle and safety with lockwire.

(3) Make sure all hose clamps are positioned on attaching hoses and lines. Position hoses on valve; then install four attach bolts.

*Ejector/Distributor valve on NA380 Airplanes

(4) Safety attach bolts with lockwire; then install and tighten all hose clamps.

(5) Check out system if necessary. (Refer to 30-10-3.)

(6) Install or close access door as required.

3. Replacement of Pneumatic Pressure Regulator

A. Removing Pneumatic Pressure Regulator

(1) Remove access covers. (Refer to Chapter 6, figure 6-00-3 or 6-00-4 as applicable.) The regulator is mounted forward of the access doors, in the top of the bay at about fuselage station 379.

(2) Remove two air lines from regulator.

(3) Cut lockwire from regulator mounting bolts; then remove four mounting bolts.

(4) Remove regulator; then cover or plug all openings.

B. Installing Pneumatic Pressure Regulator

(1) Remove covers or plugs from all openings.

(2) Position regulator on mounting brackets and install four mounting bolts. Safety bolts with lockwire.

(3) Install air lines on regulator; then torque "B" nuts.

(4) Install access doors.

4. Replacement of Pneumatic Timer

A. Removing Pneumatic Timer

(1) Remove power from 28-volt dc essential bus.

(2) Remove pneumatic timer, gain access to rack under the bottom shelf of the electrical equipment rack at RHFS150.

(3) Disconnect electrical plug and remove timer from shelf.

(4) Cover plug and timer receptacle.

B. Installing Pneumatic Timer

(1) Uncover plug and timer receptacle.

(2) Install electrical disconnect in timer electrical receptacle.

(3) Perform operational checkout of deicer boot system.

(4) Check complete installation for security and safties.

(5) Install trim panels on equipment racks.

5. Replacement of Pressure Relief Valve - NA-370-2 through -6, -8 and -9

A. Removing Pressure Relief Valve

(1) Remove access covers. (Refer to Chapter 6, figure 6-00-3.) The relief valve is aft of the pressure regulator, and is mounted forward of the access doors, in the top of the bay at about fuselage station 365.

(2) Remove one air line from relief valve.

(3) Remove two mounting bolts; then remove relief valve.

(4) Cover or plug openings.

B. Installing Pressure Relief Valve

(1) Remove covers or plugs from openings.

(2) Position relief valve on mounting bracket; then install two mounting bolts.

(3) Install air line "B" nut, and torque.

(4) Install access doors.

6. Replacement of Ejector - NA-370 Airplanes

A. Removing Ejector

(1) Remove access covers. (Refer to Chapter 6, figure 6-00-3 or 6-00-4 as applicable.) The ejector is mounted on the right side of the bay, aft of the relay compartment.

(2) Disconnect three air lines from ejector. Remove ejector; then plug or cover all openings.

B. Installing Ejector

(1) Remove plugs or covers from all openings.

(2) Position ejector between three air lines, and start "B" nuts. Tighten and torque "B" nuts.

(3) Install access doors.

7. Replacement of Vacuum Relief Valve - NA-370 Airplanes

A. Removing Vacuum Relief Valve

(1) Remove access covers. (Refer to Chapter 6, figure 6-00-3 or 6-00-4 as applicable.) The vacuum relief valve is aft of the ejector, and is on the right side, above and aft of the hydraulic reservoir.

(2) Disconnect one air line; then remove vacuum relief valve.

(3) Cover or plug openings.

B. Installing Vacuum Relief Valve

(1) Remove cover or plugs from openings.

(2) Position vacuum relief valve on air line; then tighten "B" nut. Torque "B" nut.

(3) Close access doors.

8. Replacement of Pressure Switches

The wing or wing slat pressure switches and the vacuum pressure switch are in access 24. (Refer to Chapter 6, figure 6-00-3 or 6-00-4 as applicable.) The empennage pressure switch is above and aft of the hydraulic reservoir on the right side of the fuselage rear section. Removal and installation procedures given are typical for all pressure switches.

A. Removing Pressure Switches

(1) Remove power from 28-volt dc essential bus.

(2) Remove or open access as required.

(3) Unsolder switch electrical connections. Identify wires.

(4) Loosen switch from fitting, using two wrenches: one to hold fitting, and the other to loosen switch.

(5) Remove switch; then plug or cover all openings.

B. Installing Pressure Switches

(1) Remove plug or covering from valve and line opening.

(2) Check that MS28778-3 gasket on pressure switch has not been damaged; then install switch in union. Use two wrenches to tighten switch.

(3) Solder electrical connections to switch.

(4) Install or close access as required.

9. Replacement of Deicer Boot Water Separator and Filter

One filter is located at FS401, and the other is located under the access panel aft of the speed brake (RH side).

A. To replace the filter element of the deicer boot water separator and filter, proceed as follows:

(1) Disconnect drain line from bottom of filter housing.

(2) Remove lockwire between filter drain nipple and filter housing retaining bolt.

(3) Unscrew bolt from bottom of filter housing and remove filter housing assembly from filter manifold.

(4) Remove cotter pin from upper portion of retaining bolt and remove filter element retainer and filter element. Discard filter element and "o" ring.

(5) Remove and discard "o" rings from filter manifold.

(6) Thoroughly clean and dry interior and exterior of filter housing assembly.

(7) Install new "o" rings in filter manifold. Lubricate "o" rings with Dow-Corning lubricant DC-4 (or equivalent) before assembly.

(8) Position new filter element (with new "o" ring installed) over retaining bolt in filter housing.

(9) Place filter retainer on retaining bolt and secure in place with new cotter pin.

(10) Install filter housing assembly in filter manifold and tighten retaining bolt finger tight.

(11) Connect drain line to nipple on bottom of filter housing.

(12) Torque filter housing retaining bolt to 9-11 inch-pounds and safety to filter drain nipple with lockwire.

10. Application of Sealer to De-icer Boots P/N 25S-7D5017-() and 26S-705017-() (Hydrin boots only)

A. This procedure is required to extend the service of the Hydrin de-icer boots. Hydrin De-icer boot repair kit (BFG 74-451H) is available for boot repairs. To apply sealer to de-icer boots, proceed as follows:

CAUTION: TO PREVENT DAMAGE TO THE EXTERIOR PAINTED SURFACES, DO NOT WIPE OR ALLOW THE MEK TO COME IN CONTACT WITH THESE SURFACES.

(1) Wash the entire boot surface with methyl ethyl ketone (MEK).

NOTE: To avoid recontamination of dirt and etc. when cleaning the boots, ensure that the wiping cloth is changed often.

(2) Subsequent to cleaning the boot surfaces with MEK, wipe complete surfaces with a clean dry, lint-free cloth.

(3) Obtain cement (sealer)* and accelerator* as required and mix as follows:

(a) Obtain a suitable clean container for mixing sealer.

(b) Mix eight ounces of sealer to one ounce of accelerator.

NOTE: Mix only enough sealer that will be used for the application. The mixed sealer must be applied within four hours. Discard any sealer that has been mixed over a four hour period.

(4) Apply an even brush coat of sealer over the entire surface of the boot. Allow 30 minutes drying time, then apply a second coat to boot surfaces.

NOTE: If aircraft operational status permits, allow a two hour drying time after the second coat application. If the two hour drying period is not possible, the aircraft may be flown immediately.

- The De-icer should be reinspected at subsequent 100-hour inspection periods and after flight exposure in heavy rain. Touch-up application is to be made as necessary to maintain a continuous film coating. Clean touch-up area sufficiently to remove foreign material and any loose coating.

(5) Discard remaining sealer and clean equipment as required.

11. Deicer Boot Cleaning- Hydrin or Neoprene

A. To aid in the service life of the de-icer boots, they should be cleaned whenever an excess of dust, dirt and etc. is accumulated on the boot surfaces. To clean the de-icer boot surfaces, proceed as follows:

CAUTION: DO NOT USE MEK OR OTHER SOLVENTS WHEN PERFORMING THIS PROCEDURE.

(1) Prepare a solution of a mild detergent and water.

(2) Wash boot surfaces until clean.

(3) After washing, wipe with a clean cloth dampened with plain water.

(4) Wipe boot surfaces with a dry cloth.

Kit P/N 74-451-K consists of:
*Cement P/N KE7804 Procure from B.F. Goodrich Aerospace and Defense Products,
*Accelerator P/N KE7005 500 South Main Street, Akron, Ohio 44318

1. Operational Check-out Of Pneumatic Deicing System - NA-265-70 Airplanes

A. Leakage Check of Pressure Lines in Pneumatic Deicing System

(1) Disconnect line at inlet of ejector and plug line.

(2) Disconnect and plug downstream end of line to vacuum pump in nose pressurization compartment at right-hand fuselage station 182.

(3) Disconnect pressure inlet lines and outlet lines to both forward and aft distributors. Using a Y-type manifold for forward distributor and a single line for aft distributor, connect air pressure lines and bypass both distributors.

(4) Disconnect downstream line from pressure regulator and connect air pressure source (regulator must be downstream from shutoff valve).

(5) Pressurize system to 18 (±1) psig. Shut off air pressure and hold in line for 5 minutes. Check that pressure does not drop more than one psig.

(6) Remove equipment and restore system to normal configuration.

B. Leakage Check of Vacuum Lines in Pneumatic Deicing System

(1) Disconnect vacuum upstream line at each distributor valve and plug.

(2) Remove cap from air pressure ground test fitting. Attach air pressure source (18 to 100 psi) and regulator.

(3) Remove cap from vacuum ground test fitting and install a mercury manometer or a vacuum gage.

(4) Apply air pressure of 18 to 100 psig at pressure ground test fitting.

(5) Shut off air source when vacuum gage indicates 10 in. Hg. Allow system to stabilize (between 9 and 10 in. Hg); then carefully note vacuum gage indication.

(6) Allow system to set for 5 minutes; then recheck vacuum gage reading. Vacuum should not drop more than one in. Hg in 5 minutes.

WARNING: IF POSITIVE PRESSURE IS USED TO TRACE LEAKS IN THE VACUUM SUPPLY LINE, THE VACUUM PRESSURE SWITCH MUST BE REMOVED BEFORE THE TEST OR DAMAGE TO ITS DIAPHRAGM WILL RESULT.

(7) Remove equipment and restore system to normal configuration.

C. Check-out of Pneumatic Deicing System

(1) Make sure circuit breakers "PNEU BOOTS" and "PNEU BOOT & NOSE PRESS FAIL IND" are engaged.

(2) Remove cap from air pressure ground test fitting. Attach air pressure source (18 to 100 psig) and regulator.

(3) Connect external 28-volt dc power to airplane bus system.

(4) Check that "VACUUM FAIL" light, on caution light panel, lights; then pressurize pneumatic system.

(5) Check for airflow from ejector and that "VACUUM FAIL" light goes out.

(6) Actuate "PNEUMATIC BOOTS" switch to LT.

(7) Check that "PNEUMATIC BOOTS OPERATING" lights on the instrument panel ("WING," "SLATS," and "EMP"), come on in a sequence of 5 seconds each; then are off for 165 seconds. Check complete cycle three times.

(8) Have assistant outside the airplane check that section of deicing boots that is inflating corresponds to light that is lit on instrument panel.

(9) Reposition "PNEUMATIC BOOTS OPERATING" switch to HVY.

(10) Repeat step (5), except light will be off for 45 seconds in each cycle.

(11) Remove equipment and restore system to normal configuration.

D. Check-out of Pneumatic Boot System Manual Override

(1) Make sure "PNEU BOOT & NOSE PRESS FAIL IND" circuit breaker is engaged.

(2) Remove cap from air pressure ground test fitting. Attach air pressure source (18 to 100 psig) and regulator.

(3) Connect external 28-volt dc power to airplane bus system.

(4) Check that "VACUUM FAIL" light, on caution light panel, lights; then pressurize pneumatic system.

(5) Check for airflow from ejector and that "VACUUM FAIL" light goes out.

(6) Depress "PNEUMATIC BOOTS OPERATING" lights consecutively for 5 seconds. Wait 45 seconds and repeat cycle two more times.

(7) Have assistant outside airplane check that boot inflating corresponds to light pressed.

(8) Remove equipment and restore system to normal configuration.

2. Obstruction and Pressure Check of Deicing System - NA-265-80 Airplanes

A. Checkout of Pneumatic Deicing System

(1) Disconnect pressure lines at the inlets of the forward and aft ejector/distributor valves.

(2) Position PNEUMATIC BOOTS switch in cockpit to OFF.

(3) Disconnect downstream line from pressure regulator and connect air source to downstream line.

(4) Pressurize system to 18 (±1) psig.

(5) Check for airflow at the pressure supply lines of the ejector/distributor valves. This verifies that the lines to the ejector/distributor valves are free of obstructions.

(6) Disconnect the lines at the outlet of the forward and aft ejector/distributor valves. Using jumper lines, connect these lines to the pressure lines previously removed from the ejector/distributor valves.

(7) Disconnect the supply lines from the boot assemblies. Pressurize the system to 18 (±1) psig. Verify that air flows from each boot supply line.

(8) Remove pressure from system and reconnect each boot supply line.

(9) Pressurize to 18 (±1) psig. Shut off air supply and hold pressure in lines for 5 minutes. Verify that pressure does not drop over 1 psi from the initial reading on the gage. If pressure drops more than 1 psi, locate and repair leak.

(10) Remove air pressure source from downstream side of pressure regulator and reconnect line previously removed from pressure regulator.

(11) Disconnect lines to both ejector/distributor valves. Install pressure gage on pressure lines at inlets of ejector/distributor valves.

(12) Remove ground pressure test fitting cap at line upstream of pressure regulator and apply air pressure of between 18 - 100 psig.

(13) Check that pressure indication on gage is 18 (±1) psig. This verifies that pressure regulator is functioning properly.

(14) Remove the pressure gages at the inlets of the ejector/distributor valves and the jumper lines bypassing the ejector/distributor valves.

(15) Reconnect lines to inlets and outlets of ejector/distributor valves.

(16) Perform operational checkout of deicer boot system.

3. Operational Check (Engines not operating) of Deicer Boot System - NA-265-80 Airpl

A. Perform operational check of deicer boot system as follows:

(1) Remove ground test pressure cap and apply 18 (±1) psig to system.

(2) Apply electrical power to airplane.

(3) Position PNEUMATIC BOOTS switch in cockpit to LT.

(4) Check that the deice boot system lights in the cockpit and the deicer boots cycle as follows:

Pressurized - 15 seconds
Vacuum - 165 seconds

(5) Position PNEUMATIC BOOTS switch to HVY.

(6) Check that the deice boot system lights in the cockpit and the deicer boots cycle as follows:

Pressurized - 15 seconds
Vacuum - 45 seconds

(7) Check manual operation of system by pressurizing the system to 18 (±1) psig then depressing the wing, slat and empennage illuminated switches consecutively for 5-seconds each and verify that corresponding boots inflate.

(8) Remove air pressure source from ground test fitting and install cap on ground test fitting.

4. Operational Checkout of Pneumatic Deicing System - Engine Operating-NA380Airplanes

A. To perform operational checkout of pneumatic deicing system, proceed as follows:

(1) Start engines (refer to Chapter 71).

(2) Position PNEUMATIC BOOTS deicer switch to LT.

(3) Check that PNEUMATIC BOOTS OPERATING lights (WING, SLATS and EMP) come on in a sequence of 5 seconds each, then off for 165 seconds during each cycle.

(4) Have assistant outside the airplane check that section of deicing boots that is inflating corresponds to light that is illuminated.

(5) Position PNEUMATIC BOOTS OPERATING switch to HVY.

(6) Check that PNEUMATIC BOOTS OPERATING lights come on in a sequence of 5 seconds each, then off for 45 seconds during each cycle.

(7) Position PNEUMATIC BOOTS OPERATING switch to OFF.

(8) Shut down engines and secure airplane as required.

HEATED AIR INTAKES - DESCRIPTION AND OPERATION

1. Environmental System Ram-air Inlet Duct Opening and Bird-stopper Valve* Electrical Heating System

The ram-air inlet duct and the bird-stopper valve,* which is a spring loaded flapper valve within the ram-air inlet duct, are both located in the forward end of the dorsal fin. Both the ram-air inlet and the bird-stopper valve are equipped with electrical heating elements to prevent the formation of ice. The electrical heating elements are embedded below the smooth surface of the Spraymat material and are not readily visible. When the airplane is on the ground the bird-stopper flapper valve* is spring-loaded to the vertical position closing off the entrance to the ram-air inlet duct; thus preventing birds from entering or nesting in the inlet duct. When the airplane is airborne the bird-stopper flapper valve is forced into the open position by ram-air impact pressure. Heaters for the ram-air inlet and the bird stopper valve are controlled by the ram-air inlets anti-ice switch located on the anti-ice control panel. Because the heating elements are embedded in the Spraymat material, special maintenance practices must be employed when making repairs to the heating elements. (See figures 30-20-801 and 802).

The right and left main landing gear load switches are in a load position when the airplane is on the ground (weight on both main landing gear). This prevents the ground electrical control (GEC) relay from becoming energized. (See figure 30-20-1.) Therefore, power energizes the ram inlet anti-ice ground (RIAG) relay. When the RIAG relay is energized, there is only one ground for the ram-air Spraymat heater elements. The resistance in the Spraymat is then in series and decreases the current flow to about half. thereby decreasing the heating output to about half.

When the airplane is air-borne, the landing gear load switches are actuated to the unload position, energizing the ram inlet anti-ice air-borne (RIAA) relay. Because the circuit, through the RIAG relay, now provides a second ground for the ram-air Spraymat, the resistance is now parallel and maximum current flows, inducing maximum heat for anti-icing protection.

2. Spraymat Heater

The ram-air inlet duct opening heater and the bird-stopper valve* heater are Spraymat heaters. The Spraymat heater consists of granular metallic heater elements sandwiched between layers of insulation. A thermosetting plastic with a glass-cloth filler is used for the base and outer insulation layers. The heater element is flame-sprayed on the surface of the base insulation. The heater element is applied in the form of strips with varying width and specific resistance to give the required heating intensity. Bus bars are applied over the ends of the heater element where there is a change of direction, and over terminals. The terminals, which are usually bonded to the inside of the structure, have a terminal post which is flush with the outer surface of the base insulation. The conductor element is flame-

*Airplanes Changed by SOK #10

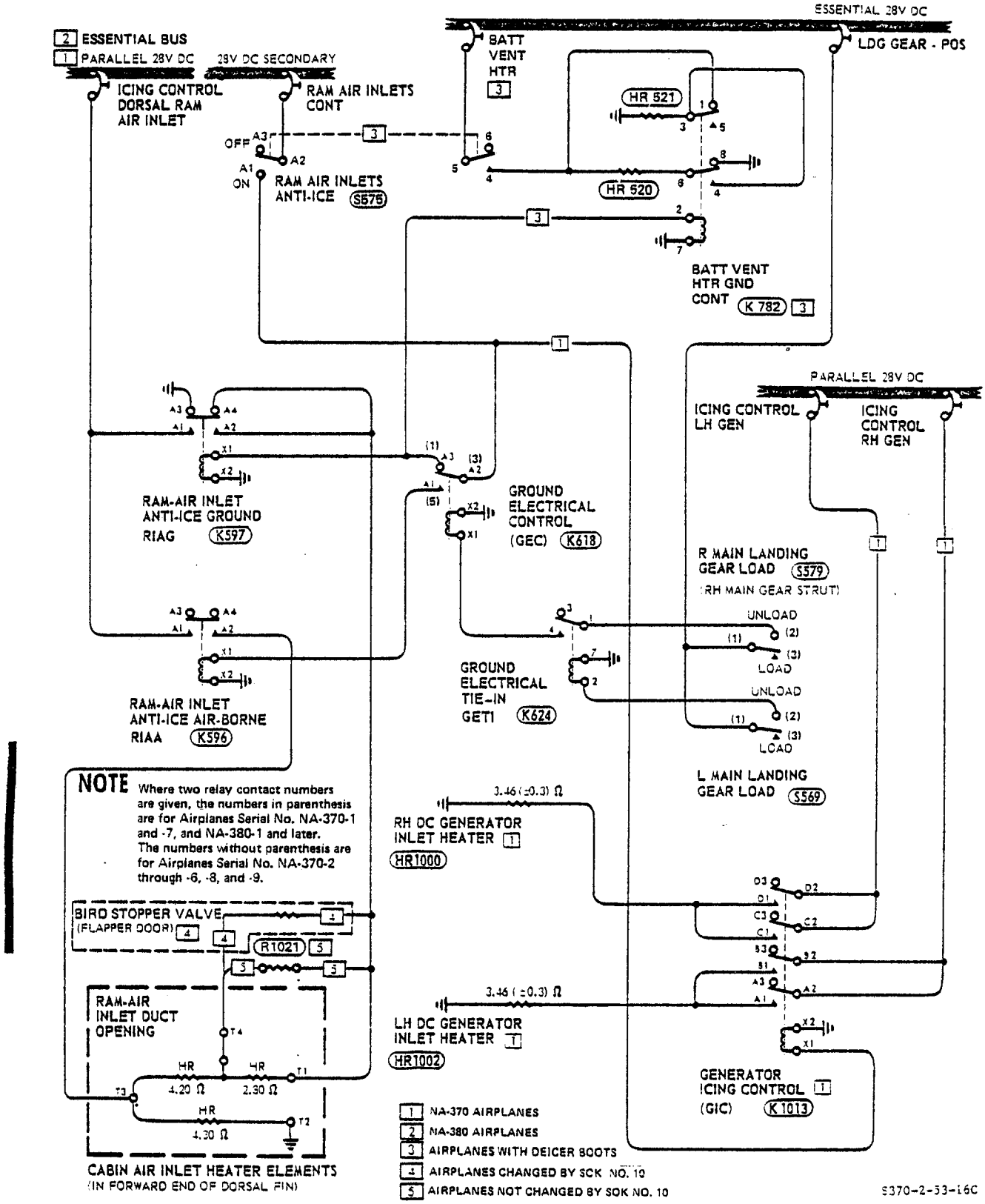
sprayed over the end of the terminal post. The outer insulation is applied over the surface of the heater elements and bus bars. Stoneguard is then applied over the outer insulation in areas requiring additional impact protection. Stoneguard is a metal-filled thermosetting plastic coating which is usually electrically bonded to the base structure by direct contact or with a special terminal. The Spraymat heater is then painted with Erocoat, a special static-and rain-erosion-resistant finish.

3. Battery Vent Electrical Heating System-NA380 Airplanes with Deicer Boots

The forward and aft external battery vents on the left side of the nose compartment are equipped with electrical heating elements to prevent ice formation on the exposed surfaces of the vent tubes. The battery vents are energized by positioning the ram-air inlet anti-ice switch at ON. Battery vent heater RDI HR521 is the aft vent heater and HR520 is the forward heater.

When the airplane is on the ground, the main landing gear load switches are in the loaded position which prevents the ground electrical tie-in and the ground electrical control relays from being energized (see figure 30-20-1). When the ram-air inlet switch (S575) is positioned at ON, electrical power is routed to the coil of the battery vent heater ground control relay (K782). With relay K782 energized, the forward and aft battery vent heaters are connected in series which decreases the current flow through the heaters, thereby decreasing the heater output to approximately half power.

When the airplane is airborne, the landing gear load switches are actuated to the UNLOAD position, energizing the ground electrical control and tie-in relays. When the ground electrical control relay (K618) energizes, the circuit to the battery vent heater ground control relay (K782) is broken. With relay K782 de-energized, a separate ground path is provided to the forward battery vent heater. In this condition, both heaters are connected in parellel which allows maximum current flow and inducing maximum heat for anti-icing protection.


ESSENTIAL 28V DC
2 ESSENTIAL BUS
1 PARALLEL 28V DC
28V DC SECONDARY
ICING CONTROL DORSAL RAM AIR INLET
RAM AIR INLETS CONT
OFF
ON
RAM AIR INLETS ANTI-ICE S576
BATT VENT HTR 3
HR 521
HR 520
BATT VENT HTR GND CONT K 782 3
PARALLEL 28V DC
ICING CONTROL LH GEN
ICING CONTROL RH GEN
RAM-AIR INLET ANTI-ICE GROUND RIAG K597
GROUND ELECTRICAL CONTROL (GEC) K618
R MAIN LANDING GEAR LOAD S579
(RH MAIN GEAR STRUT)
UNLOAD
LOAD
RAM-AIR INLET ANTI-ICE AIR-BORNE RIAA K596
GROUND ELECTRICAL TIE-IN GETI K624
L MAIN LANDING GEAR LOAD S569
NOTE Where two relay contact numbers are given, the numbers in parenthesis are for Airplanes Serial No. NA-370-1 and -7, and NA-380-1 and later. The numbers without parenthesis are for Airplanes Serial No. NA-370-2 through -6, -8, and -9.
3.46 (±0.3) Ω
RH DC GENERATOR INLET HEATER 1 HR1000
BIRD STOPPER VALVE (FLAPPER DOOR) 4
R1021 5
RAM-AIR INLET DUCT OPENING
HR 4.20 Ω
HR 2.30 Ω
HR 4.30 Ω
CABIN AIR INLET HEATER ELEMENTS (IN FORWARD END OF DORSAL FIN)
3.46 (±0.3) Ω
LH DC GENERATOR INLET HEATER 1 HR1002
GENERATOR ICING CONTROL 1 (GIC) K1013
1 NA-370 AIRPLANES
2 NA-380 AIRPLANES
3 AIRPLANES WITH DEICER BOOTS
4 AIRPLANES CHANGED BY SCK NO. 10
5 AIRPLANES NOT CHANGED BY SCK NO. 10
S370-2-53-16C

Air Intake Anti-icing System Electrical Schematic
Figure 30-20-1

HEATED AIR INTAKES - TROUBLE SHOOTING

1. Trouble Shooting Environmental System Ram-air Inlet Duct Opening and Bird-stopper valve* Electrical Heating System

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. RAM-AIR INLET DUCT OPENING HEATING ELEMENTS DO NOT HEAT.		
(1) Defective elements.	See figure 30-20-1 for ohmic values; then check heating elements for open circuit, using an ohmmeter. (Refer to Wiring Diagram Manual.)	If heating elements are open, repair or replace.
(2) Defective relays or switches.	Check for proper switch and relay operation.	Replace defective switch or relay.
B. BIRD-STOPPER VALVE HEATING ELEMENT DOES NOT HEAT*		
(1) Defective heating element, or broken electrical leads.	Check for proper switch and relay operation. If circuit is operating correctly, check heating element for open circuit by use of ohmmeter. (Refer to Wiring Diagram Manual.)	If heating element is open, remove and replace bird-stopper valve, or repair broken electrical leads.

2. Trouble Shooting Battery Vent Electrical Heating System-NA380 Airplanes with Deicer Boots

PROBABLE CAUSE	ISOLATION PROCEDURES	REMEDY
A. FORWARD BATTERY VENT HEATER DOES NOT HEAT		
(1) Defective relay K782.	With electrical power removed from airplane, check terminal "6" of relay K782 to airplane ground. If discontinuity is present, replace relay K782. If continuity exists, check for open wiring to heater.	Replace relay or repair open wiring.
(2) Open heater element.	Check for resistance between terminals "6" and "1" or relay K782. If open circuit exists, replace heater element.	Replace heater element.

*Airplanes changed by SOK #10

B. AFT BATTERY VENT HEATER DOES NOT HEAT		
(1) Defective heater element or open wiring.	With electrical power applied to airplane and the ram-air inlet switch at ON, check for 28 VDC at terminal "3" of relay K782. If voltage is present, check heater ground for an open circuit. If circuit is not open, replace heater element.	Replace heater element or repair open wiring.

*Airplanes changed by SOK #10

HEATED AIR INTAKES - MAINTENANCE PRACTICES

1. Replacement of Ram-air Inlet Heater

A. Removing Ram-air Inlet Heater

(1) Remove access covers. (Refer to Chapter 6, figure 6-00-3 or 6-00-4 as applicable.)

(2) Through access doors, remove the four heater element electrical connections. Tape electrical connections and identify for reinstallation.

(3) Install MS20426AD4 Rivets.

(4) Tighten the four mounting bolts if not previously done. Remove tape from electrical terminals and connect terminals to proper locations.

(5) Fill gaps between heater assembly and dorsal fin.

(6) Install access doors.

2. Replacement of Battery Vent Heater(s)-NA380 Airplanes with Deicer Boots

A. Removing Battery Vent Heater(s).

(1) Remove all electrical power from airplane.

(2) Open access door on left side of nose.

(3) Remove battery sump jar.

(4) Remove screws from battery sump jar and transformer mounting plate, then raise and secure plate.

(5) Disconnect vent hoses and electrical wires to vents.

(6) Remove rivets securing vents to fuselage skin.

(7) Remove vents from airplane.

B. Installing Battery Vent Heater(s).

(1) Place serviceable vent through fuselage skin. Vent hose connections should point up.

(2) Secure vents to fuselage skin.

(3) Check that vent tube does not contact fuselage skin.

(4) Connect vent hoses and electrical wires to vent(s).

(5) Apply electrical power to airplane and check vent heaters for proper operation.

(6) Install battery sump jar and transformer mounting plate.

(7) Install battery sump jar.

(8) Check complete installation for security and proper mounting.

(9) Apply sealant around vent outlet and fuselage skin.

(10) Install access door.

3. Replacement of Bird-stopper Valve (Spring-loaded Flapper Door) - Airplanes Changed by SOK #10

A. Removing Bird-stopper Valve

(1) Remove access doors. (Refer to Chapter 6).

(2) Press spring-loaded flapper door down; then remove heater element electrical connections from aft side of door. Tape electrical wires and identify for reinstallation.

(3) Remove cotter pin from door hinge pivot pin.

(4) Check position of door spring on pivot pin; then remove pivot pin and spring at same time.

(5) Remove door.

B. Installing Bird-stopper Valve - Airplanes changed by SOK #10.

(1) Position bird-stopper valve in ram-air opening, aligning hinge holes and mounting bracket holes for pivot pin installation.

(2) Start pivot pin through one mounting bracket hole and door hinge hole. Place spring between door hinges; then slide pivot pin through spring and remaining hinge and bracket holes.

(3) Place hooked end of spring over edge of door hinge. Put cotter pin in end of pivot pin.

(4) Remove tape from electrical terminals and connect to valve. Make sure wire loop clears structure after connections are tightened.

(5) Open and close valve flapper door and check freedom of movement. Make sure heating element connecting electrical wires have sufficient loop to allow proper flapper-door operation, and that no wire insulation chafing will occur.

(6) Install access doors.

(7) Check for proper operation of bird-stopper valve heater.

4. Replacement of Dorsal Fin Leading Edge/Ram-air Inlet

A. To replace the dorsal fin leading edge, proceed as follows:

(1) Remove all electrical power from airplane.

(2) Remove the two circular access doors on the upper fuselage adjacent to the dorsal fin leading edge.

(3) Remove bird-stopper valve on Airplanes changed by SOK #10.

(4) Through access door openings, disconnect the four electrical connections to the heater elements. Protect and identify each electrical connector for reinstallation.

(5) Carefully drill out the 14 double flushed 1/8 inch diameter rivets securing the dorsal fin leading edge to the inner duct skin and dorsal fin skin.

(6) Through access door opening, remove four bolts securing dorsal fin leading edge to attach brackets on frame at fuselage station 333 625.

(7) Remove dorsal fin leading edge from airplane.

B. To install dorsal fin leading edge, proceed as follows:

(1) Place dorsal fin leading edge in position and install four bolts through dorsal fin leading edge and attach brackets on frame at fuselage station 333 625.

(2) Install and double flush the fourteen MS20426AD4 rivets through the dorsal fin skin, the dorsal fin leading edge and the inner duct skin.

(3) Through access doors, reconnect the four electrical connections to the heater elements. Make certain the electrical leads are connected to the correct terminal.

(4) Reinstall bird-stopper valve if applicable.

(5) Check complete installation for security.

(6) Perform an operational check of ram-air inlet duct heating system.

1. Spraymat Heater

A. Observe the following when testing Spraymat heater:

CAUTION: DO NOT OPERATE SPRAYMAT HEATERS AT FULL VOLTAGE ON THE GROUND EXCEPT UNDER CONTROLLED TEST CONDITIONS, OR SERIOUS DAMAGE WILL RESULT.

(1) Use any cleaning fluid approved for use on aircraft paint to remove dirt and oil deposits from heater surface. Do not soak or immerse heater in solvent.

(2) Inspect surface for scoring, abrasions, cracks, and other indications that the outer insulation may be damaged. Cracks, dents, discoloration, or blistering may indicate damage to heater element or insulation.

(3) Check terminals for breakage.

(4) Check all electrical connections for soundness.

2. Spraymat Heater Trouble Indications

A. Cracks, discoloration, or charring in the Stoneguard coating will cause local overheating and outer insulation resistance breakdown.

B. Impact damage may be easily seen.

C. Heater.element malfunctions result in local overheating, causing discolored patches which are associated with blistering or charring.

D. A severe blow that dents the heater usually breaks or cracks the conductor element, resulting in an open circuit or an abnormally high resistance.

3. Spraymat Heater Outer Insulation Test

A. Perform test as follows:

(1) Connect 500-volt dc megger ground to a terminal controlling heater area to be tested.

(2) Attach a damp swab to line lead, and pass swab over heater surface with megger on.

CAUTION: RUBBER GLOVES SHOULD BE WORN DURING THIS TEST FOR PERSONAL SAFETY.

(3) Check resistance values. The resistance value must be no less than 2 megohms at any point. (Use resistance bridge.)

4. Spraymat Heater Base Insulation Test

A. Perform test as follows:

(1) Connect leads of resistance bridge to circuit terminals of individual circuits to check for element resistance. (See figure 30-20-803 for circuit resistance values.)

(2) Measure resistance between terminal and structure with 500-volt megger. The reading must be taken after a 5-second uninterrupted application of the test voltage.

(3) Measure resistance. The resistance should measure 2 megohms minimum.

(4) Check resistance between dissimilar circuits. (See figure 30-20-803 for resistance values.) The resistance should not be less than 2 megohms.

NOTE: If a Spraymat terminal is damaged, or an element must be replaced over a temperature-sensing unit, or a heater element must be bonded to a terminal post, or more than 6 square inches of Stoneguard must be replaced, the Spraymat heater must be replace.

1. Environmental System Ram-air Inlet Duct Opening and Bird-stopper Valve* Electrical Heating System Check-out Procedure

A. Checking Ram-air Inlet Duct Opening and Bird-stopper Valve* Anti-Icing Elements

(1) Connect external dc power to airplane bus system.

(2) Make sure ICE CONTROL - "RAM AIR INLETS," "DORSAL RAM AIR INLETS" and "LANDING GEAR - POS" circuit breakers are engaged.

(3) Activate ram-air inlet anti-ice switch to ON. Check for presence of heat at ram-air opening and bird-stopper valve*. On aircraft with de-icer boots, check battery vent heaters for presence of heat. Place switch at OFF.

(4) Have helper disconnect actuating bell crank from right and left main gear load switches (on main gear scissors).

(5) Alternately actuate both main gear load switches to unload position. Actuate wing and ram-air inlet anti-ice switch to ON.

(6) Check for presence of heat at units being checked. Actuate wing and ram-air inlet anti-ice switch to OFF and main gear load switches to load.

(7) Reconnect and saftey load switches.

2. Battery Vent Heater System Checkout Procedure-NA380 Airplanes with Deicer Boots

A. Checking Battery Vent Heaters

(1) Apply external electrical power to airplane.

(2) Engage "BATT VENT HEAT" circuit breaker.

(3) Position RAM AIR INLET switch on ON.

(4) Check battery vent heaters for "low" heat.

CAUTION: TO PREVENT OVERHEATING OF RAM AIR INLET DUCT AND BATTERY VENT HEATERS, DO NOT LEAVE MAIN LANDING GEAR LOAD SWITCH IN THE UNLOADED (AIRBORNE) POSITION FOR AN EXTENDED PERIOD.

(5) Disconnect LH main landing gear load switch and position switch to the unloaded (airborne) position.

(6) Check battery vent heaters for "high" heat.

(7) Position LH main landing gear load switch to ground (loaded)position. Connect and safety link.

(8) Position RAM AIR INLET switch to OFF.

(9) Remove external electrical power from airplane.

(10) Secure airplane as required.

* Airplanes changed by SOK #10

1. Base Insulation and Heater Element Repair

If impact damage causes a break in conductor element and/or a dielectric breakdown of base insulation, repair heater element and base insulation.

A. The following equipment is required:

- Spraymat field repair kit [manufactured by Spraymat Division (subsidiary of Sierracin Corp), Burbank California]
- Trichlorethylene
- 250-watt infrared heat lamp
- Emery cloth No. 120, 240, and 320
- Soldering iron
- Paint brush
- Sharp-edge scraper
- Protective hand covering

WARNING: RESIN COMPOUNDS IN FIELD REPAIR KIT ARE SKIN IRRITANTS. PROTECTIVE HAND COVERING SHOULD BE WORN. IF NO PROTECTION IS USED, WASH SKIN THOROUGHLY WITH SOAP AND COLD WATER AT COMPLETION OF PROCEDURES.

B. Repair heater and insulation as follows:

(1) Check repair limitations as shown in figure 30-20-802.

(2) Use scraper to remove outer insulation from around damaged area. Remove all indications of cracks and surface damage.

(3) Remove damaged part of conductor element.

(4) Remove all indications of charring or cracks in base insulation. (See figure 30-20-802.)

(5) Roughen exposed surface of structure with emery cloth.

(6) Remove components "A" and "B" from Spraymat field repair kit. Extrude equal lengths of components "A" and "B" and mix thoroughly in a clean container. Apply resin with a spatula to damaged base insulation. Cure with a 250-watt, infrared heat lamp, placed 8 to 10 inches distant, until resin is hard (30 minutes at 203°F or 12 hours at 70°F).

NOTE: Components with dented surfaces can be filled with this resin mixture, eliminating the necessity of repairing the shell. Then the surface should be built up to the original base insulation contour.

(7) Sand base insulation smooth; then check base insulation resistance. (Refer to procedure for Spraymat heater base insulation test.)

(8) See figure 30-20-801 for proper repair strip number and width. See figure 30-20-802 for proper length of heater element repair strip. Identify damaged heater element by removing Erocoat finish with emery cloth in a 1/2-inch-wide strip to edge of component. (Do not remove insulation.) Moisten exposed area. By counting the elements from the base to the damaged area and referring to figure 30-20-803, the damaged element can be identified.

(9) Trim repair strip to proper size. The gap between the element and the repair strip should be 0.050 inch minimum. Using only a scraper, carefully remove outer insulation to expose heater element for 0.10 inch over full width at each end of break. (See figure 30-20-801.) By same method, expose 0.10 inch over full width at each end of repair strip.

(10) Tin ends of repair strip and heater element for 0.10 inch only. Cut a gauze repair strip 0.20 inch by width of heater element. Solder gauze repair strip to tinned surface of repair strip. (See figure 30-20-801.) Overlap gauze strip 0.10 inch only. Solder overlap quickly, without overheating. This is most easily done with a very hot soldering iron. Overheating of the repair strip causes the heater element to lift from the backing.

(11) Coat back and edges of repair strip with a minimum amount of resin compound. (Refer to step (6).) Press repair strip into position to remove any air trapped beneath.

(12) Solder gauze strips to exposed ends of heater element, soldering full area of contact.

(13) Repair copper bus bars and sprayed copper leads by same method as stated in steps (8) through (12), except do not use element repair strip. Cut tinned gauze to width and shape of bus bar or lead to be repaired. Cut gauze 0.20 inch longer than gap to be bridged, and then solder ends of gauze so that there is 0.10 inch of overlap at each end.

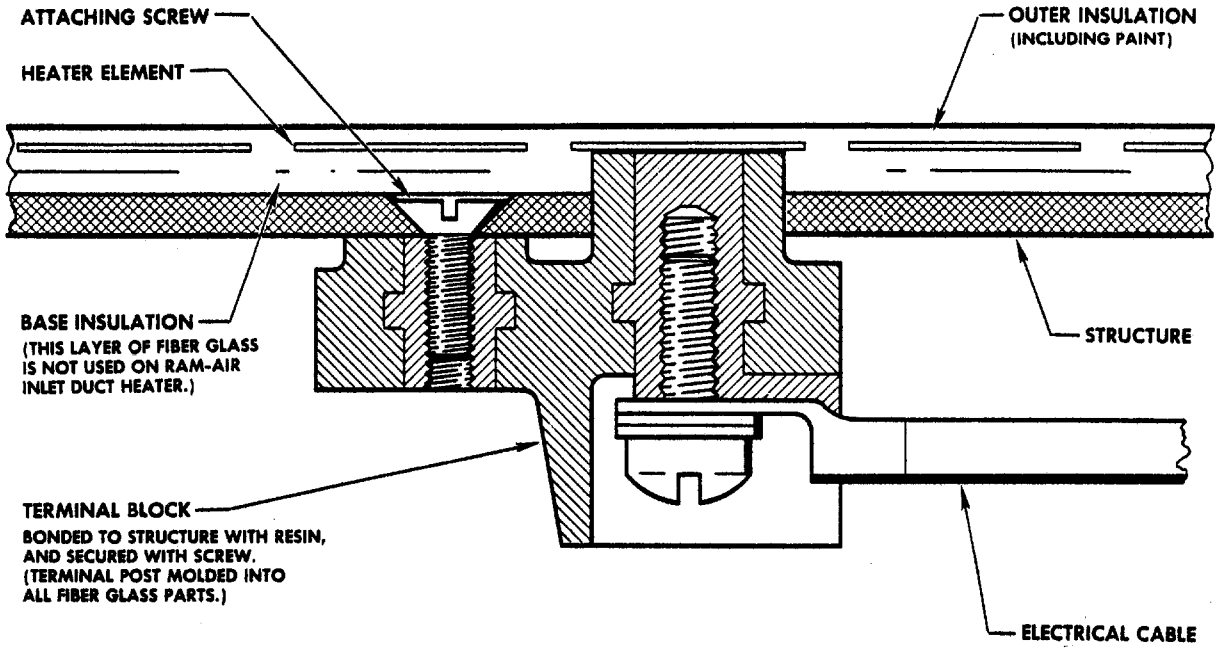
(14) Check repaired circuit for continuity. (See figure 30-20-803.) Check repaired circuit resistance.

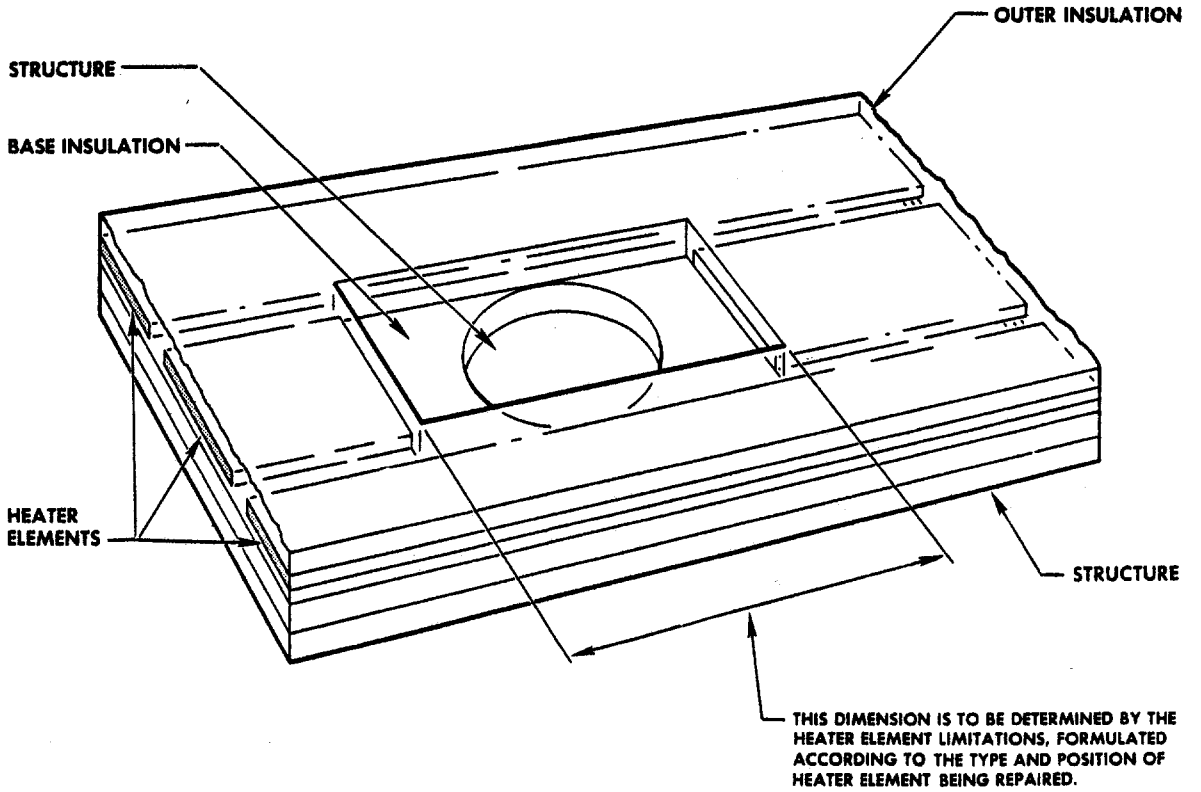
(15) Repair outer insulation and Stoneguard.

2. Outer Insulation and Stoneguard Repair

A. Repair Stoneguard and outer insulation after damages caused by scoring, abrasion, cracking, dielectric breakdown, or repair of element and base insulation.

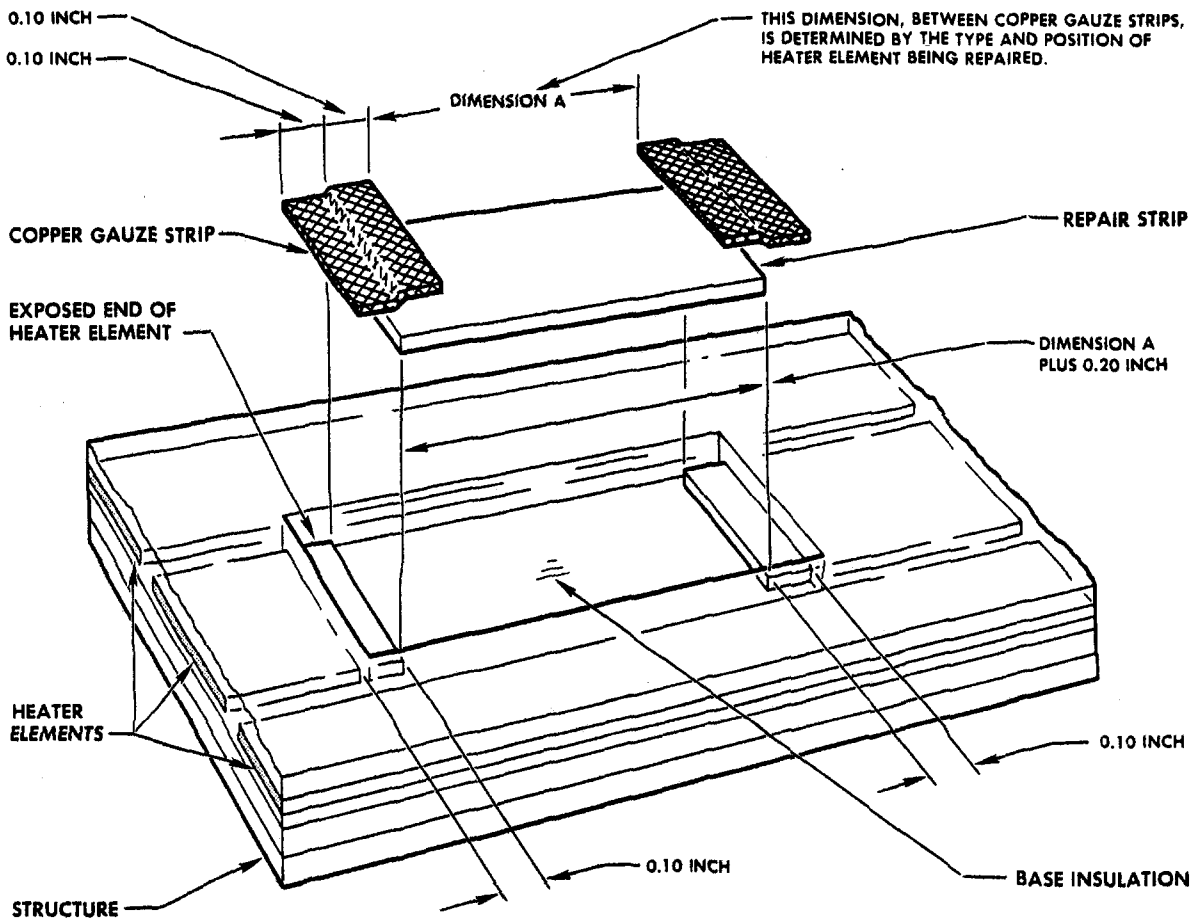
(1) Use equipment noted in base insulation and heater element repair procedure.

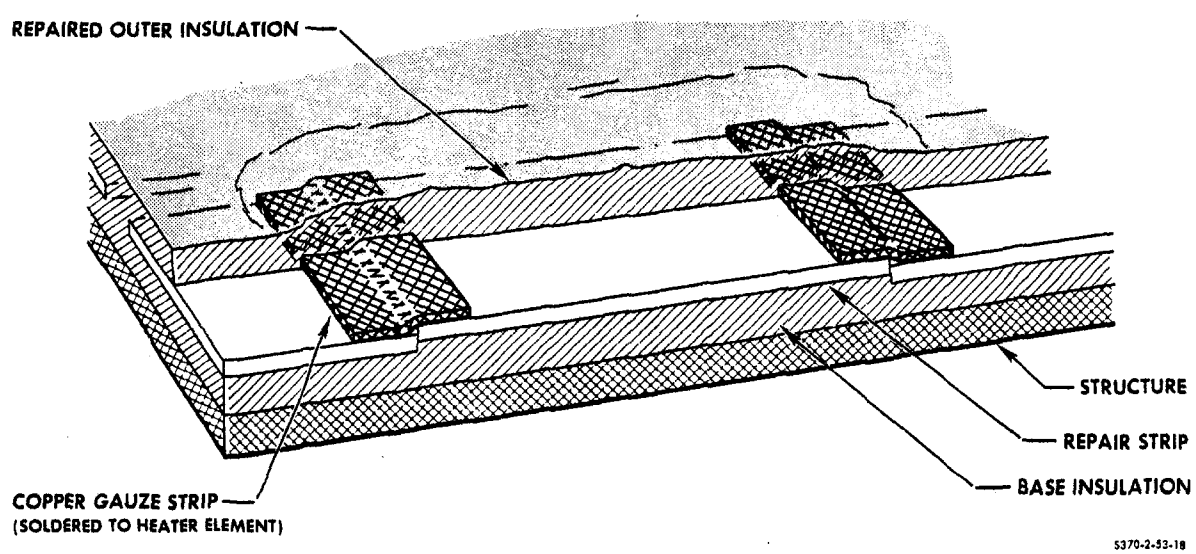

ATTACHING SCREW
HEATER ELEMENT
OUTER INSULATION
(INCLUDING PAINT)
BASE INSULATION
(THIS LAYER OF FIBER GLASS IS NOT USED ON RAM-AIR INLET DUCT HEATER.)
STRUCTURE
TERMINAL BLOCK
BONDED TO STRUCTURE WITH RESIN, AND SECURED WITH SCREW. (TERMINAL POST MOLDED INTO ALL FIBER GLASS PARTS.)
ELECTRICAL CABLE


OUTER INSULATION
STRUCTURE
BASE INSULATION
HEATER ELEMENTS
STRUCTURE
THIS DIMENSION IS TO BE DETERMINED BY THE HEATER ELEMENT LIMITATIONS, FORMULATED ACCORDING TO THE TYPE AND POSITION OF HEATER ELEMENT BEING REPAIRED.

NOTE Ends of heater element being repaired are exposed at a later stage.

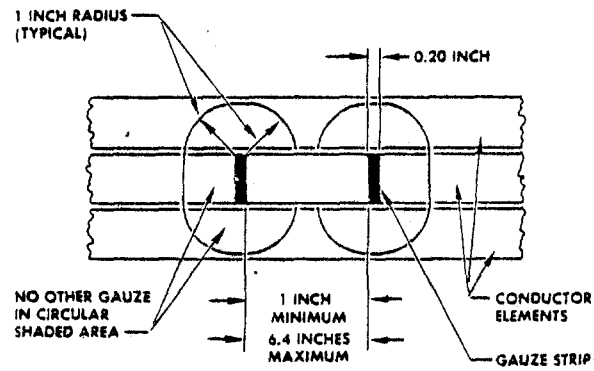
5370-2-53-17



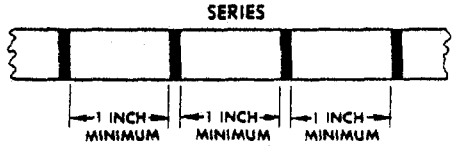


Repairing Spraymat Heaters
Figure 30-20-801 (Sheet 2 of 2)

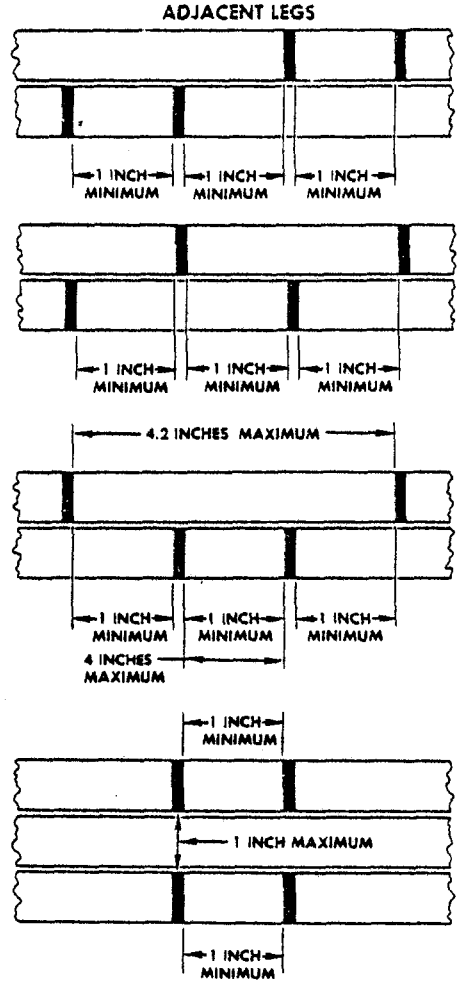
STANDARD LIMITS



SERIES



ADJACENT LEGS

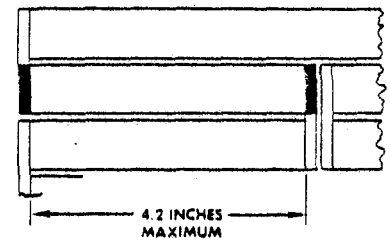


VARIATIONS

BUS BARS LESS TRAN 4.2 INCHES APART

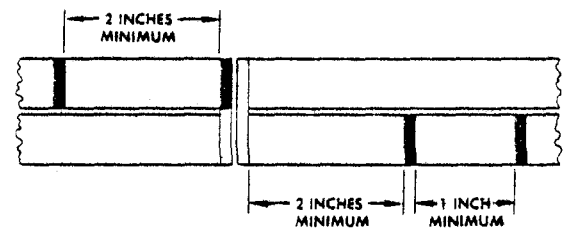
GAUZE STRIP MUST BE ON EACH BUS BAR.

NOTE A single repair strip may also be used in this manner if the distance between bus bars does not exceed 6.4 inches.

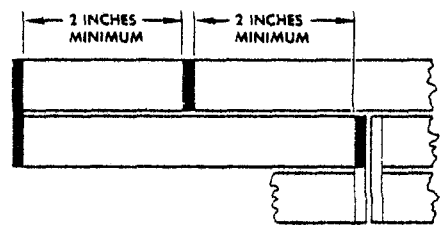


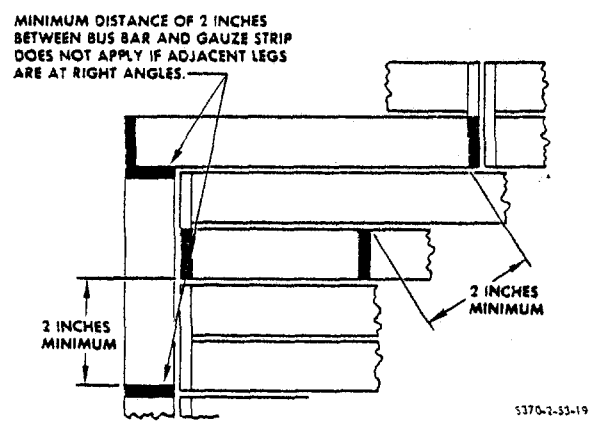
ON OR NEAR A BUS BAR

GAUZE STRIP MAY BE ON A BUS BAR OR AT LEAST 2 INCHES FROM IT.

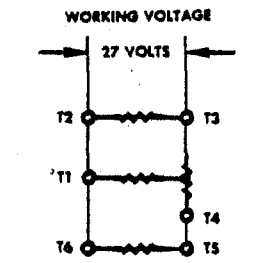


ON OR NEAR BUS BARS IN ADJACENT LEGS

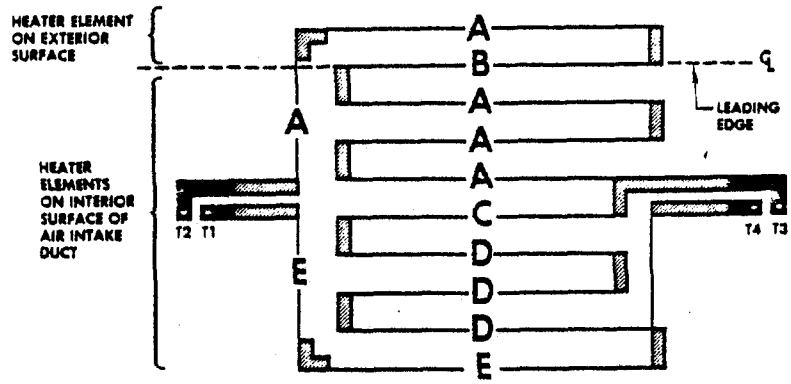



MINIMUM DISTANCE OF 2 INCHES BETWEEN BUS BAR AND GAUZE STRIP DOES NOT APPLY IF ADJACENT LEGS ARE AT RIGHT ANGLES.
2 INCHES MINIMUM
2 INCHES MINIMUM
5370-2-53-19

Spraymat Heater Repair Limitations
Figure 30-20-802

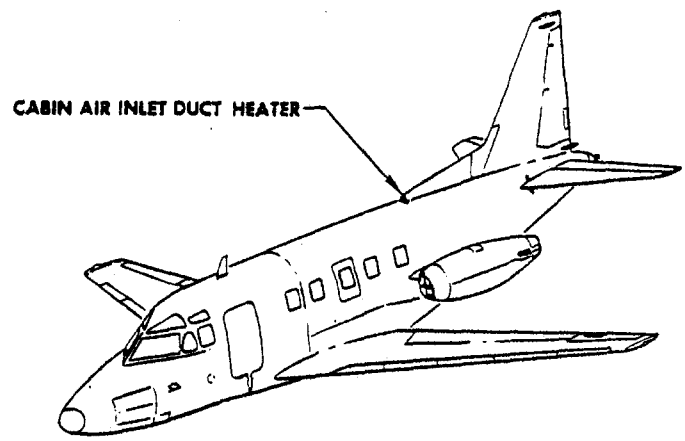


TROUBLE SHOOTING CIRCUIT
CABIN AIR INLET DUCT HEATER
AND BIRD STOPPER VALVE HEATER



CABIN AIR INLET DUCT HEATER
ELECTRICALLY NONCONDUCTIVE BASE PART. BETWEEN-CIRCUIT CHECK OR BASE-INSULATION CHECK REQUIRED. TERMINAL RESISTANCES: T2-T3 = 4.083 TO 4.513 OHMS, T1-T4 = 2.676 TO 2.958 OHMS, T3-T4 = 3.965 TO 4.382 OHMS.

ORIGINAL
As Received By
ATP



HEATER ELEMENTS			
CODE LETTER	REPAIR STRIP		ELEMENT WIDTH (IN.)
	KIT NO. 72156-100	KIT NO. 72156	
A	S-15	S-3	0.31
B	S-45	S-2	0.35
C	S-35	S-2	0.30
D	S-35	S-2	0.33
E	S-55	S-1	0.25

NOTE Finish all heaters, using black Erocoat finish. Apply three coats.

- To check terminal resistances, use ZM-4( )/U resistance bridge (Federal Stock No. 6625-777-4402).
- Maintain 0.050-inch minimum gap between all adjacent heater elements.

5370-2-33-20

Spraymat Heater Repair Orientation Drawings
Figure 30-20-803

(2) Use a scraper to remove outer insulation from around damaged area. Remove all indications of cracks and surface damage. Be careful not to damage heater element.

(3) Remove from Spraymat field repair kit components "A" and "B." Extrude equal lengths of components "A" and "B" into a clean container. Mix thoroughly with a spatula. The application of a small amount of heat will aid in mixing.

(4) Build up outer insulation slightly above original contour to allow for sanding smooth.

(5) Cure with 250-watt infrared heat lamp, placed 8 to 10 inches away from repair work, until resin is hard (30 minutes at 203°F or 12 hours at 70°F).

(6) Sand repaired area smooth with emery cloth, using water.

(7) Check resistance of outer insulation. A minute crack or pinhole in the repair can cause an unacceptable resistance check. The high voltage will arc within and char the area. Clean out crack or pinhole and repair according to steps (3) through (5); then recheck resistance.

(8) If Stoneguard protection is required in repaired area (figure 30-20-803), thoroughly mix contents of component "C" container from Spraymat repair kit.

(9) Mix equal volumes of component "A" and "C" in a clean container.

(10) Spread Stoneguard over area with spatula and cure as directed in step (5).

(11) Sand smooth to contour of Stoneguard, and cover with Erocoat. (Refer to procedure for Erocoat finish repair.)

3. Erocoat Finish Repair

A. Repair Erocoat finish after damage caused by scoring, abrasion, repairing of outer insulation, or repairing of Stoneguard.

(1) Remove dirt and oil deposits from area to be repaired. Remove any surface roughness by rubbing down with increasingly fine emery cloth, using water.

(2) Check resistance of outer insulation. If resistance is less than 2 megohms, repair outer insulation.

(3) See figure 30-20-803 for color and coats of Erocoat finish.

(4) Remove Erocoat hardener container and Erocoat black from Spraymat field repair kit. Thoroughly mix Erocoat finishes as follows: Erocoat, four parts; Erocoat hardener, one part.

(5) Brush Erocoat finish. Allow 15 minutes of drying time between coats.

(6) Air-cure Erocoat finishes for 8 hours, and then cure for one hour under 250-watt heat lamp placed 8 to 10 inches from repaired area.

CAUTION: DO NOT ATTEMPT TO PLACE HEAT LAMP CLOSE TO SPEED UP CURING. DO NOT USE HEATER ELEMENT AS A SOURCE OF HEAT, AND DO NOT USE OPEN FLAME.

(7) Sand down with fine emery paper, using water.

(8) Cover black Erocoat with aluminum paint.

DC GENERATOR RAM-AIR DUCT HEATER SYSTEM AIRPLANES SERIAL NUMBERS NA-370-2 THROUGH 6, -8 AND -9 (OPTIONAL EQUIPMENT) - DESCRIPTION AND OPERATION

1. DC Generator Ram-air Duct Heater System

Ram-air ducts in the lower section of the engine cowls provide cooling air to the dc generators. The opening of the ducts may be equipped with electrical heating elements to prevent icing conditions. The heating units are rubber boots with electrical heating elements embedded within the boot. These boots are cemented to the ram-air duct openings.

Actuating the ram-air inlet anti-ice switch to ON allows power from the "ICE CONTROL - RAM AIR INLETS" circuit breaker, through the ram-air inlet anti-ice switch ON position, to energize the generator icing control (GIC) relay. When the GIC relay is energized, power from the "ICE CONTROL - ENG INLET AND VANE - RH" and "ICE CONTROL - ENG INLET AND VANE - LH" circuit breakers is routed through the GIC relay and energizes the two generator ram-air inlet heaters. (See figures 30-00-1 and 30-20-1.)

DC GENERATOR RAM-AIR DUCT HEATER SYSTEM (OPTIONAL EQUIPMENT) - TROUBLE SHOOTING

1. Trouble Shooting DC Generator Ram-air Duct Heaters System

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. DC GENERATOR RAM-AIR DUCT HEATER ELEMENTS DO NOT HEAT.		
(1) Defective heating element or broken electrical lead.	See figure 30-20-1 for ohmic values. Check for proper switch and relay operation.	Repair or replace heater element. Replace defective switch or relay.

DC GENERATOR RAM-AIR DUCT HEATER SYSTEM (OPTIONAL EQUIPMENT) - MAINTENANCE PRACTICES

1. Replacement of DC Generator Ram-air Duct Heaters

A. Materials Required

- Cement - EC-1403 (Minnesota Mining & Mfg Co)
- Cleaning solvent - MEK (methyl ethyl ketone)
- Tackifying solvent - toluol
- Clean cloth - lint-free
- Paint brush - one-inch
- Rubber hand roller - 2-inch
- Scissors
- Masking tape

B. Preparation

(1) Dry-fit heating unit to inlet casting by taping in place and trimming, if necessary. Start by taping lead wires together and inserting lead wires in lead exit only. Mask off an area approximately 1/4 inch larger than heating unit on each side of heating unit. To prevent the feathered edges of the heating unit from curling when the cement is brushed onto them, the back side should be covered with a ply of masking tape. Mark starting points and make match marks on part and masking tape to aid in locating heating unit during final installation.

(2) Remove heating unit from casting and clean down to bare metal. Remove all paint. Wash with cleaning solvent (MEK); for final cleaning, wipe solvent off quickly with a clean, dry cloth to avoid leaving a film.

C. Application of Cement

(1) Mix EC-1403 cement thoroughly. Apply one even brush coat of EC-1403 cement to clean metal surface. Allow to air-dry for one hour.

(2) While cement on casting is drying, wash back side of heating unit (side to be cemented) with MEK. Wipe solvent quickly with clean cloth to avoid leaving a film.

(3) After first coat of cement on inlet casting has dried for one hour, apply second coat of cement to inlet and first coat to heating unit.

D. Installation

(1) When cement has become tacky, start installing heating unit on inlet casting. A clean liner cloth can be laid over the cemented surface to prevent the boot from sticking while the installation is being made. Insert lead wires through lead exit holes and line up match marks at starting point. Follow match marks in locating heating unit. If cement becomes dry, retackify with a cloth dampened with toluol. Remove enough of liner to allow sticking heating unit boot to casting for approximately a 3-inch area by pushing heel of hand near center. Work progressively toward each side to avoid trapping air under boot. Repeat liner removal and pressing-in-place procedure until entire heating unit is installed. Retackify as necessary. Roll entire heating unit with 2-inch rubber hand roller. Work roller carefully to prevent damaging part and to ensure that all trapped air is removed. Remove masking tape from feathered edges.

E. Final Finish

(1) Remove masking tape and wipe surface of inlet casting with a clean cloth dampened with toluol. Be careful not to get toluol on edge of cemented-down heating unit.

(2) Connect lead wires to proper terminals.

(3) After complete heating unit installation, a razor blade may be used to trim trailing edges of heating unit for appearance, as necessary.

(4) Areas adjacent to heating unit may be reprimed and repainted, as desired.

F. DC Generator Ram-air Duct Heater Removal

Refer to F. F. Goodrich Aviation Products, Akron, Ohio, Report No. 57-105C.

2. Checking DC Generator Ram-air Duct Heater Elements

A. Check heater elements as follows:

(1) Connect external dc power to airplane bus system.

(2) Make sure "ICE CONTROL" circuit breakers ("RAM AIR INLETS," "ENG INLET AND VANE - RH," and "ENG INLET and VANE - LH") are engaged. (See figure 30-20-1.)

(3) Have man in cockpit activate ram-air inlet anti-ice switch to ON. Check for presence of heat at ac and dc generator ram-air openings on bottom of right and left engine cowl.

CAUTION: TURN RAM-AIR INLET ANTI-ICE SWITCH ON ONLY UNTIL PRESENCE OF HEAT IS FELT; THEN RETURN SWITCH TO OFF. PROLONGED USE OF THE HEATERS ON THE GROUND COULD CAUSE THEM TO MELT.

PITOT HEATER SYSTEMS - DESCRIPTION AND OPERATION

1. Airspeed Pitot Heater System

The right and left pitot heads and on airway checker airplanes, the total air temperature sensor heads are equipped with electrical heating elements to prevent icing. The heaters are operated on the ground by positioning the "PITOT & STALL WNG" switch at ON. When the airplane is air-borne, the heaters are energized as soon as the main gear struts are extended, bypassing the "PITOT & STALL WNG" switch. (See figure 30-30-1 or 30-30-1A as applicable.)

When the airplane is on the ground, actuating the "PITOT & STALL WNG" switch to ON provides power from the "LH PITOT HT" circuit breaker to energize the left airspeed pitot head heater and power from the "RH PITOT HT" circuit breaker to energize the right airspeed pitot head heater. (See figure 30-30-1 or 30-30-1A as applicable.)

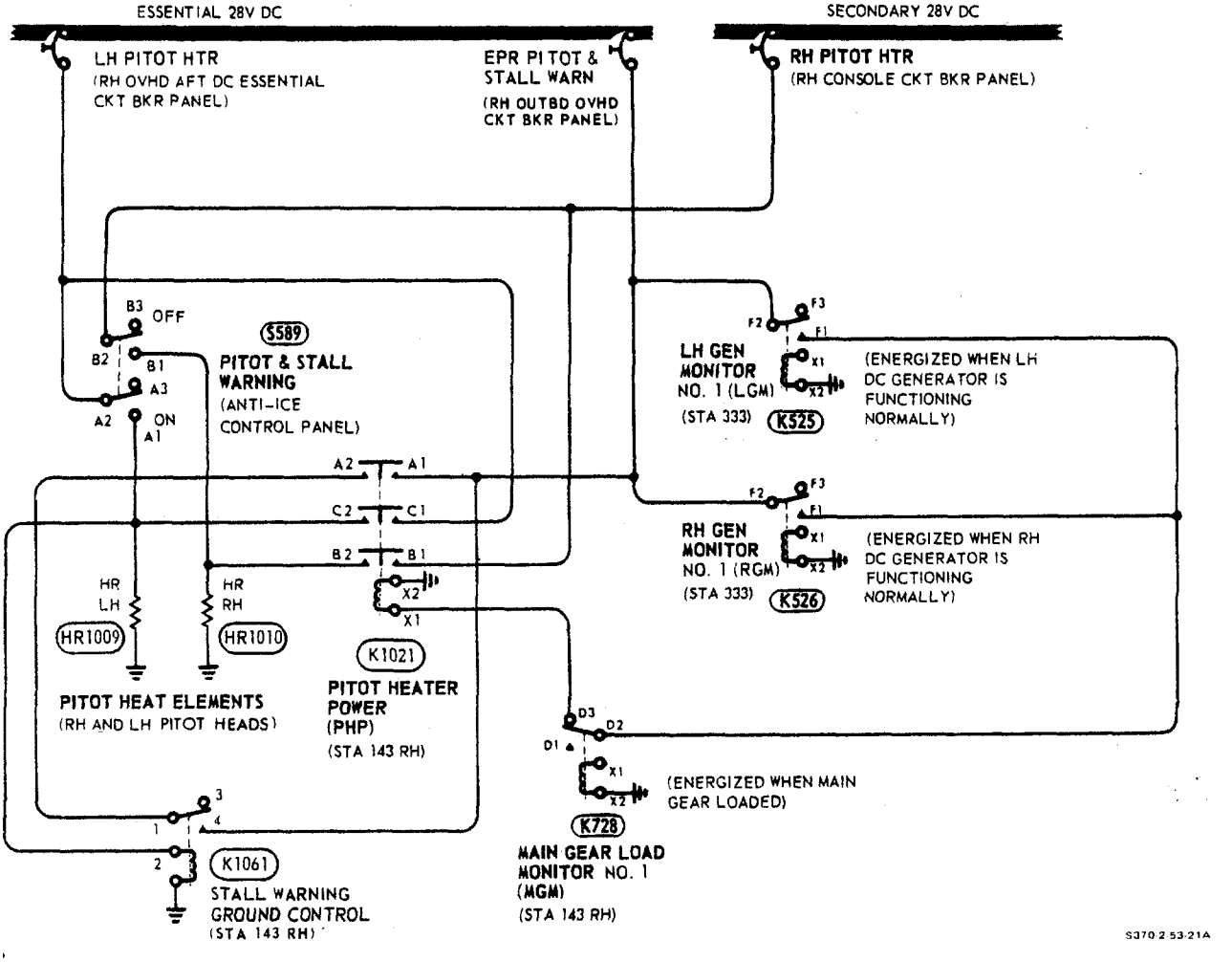
On takeoff, as soon as the main gear struts are extended, the heaters for the pitot heads are energized. The "PITOT & STALL WNG" switch is bypassed as follows: the left generator monitor (LGM) relay is energized when the left dc generator is functioning normally, and the right generator monitor (RGM) relay is energized when the right dc generator is functioning normally. Therefore, when either or both engines are operating, power from the "EPR PITOT STALL WARN HTR CONTROL" circuit breaker is routed through the LGM or RGM relays to the main gear load monitor (MGLM) relay. When the airplane is on the ground, the MGLM relay is energized, preventing power from energizing the pitot heater power (PHP) relay. Extending the main gear struts (take-off) de-energizes the MGLM relay, providing power to energize the PHP relay. Energizing the PHP relay routes power through the PHP relay contacts (by-passing the "PITOT & STALL WNG" switch) to energize the left and right air-speed pitot heaters and on airway checker airplanes, providing power to energize the total temperature sensor (TTS) relay. When the TTS relay is energized, power from the "TOTAL TEMPERATURE SENSOR HTR" circuit breaker is routed through the contacts of TTS relay to energize the heater element of total air temperature probe on the left side of the airplane at fuselage station 62.

CAUTION: THE AIRSPEED PITOT HEATERS SHOULD NOT BE OPERATED CONTINUOUSLY ON THE GROUND, BECAUSE THE EPR (ENGINE PRESSURE RATIO) PITOT HEATERS ARE ALSO ENERGIZED.

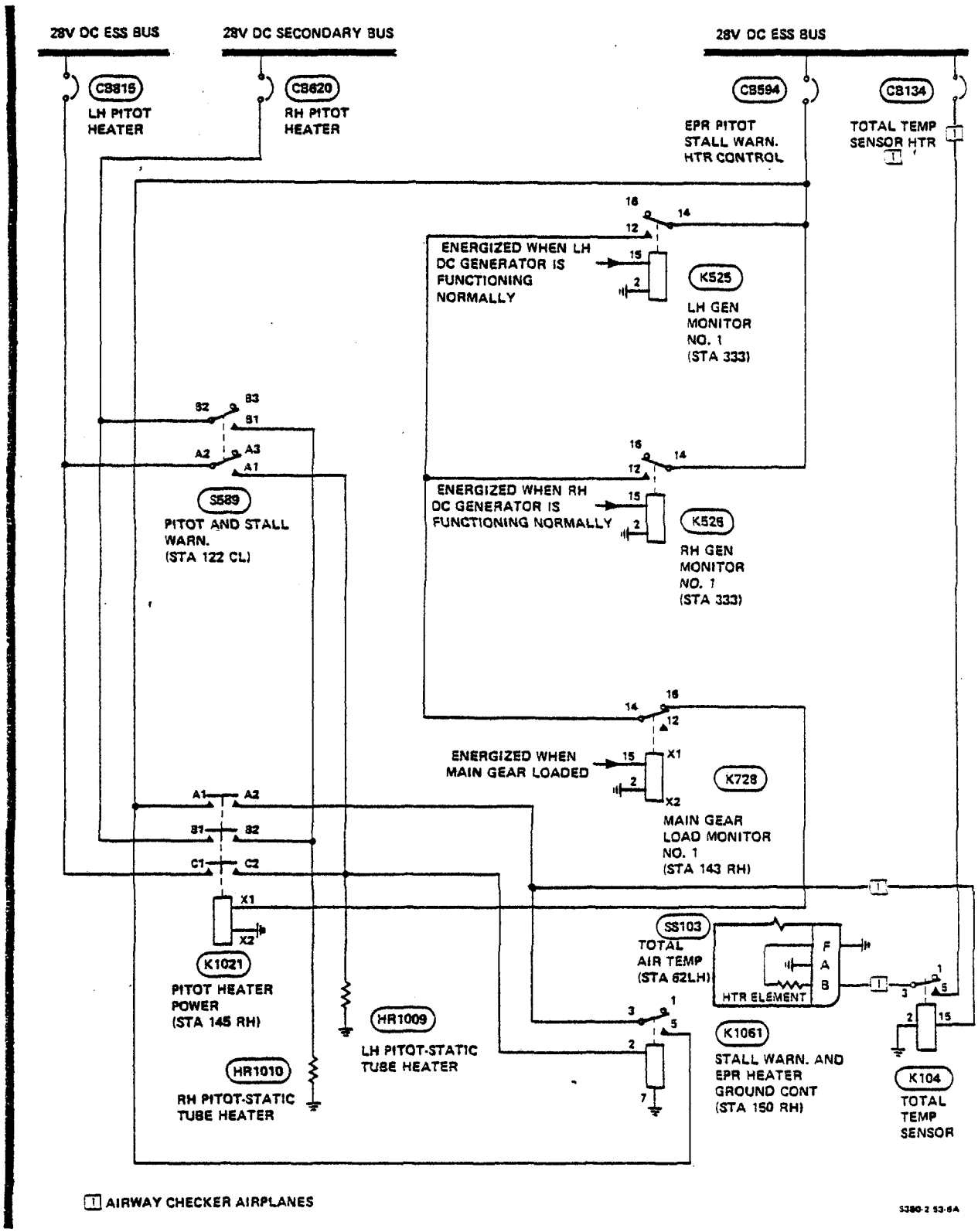
- DO NOT TOUCH ANY ONE OF THE FOUR PITOT HEATERS IF THEY HAVE BEEN ENERGIZED; SEVERE BURNS CAN BE THE RESULT OF TOUCHING.

2. Engine Pressure Ratio Pitot Heater System

Two pitot heads are installed in the leading edge of the vertical stabilizer to indicate ambient pressure for the engine pressure ratio (EPR) system. Electrical heating elements protect these pitot heads from icing conditions. These heaters can be manually operated on the ground by actuating the "PITOT & STALL WNG" switch to ON. When the airplane is air-borne, the heaters are energized as soon as the main gear struts are extended, bypassing the "PITOT & STALL WNG" switch.



Pitot Heating System Electrical Schematic - NA-370-2 Through -6, -8 and -9



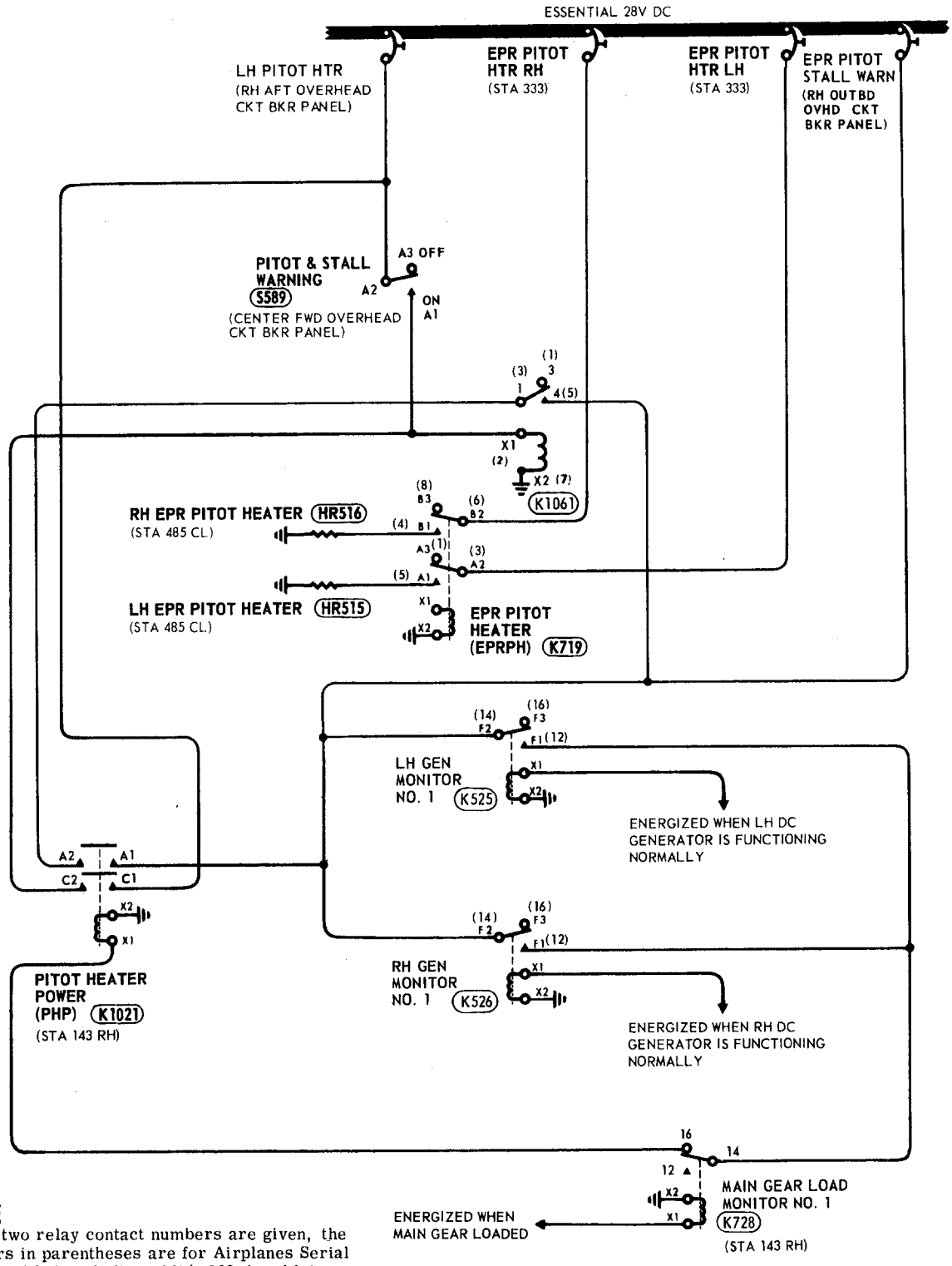
Pitot Heating System Electrical Schematic -
NA-370-1 and -7 and NA-380-1 and Later
Figure 30-30-1A

When the airplane is on the ground, actuating the "PITOT & STALL WNG" switch to ON provides power from the "LH & EPR CONT - PITOT HIT" circuit breaker to energize the EPR heater (EPRPH) relay. Energizing the EPRPH relay provides power from the "EPR PITOT HTR RH" and "EPR PITOT HTR LH" circuit breakers to energize both right and left EPR pitot head heaters. (See figure 30-30-2.)

On take-off, as soon as the main gear struts are extended, the EPR pitot head heaters are energized. The "PITOT & STALL WNG" switch is bypassed as follows: the left generator monitor (LGM) relay is energized when the left dc generator is functioning normally, and the right generator monitor (RGM) relay is energized when the right dc generator is functioning normally. Therefore, when either or both engines are operating, power from the "LH & EPR CONT - PITOT HT" circuit breaker is routed through the LGM or RGM relays to the main gear load monitor (MGLM) relay. When the airplane is on the ground, the MGLM relay is energized, preventing power from energizing the pitot heater power (PHP) relay. Extending the main gear struts (take-off) de-energizes the MGLM relay, providing power to energize the PHP relay. Energizing the PHP relay routes power through the PHP relay contacts (by-passing the "PITOT & STALL WNG" switch) to energize the left and right EPR pitot heaters.

CAUTION: THE EPR PITOT HEATERS SHOULD NOT BE OPERATED CONTINUOUSLY ON THE GROUND, BECAUSE THE AIRSPEED INDICATOR PITOT HEATERS ARE ALSO ENERGIZED.

WARNING: DO NOT TOUCH ANY ONE OF THE FOUR PITOT HEATERS IF THEY HAVE BEEN ENERGIZED; SEVERE BURNS CAN BE THE RESULT OF TOUCHING.



NOTE
Where two relay contact numbers are given, the numbers in parentheses are for Airplanes Serial No. NA-370-1 and -7, and NA-380-1 and later. The numbers without parentheses are for Airplanes Serial No. NA-370-2 through -6, -8, and -9.

Engine Pressure Ratio Pitot Heating System
Electrical Schematic
Figure 30-30-2

PITOT HEATER SYSTEMS - TROUBLE SHOOTING

1. Trouble Shooting Pitot Heater System

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. AIRSPEED PITOT HEAD OR EPR PITOT HEAD HEATERS DO NOT HEAT WHEN "PITOT & STALL WNG" SWITCH IS ACTUATED ON GROUND.		
(1) Defective pitot head heating element.	With electrical power on airplane bus, actuate "PITOT & STALL WNG" switch to ON and check pitot heads for heat	Remove defective pitot head if electrical power is available at pitot head and heater does not operate.
(2) Defective "PITOT & STALL WNG" switch.	See figure 30-30-1, 30-30-1A or 30-30-2; then with power on 29-volt dc essential bus, actuate switch and check for power on both switch contacts.	Replace switch.
B. AIRSPEED PITOT HEAD OR EPR PITOT HEAD HEATERS DO NOT HEAT WHEN AIRPLANE IS AIR-BORNE AND "PITOT & STALL WNG" SWITCH IS NOT ACTUATED.		
(1) Relay K728 stuck in energized position.	See figure 30-30-1, 30-30-1A or 30-30-2. Place main gear load switches in unloaded position. Apply power to 28-volt dc bus. Check for continuity between contacts D2 and D3 of relay K728.	Replace relay.
(2) Relay K1021 stuck in de-energized position.	See figure 30-30-1, 30-30-1A or 30-30-2. Jumper pins F2 to F1 of relay K525 or K526; then place main gear load switches in unload position. Apply power to 28-volt dc bus. Check for continuity between contacts A1 and A2 and B1 and B2 of relay K1021.	Replace relay.

PITOT HEATER SYSTEMS - MAINTENANCE PRACTICES

1. Operational Checkout of Airspeed Pitot Heater System

A. Checking Manual Operation of Airspeed Pitot Heaters

(1) Make sure the following circuit breakers are engaged:

"LH & EPR CONT - PITOT HT"

"RH PITOT HT"

"TOTAL TEMP SENSOR HTR"*

(2) Connect external 28-volt dc power to airplane bus system.

(3) Position "PITOT & STALL WNG" switch to ON.

(3A*) Have assistant place hand on total air temperature sensor at fuselage station 62 LH and check that heat is present.

(4) Have assistant place hands on pitot heads and check for presence of heat. Position "PITOT & STALL WNG" switch at OFF when pitot heads feel warm.

(5) Remove external power.

B. Checking Automatic function and Operation of Airspeed Pitot Heaters

(1) Make sure the following circuit breakers are engaged:

"LH & EPR CONT - PITOT HT"

"RH PITOT HT"

"TOTAL TEMP SENSOR HTR"*

(2) Disconnect right and left main gear load switches at left and right main gear strut scissors. Keep switches in load position.

(3) Start right engine.

(4) Have assistant place hands on pitot heads. There should be no indication of heating.

(4A*) Have assistant place hand on total air temperature sensor and check that there is no indication of heating.

(5) Position right and left main gear load switches to the extended (unloaded) position. An assistant should immediately check pitot heads, and on airway checker airplanes the total air temperature sensor, for presence of heat. The heat should be felt within 10 seconds.

*Airway Checker Airplanes

(6) Position right and left main gear load switches to loaded position. The pitot heads should cool off (no heat indications).

(7) Deleted.

(8) Shut down right engine.

(9) Start left engine. Repeat steps (4) and (5).

(10) Shut down left engine, and connect right and left main gear load switches.

(11) Remove power from airplane.

2. Operational Check-out of Engine Pressure Ratio Pitot Heater System

A. Checking Manual Operation of EPR Pitot Heaters

(1) Make sure the following circuit breakers are engaged:

"LH & EPR CONT - PITOT HT"

"EPR PITOT HTR RH"

"EPR PITOT HTR LH"

(2) Connect external 28-volt dc power to airplane bus system.

(3) Position "PITOT & STALL WNG" switch to ON. (See figure 30-30-1 or 30-30-1A as applicable.)

(4) Have assistant place hands on EPR pitot heads and check for presence of heat. Position "PITOT & STALL WNG" switch at OFF when pitot heads feel warm.

(5) Remove external power.

B. Checking Automatic Function and Operation of EPR Pitot Heaters

(1) Make sure the following circuit breakers are engaged:

"LH & EPR CONT - PITOT HT"

"EPR PITOT HTR RH"

"EPR PITOT HTR LH"

"LANDING GEAR - POS"

(2) Disconnect right and left main gear load switches at left and right main gear strut scissors. Keep switches in load position.

(3) Start right engine.

(4) Place ladder by left side of empennage and have assistant place hands on EPR pitot heads. There should be no indication of heating.

(5) Position right and left main gear load switches to extended (unloaded) position. An assistant should immediately check EPR pitot heads for presence of heat. The heat should be felt within 10 seconds.

(6) Position right main gear load switch to loaded position. The EPR pitot heads should cool off (no heat indications).

(7) Position right main gear load switch to unload position, and left main gear load switch to load position. The EPR pitot heads should remain cold.

(8) Shut down right engine, remove ladder from left side of empennage, and position right main gear load switch to load position.

(9) Start left engine. Repeat steps (4) and (5), approaching empennage, from right side.

(10) Shut down left engine, and connect right and left main gear load switches.

(11) Remove power from airplane.

WINDSHIELD ANTI-ICING SYSTEM - DESCRIPTION AND OPERATION

1. Windshield AC-powered Anti-icing System

A. Windshields

Two heated windshields (left and right) of either glass or acrylic are used on the airplane. Each windshield panel contains a single integral conductive-coated heating element on the inner face of the outside laminate, next to a vinyl interlayer. For maximum bird-impact strength, the vinyl interlayer must be maintained at an optimum temperature. Two temperature-sensing elements, embedded in the vinyl layer of each panel, are used with the windshield temperature controller to maintain optimum temperature. One of the sensors is for temperature regulation, and the other is for overtemperature indication and protection. The resistance of each temperature-sensing element used with either the acrylic or the glass windshield is 336 (±3) ohms at 115°F for the acrylic windshield, or 105°F for the glass, with no sensor heating due to current flow in the sensor circuit. Sensitivity of the sensors is about 0.7 ohm for each degree Fahrenheit of temperature; therefore, either type of windshield (glass or acrylic) may be used with the temperature controller.

B. Windshield Heating Inverter Power

The windshield heating inverter power is supplied by the No. 2 inverter, which has a 2500-volt-ampere (VA) rating for normal windshield heating system operation. Windshield heating emergency power is supplied by the instrument inverter, which has a rating of 750 VA. The maximum power used by the windshield anti-icing system is 2900 VA (noncontinuous) for normal system operation. The maximum power used for emergency operation is 750 VA, which provides reduced power to only one windshield. If the No. 2 inverter fails, the instrument inverter takes over and an external automatic relay circuit connects the power output of the temperature controller to a transformer. Output of the transformer, at reduced voltage, is applied to the heating element of one of the windshield panels. The transformer is required because of the reduced (750 VA) power output of the instrument inverter. Power output for the No. 2 inverter and instrument inverter is Y-connected with 200-volt connections from phase A to phase B when used for windshield heating.

C. Windshield Temperature Controllers

Two windshield temperature controllers, in the nose section (one for each windshield), are provided to monitor and control windshield temperature. The controller provides external indication of windshield power off and windshield overheat. Power to the windshields is automatically reduced to zero if an overheat condition is sensed. An integral test switch and indicator light is provided on the controller to test the overheat protection circuits. Adjustment potentiometers are provided for adjustment of temperature control and overheat.

ORIGINAL AS
ISSUED BY

D. Transformers

A voltage step-down autotransformer is provided for each windshield to reduce the maximum load under emergency conditions with instrument inverter power. Each transformer has three output voltage taps to provide for variations in individual windshield heating element resistance.

E. System Control

A rotary switch is used to control the on-off operation of the windshield heating system. With the switch positioned at BOTH, each of the two windshields is provided with full power. In case of No. 2 inverter failure with the switch at BOTH, reduced emergency power from the instrument inverter is available only to the left (pilot's) windshield. With the switch at LH or RH, either full normal or reduced emergency power is available to the selected windshield.

F. System Caution and Indicator Lights

A windshield power-off light and a windshield overheat caution light are provided for each windshield, and an inverter No. 1 power-off light and an inverter No. 2 power-off light are provided to indicate power source availability for windshield heating.

G. System Test

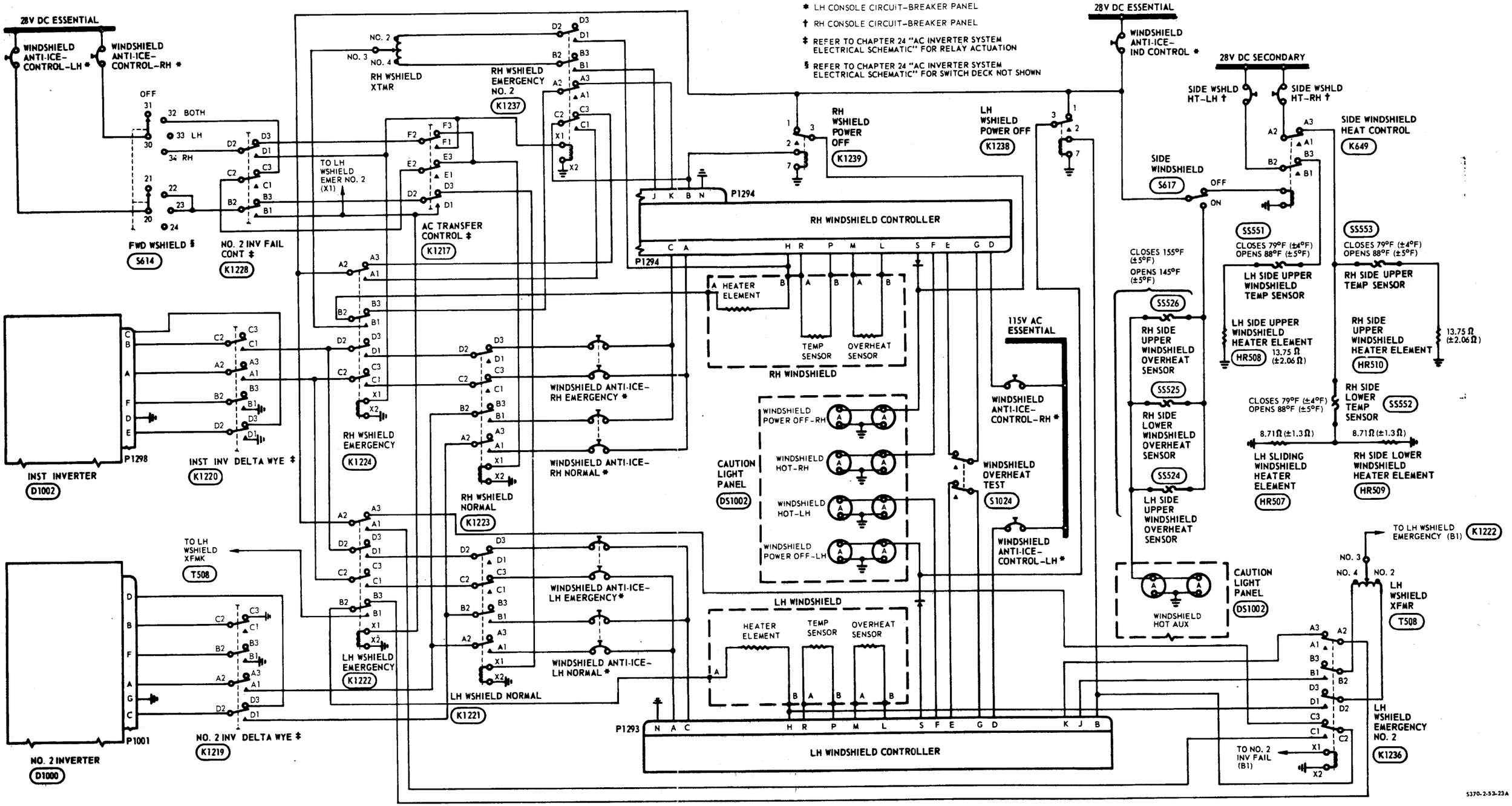
A push-button test switch is provided in the cockpit to test the windshield overheat protection circuit in each windshield temperature controller. When the test switch is depressed, both windshield overheat caution lights should come on.

2. Windshield DC-powered Anti-icing System

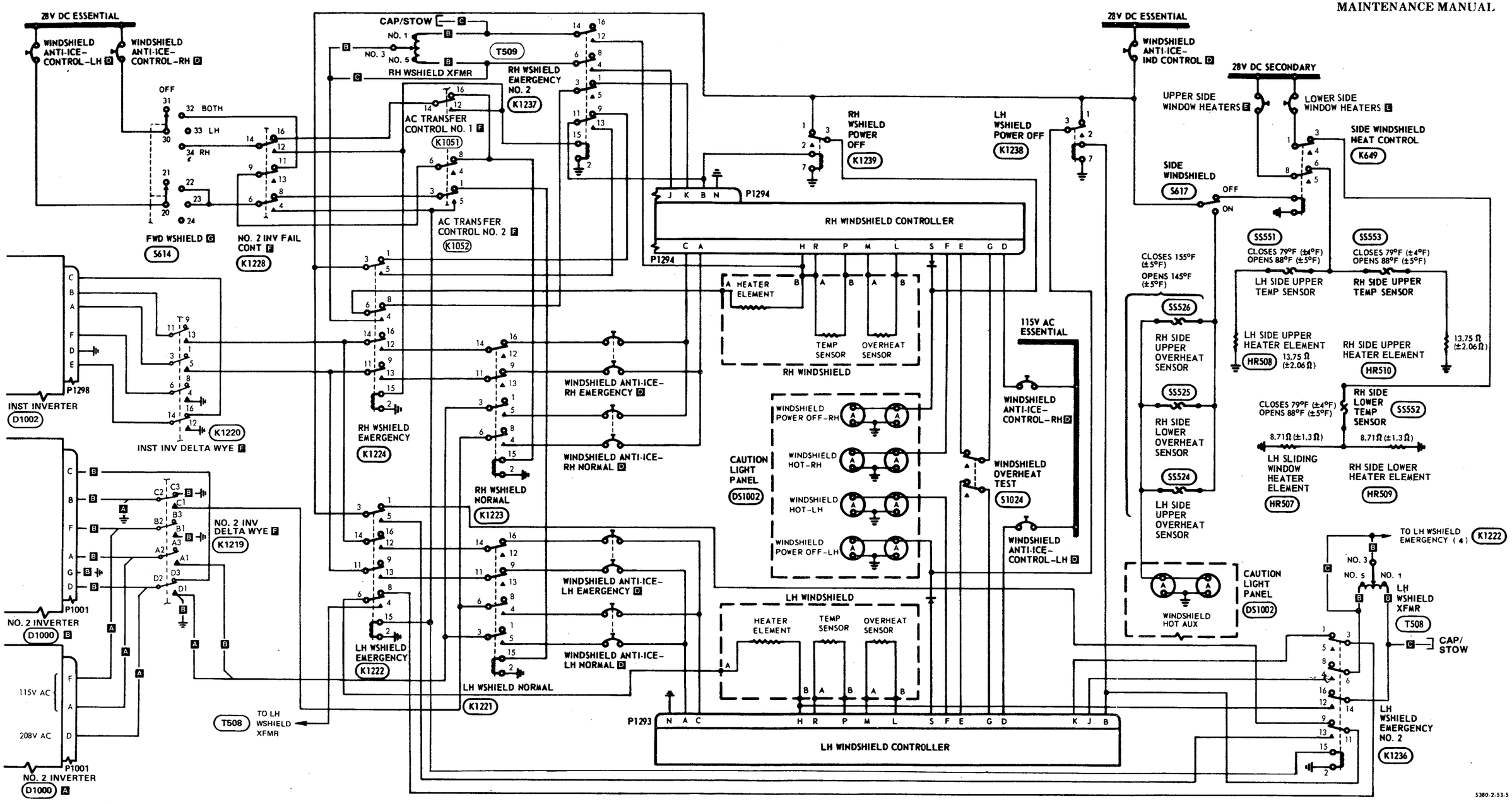
The pilot's and copilot's side windshields and the right and left overhead windshields are equipped with conductive-film type heating elements. Anti-icing, antifogging, and bird-impact-proofing are accomplished by maintaining the right and left overhead windshield and the pilot's and copilot's side windshield vinyl laminate at 75°F (±5°F).

DC power for the right and left overhead windshields, copilot's side windshield and pilot's sliding windshield, is supplied by the secondary 28-volt dc bus. These windshield panel heating elements are an integral conductive coating on the inner face of the outside acrylic laminate next to a vinyl interlayer. Two temperature-sensing units are embedded in the right and left overhead windshields and copilot's side windshield. One unit controls the panel heat; the other, because of overheat, illuminates a windshield overheat caution light on the caution-warning light panel. The pilot's sliding windshield has no embedded temperature-sensing units. This panel heater is controlled by the heat control temperature-sensing unit that is embedded in the copilot's side windshield. There is no overheat indication for the pilot's sliding windshield. (See figure 30-40-1 or 30-40-1A as applicable.)

The four panels are controlled by the "SIDE WSHIELD" anti-ice control switch simultaneously; there is no panel selectivity. Actuating the "SIDE WSHIELD" switch to ON de-energizes the windshield heat control relay (K649), allowing power from the secondary 28-volt dc bus to energize the heaters and the heat control temperature-sensing units. The primary 28-volt dc bus energizes the overheat temperature-sensing unit. The heat control temperature-sensing unit opens at 88°F (±5°F) and closes at 79°F (±4°F). The overheat temperature-sensing unit closes at 155°F (±5°F) and opens at 145°F (±5°F).



Windshield Anti-icing System Electrical Schematic
NA-370-1 Through -9
Figure 30-40-1



NOTES

A AIRPLANES WITH SOLID STATE INVERTERS

B AIRPLANES WITHOUT SOLID STATE INVERTERS

C AIRPLANES WITH SOLID STATE INVERTERS AFTER S/B 40

D LH CONSOLE CIRCUIT-BREAKER PANEL

E LH CONSOLE CIRCUIT-BREAKER PANEL

F REFER TO CHAPTER 24 "AC INVERTER SYSTEM ELECTRICAL SCHEMATIC" FOR RELAY ACTUATION

G REFER TO CHAPTER 24 "AC INVERTER SYSTEM ELECTRICAL SCHEMATIC" FOR SWITCH DECK NOT SHOWN

Windshield Anti-icing System Electrical Schematic
S/N 380-1 and Subs
Figure 30-40-1A

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MAJOR CHANGE

WINDSHIELD ANTI-ICING SYSTEM - TROUBLE SHOOTING

1. Trouble Shooting Windshield Anti-icing System

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. WINDSHIELD PANELS NOT HEATING.		
(1) Open or shorted windshield temperature sensing element.	See figure 30-40-1 or -1A for ohmic values; then check heating elements and temperature-sensing elements for an open or shorted circuit, using an ohmmeter.	Replace applicable panel. To replace windshield panel, refer to Chapter 56.
(2) Defective windshield temperature control.	Perform testing and adjusting windshield Anti-ice System using the Leland TEH-209-4 or -5 windshield control system tester.	Replace temperature control.
(3) Open conductive film-type windshield heating element.	Perform testing and adjusting windshield Anti-ice System using the Leland TEH-209-4 or -5 windshield control system tester.	If heating element is open, replace windshield panel. (Refer to Chapter 56.)
(4) Defective ac inverter.	Check ac inverter system operation. (Refer to Chapter 24.)	
(5) Defective wiring from ac inverters to controller.	Perform continuity check. (See figure 30-40-1 or 30-40-1A.)	Repair electrical wiring as necessary.
B. WINDSHIELD HOT CAUTION LIGHT COMES ON.		
NOTE: The following trouble shooting information is applicable for any or all of the three windshield hot caution lights, unless specified.		
(1) Temperature sensor element faulty.	Check sensor elements for open circuit, using an ohmmeter.	Replace windshield panel.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
(2) Overheat temperature-sensing element faulty.	Check sensors in dc-power-heated panels for closed contacts. Check sensors in ac-power-heated panels for resistance of 300 (±15) ohms.	Replace windshield panel.
(3) Heat control temperature-sensing element in windshield heated by dc-powered heater defective.	With "SIDE WSHIELD" switch at ON, check that control temperature-sensing elements contacts open (no 28-volt dc power at terminal "B") when windshield temperature reaches 88°F (±5°F).	Replace windshield panel.
(4) Overheat sensor element in dc-power-heated windshields closes below desired temperature of 155°F (±5°F).	Check overheat sensor element for closed contacts.	Replace windshield panel.
C. OVERHEAT CONDITION NOTED ON AC-POWER-HEATED WINDSHIELD, BUT WINDSHIELD OVERHEAT CAUTION LIGHT DOES NOT ILLUMINATE.		
(1) Defective overheat temperature-sensing element.	Check sensor element in windshield for resistance of 300 (±15) ohms.	Replace windshield panel if sensor element resistance is not within limits.

1. Postinstallation and Test Adjustments

A. Windshield Resistance Check

CAUTION: WHENEVER A WINDSHIELD IS REPLACED, THE FOLLOWING WINDSHIELD HEATING ELEMENT RESISTANCE CHECK MUST BE MADE TO DETERMINE WHICH TRANSFORMER TAP IS TO BE UTILIZED. IF THE PROPER TRANSFORMER TAP IS NOT CONNECTED, THE NEW WINDSHIELD MAY BE DAMAGED.

The resistance of the heating element in the windshield determines the terminal tap to be utilized on the windshield transformer. This resistance will vary from one windshield to another. Whenever a windshield is replaced, the heating element resistance should be checked with a bridge-type ohmmeter to determine this value:

WINDSHIELD RESISTANCE - OHMS	TRANSFORMER TAP	VAC
23.3 to 26.0	2	136
26.0 to 27.9	3	142
27.9 to 29.7	4	148

B. Preliminary System Check

Before the system is operated after replacement of any components, the inverter system and windshield heating system wiring and connections should be checked for proper installation and continuity. Also, the inverter ac power output should be checked for proper values.

C. Windshield Overheat and Temperature Sensor Check

Each windshield has one overheat sensor and one temperature control sensor. The resistance of each sensor should be checked with an ohmmeter; resistance should be approximately 300 (±15) ohms at 60°F, and should be above ground with no shorted connections. Refer to figure 30-40-3 for resistance values at temperatures above or below 60°F.

NOTE: The resistance of the sensors will vary with the temperature and will have a wide tolerance at normal ambient temperature.

D. Windshield System Caution Light Check

The resistance and proper polarity of the diodes in the windshield system caution light circuits may be checked with an ohmmeter as follows: Disconnect plug P1293 from left windshield controller. Connect negative lead of ohmmeter to pin "F" and positive lead to pin "S" of P1293. The ohmmeter should read approximately 600 to 800 ohms. When the polarity of the ohmmeter connections is reversed, the reading should be infinity. Disconnect plug P1294 from right windshield controller and make the same check on pins "S" and "F". The results should be the same.

NOTE: The polarity of the instrument used must be known to prevent a false reading. This check was performed using a Simpson 260 VOM set on the R X 10 scale.

E. Inverter Phase Rotation Check

The inverter phase rotation and proper connections may be checked with a phase sequence indicator as follows: Connect phase sequence indicator to left controller disconnect plug P1293 with phase A to pin "A," phase B to pin "C," and phase C to phase C output of inverter. The phase sequence may be checked with either the No. 2 inverter or the instrument inverter operating to supply windshield heating power.

2. Testing and Adjusting Windshield Anti-icing System the Leland TEH-209-4 or -5 Windshield Control System Tester

A. To properly check and adjust the windshield anti-icing system controllers (Garrett P/N 607936-1-1), the Leland Airborne Products TEH-209-4 or -5 control tester is used. A load bank is provided with the tester. An additional locally manufactured cable (figure 30-40-2) is also required to connect the tester to the system controllers. The generator control switches and indicator lights on the tester are identified as follows:

(1) The Generator Off-On Switches are identified as "PILOT" and "CO-PILOT" Generator Switches. These switches control power from the airplane to the respective LH and RH windshield temperature controllers.

(2) The Generator Fail Lights when illuminated are used to indicate the respective LH or RH "WINDSHIELD POWER OFF".

(3) The Generator Overheat lights when illuminated, are used to indicate the respective LH or RH WINDSHIELD OVERHEAT" condition.

B. To connect the TEH-209-4 or -5 windshield control system tester, proceed as follows:

(1) Place tester near nose of airplane, then remove load bank and position so that heat radiated from load bank is directed away from airplane.

(2) Disconnect electrical connectors from controllers, then connect locally manufactured cable (figure 30-40-2) to controllers.

(3) Check that "CONTROLLER POWER" and "GENERATOR" switches on tester are off, then connect trunk end of cable to tester.

(4) Disengage (Pull-out) the "INSTRUMENT INVERTER" circuit breakers.

CAUTION: If the preceding step is not accomplished, damage to the system may result.

(5) Apply external dc electrical power to airplane.

(6) Place tester controls as follows:

(a) "MODE" switch at "MONITOR".

(b) "METER FUNCTION" switch to "CO-PILOT DC-IN".

(c) "CONTROLLER" switch at ON.

(7) Check that controller power indicator light is on, and that voltmeter reads 24 to 30 volts, dc.

C. Checking Windshield Temperature Control Sensor Resistance.

(1) To check left windshield temperature control sensor resistance, position tester controls as follows:

(a) "MODE" switch at "PILOT TEST".

(b) "METER FUNCTION" switch at "PILOT SENSOR".

(c) Position "AUX SENSOR" control to center null meter, then read sensor resistance from "AUX SENSOR" dial.

(2) Using the resistance valve obtained in step (C), see figure 30-40-3, for proper resistance to temperature reading.

(3) To check right windshield temperature control sensor resistance, position tester controls as follows:

(a) "MODE" switch at "CO-PILOT TEST".

(b) "METER FUNCTION" switch at "CO-PILOT SENSOR".

(c) Position "AUX SENSOR" control to center null meter, then read sensor resistance from "AUX SENSOR" dial.

(4) Using the resistance value obtained in step (c) see figure 30-40-3, for proper resistance to temperature reading.

D. Checking Windshield Heater Element Resistance

(1) To check left windshield heater element resistance, position tester controls as follows:

(a) "MODE" switch at "PILOT TEST".

(b) "METER FUNCTION" switch at "PILOT HEATER RES".

(c) Position "AUX SENSOR" control to center null meter, then read heater resistance from "AUX SENSOR" dial.

(2) Using the resistance value obtained in step (c), after ignoring the first digit of the reading (i.e., a reading of 325.5 will be 25.5) the resistance must be between 23.0 and 30.5 ohms.

(3) To check right windshield heater element resistance, position tester controls as follows:

(a) "MODE" switch at "CO-PILOTS TEST".

(b) "METER FUNCTION" switch at "CO-PILOTS HEATER RES".

(c) Position "AUX SENSOR" control to center null meter, then read heater resistance from "AUX SENSOR" dial.

(4) Using the resistance value obtained in step (c), after ignoring the first digit of the reading (i.e., a reading of 325.5 will be 25.5) the resistance must be between 23.0 and 30.5 ohms.

E. Windshield System Operational Check-out

(1) To check operating function of the controllers, position airplane and tester switches as follows:

(a) Place "ELECTRICAL MASTER" switch at ON.

(b) Place "BATTERY" switch at ON.

(c) Place "INVERTER" switch at NO. 1.

(d) Ensure that "CONTROLLER" switch is at OFF.

(2) To check left windshield system, proceed as follows:

(a) Place "FWD WSHIELD" switch to LH.

(b) Place tester "MODE" switch to "PILOT TEST" and adjust "AUX SENSOR" control to 330 ohms.

(c) Place "CONTROLLER" switch at ON. Check that controller power indicator light is on.

(d) Place "METER FUNCTION" switch to "PILOT DC-IN". Check for 24 to 30 volts.

(e) Place "METER FUNCTION" switch to "PILOT AC-IN". Check for 190 to 205 volts.

(f) Place "METER FUNCTION" switch to "PILOT AC-OUT".

(g) Place "PILOT GENERATOR" switch at ON. The volt meter reading slowly increases to a full output (190 volts minimum) in a period of 100 ± 20 seconds. Check that "WINSHIELD PWR OFF-LH" light is out.

(h) Momentarily place "SENSOR" switch at SHORTED and at OPEN. In either position, the ac voltage shall indicate zero volts.

(i) Adjust "AUX SENSOR" control to 336 ohms. AC voltage should be 50% ± 10% of the voltage reading in step (g). This reading should be 120 to 155 volts.

(j) Adjust "AUX SENSOR" control to 333.5 (cold windshield simulation) ohms. Ensure that full ac voltage output is achieved within ± 1 ohm of this setting (i.e. setting between 332.5 ohms to 334.5 ohms).

(k) Adjust "AUX SENSOR" control to 338.5 (hot windshield simulation) ± 1 ohm. Check that AC voltage drops to zero.

(l) Slowly reduce resistance by adjusting the "AUX SENSOR" control. Complete AC voltage should be restored when helipot resistance is below 333.5 ohms.

(m) Adjust "AUX SENSOR" control to 336 ohms. Check that AC voltage is 120 to 155 volts.

(n) Loosen the locking device on the LH windshield controller overheat adjustment control, then rotate control counter clockwise until "LH GENERATOR OVERHEAT" light comes on. Rotate control clockwise until light goes out. Monitor this position carefully, then rotate control clockwise 3.5 increments of the dial divisions.

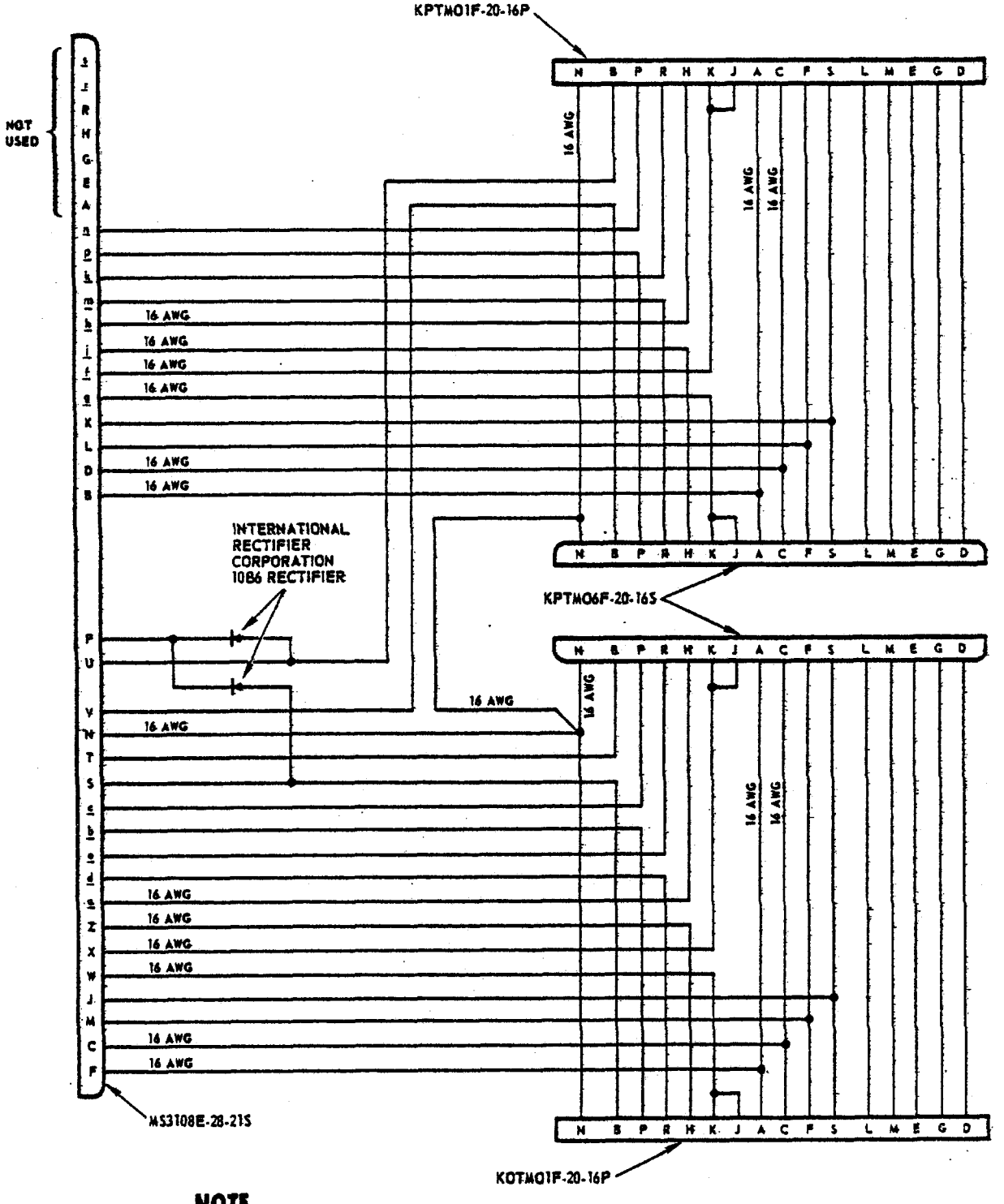
(o) Engage overheat adjustment control locking device.

(p) Depress "WINDSHIELD OVERHEAT TEST" switch (in airplane or on the LH controller). Check that "LH GENERATOR OVERHEAT" light comes on and that AC voltage drops to zero.

(3) To check right windshield system, repeat steps (a) thru (q) of E. (2). When checking the right hand system, all tester switch position will be CO-PILOT or RH as applicable.

(4) Remove external power from airplane.

(5) Disconnect tester cables from airplane and connect airplane wiring to controllers. Close access doors.



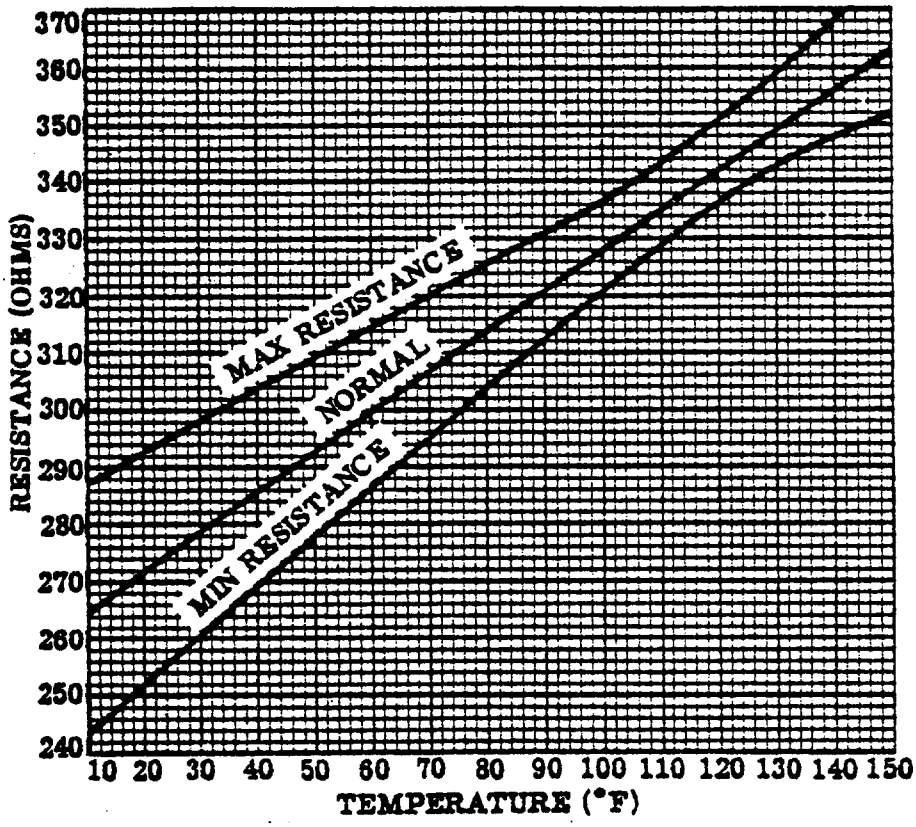
NOTE

USE NUMBER 20 AWG WIRE
UNLESS OTHERWISE SPECIFIED

5370-2-53-15

Windshield Temp Control Test Cable
Figure 30-40-2

ORIGINAL
As Received By
ATP



The graph is solely for purposes of confirming continuity of sensor and related wiring. Determination of exact windshield temperature requires conditions and equipment not normally available at flight line maintenance and is not necessary, as controller calibration and limit resistance values are well within the tolerance range of all windshields.

Windshield Sensor Versus Windshield Temperature
Conversion Chart
Figure 30-40-3

1. Operational Check-out of Windshield Anti-Icing System-NA370-2 thru -6 -8 and -9 Airplanes

A. Check-out of Windshield Anti-Icing System

The Windshield Anti-Icing System check-out procedures consists of applying or removing electrical power to the windshield heating and inverter system. This will ensure that the applicable caution lights will illuminate and that the heating elements in the windshield are operating properly. If it is found that the inverter change-over system is at fault, refer to Chapter 24.

(1) Position all switches at OFF or in airplane-parked position. Engage circuit breakers.

(2) Connect a 28-volt dc electrical power source to airplane.

(3) Place "INVERTER" switch at No. 1. Check that "AC INST PWR OFF" and "NO. 1 INV OFF" lights go out.

(4) Place "FWD WSHIELD" switch at LH. Check that "NO. 2 INV OFF"and "WINDSHIELD PWR OFF - LH" lights go out.

(5) Place "FWD WSHIELD" switch at RH. Check that "WINDSHIELD PWR OFF - LH" light comes on and the windshield PWR OFF - RH light goes out.

NOTE: Ensure that the windshield panel temperature is below 100°F before performing the following step. This will allow the mechanic to feel the temperature rise of both windshield panels.

(6) Place "FWD WSHIELD" switch at BOTH. Check that "WINDSHIELD PWR OFF - LH" goes out.

(7) Depress "WINDSHIELD OVERHEAT TEST" switch. Check that "WINDSHIELD OVERHEAT" light comes on. Release switch.

(8) Check for heat on exterior surface of both RH and LH windshield panels.

(9) Disengage (pull out) "NO 1 INV" circuit breaker. Check that "NO. 1 INV OFF" and WINDSHIELD PWR OFF-RH lights come on.

(10) Place "FWD WSHIELD" switch to LH. Caution lights should remain the same as described Step (9).

(11) Place "FWD WSHIELD" switch to RH. Check that "WSHIELD PWR OFF-RH" light goes out, and that "WSHIELD PWR OFF-LH" light comes on.

(12) Disengage (pullout) "INVERTERS-PWR TRIP - NO. 2" circuit breaker. Check that "WINDSHIELD PWR OFF - LH" and "NO. 2 INV OFF" lights come on.

(13) Place "FWD WSHIELD" switch to OFF.

(14) Engage (push in) "NO. 1 INV" and INVERTERS-PWR TRIP - NO. 2" circuit breakers. Check that "NO. 1 INV OFF" and "NO. 2 INV OFF" lights go out.

(15) Place "INVERTER" switch at NO. 2. Check that "NO. 1 INV OFF" light comes on.

(16) Place "FWD WSHIELD" switch to BOTH. Check that "WINDSHIELD PWR OFF-RH" light comes on.

(17) Disengage (pull out) "INVERTERS-PWR TRIP-NO. 2" circuit breaker. Check that "NO. 1 INV OFF","NO. 2 INV OFF", "AC INST PWR" and WINDSHIELD PWR OFF - RH lights come on.

(18) Place all switches at OFF or in airplane-parked position. Engage circuit breakers.

(19) Close all applicable access doors (if opened) and disconnect external dc electrical power.

2. Operational Check-out of Windshield Anti-Icing System-NA380 Airplanes

A. Check-out of Windshield Anti-Icing System

The Windshield Anti-Icing System check-out procedures consists of applying or removing electrical power to the windshield heating and inverter system. This will ensure that the applicable caution lights will illuminate and that the heating elements in the windshield are operating properly. If it is found that the inverter change-over system is at fault, refer to Chapter 24.

(1) Position all switches at OFF or in airplane-parked position. Engage circuit breakers.

(2) Connect external electrical power to airplane.

(3) Place communication and navigation equipment switches to OFF.

(4) Place INVERTER switch to OFF. Set FWD WSHIELD rotary switch to OFF. Check the following indicating lights on the caution-warning light panel.

#1 INV OFF - Illuminated
#2 INV OFF - Illuminated
AC INST PWR OFF - Illuminated
WINDSHIELD PWR OFF LH - Illuminated
WINDSHIELD PWR OFF RH - Illuminated
WINDSHIELD HOT LH - Out
WINDSHIELD HOT RH - Out

(5) Place INVERTER switch to NO. 1. Set FWD WSHIELD rotary switch to BOTH. Check the following indicating lights:

#1 INV OFF - Out
#2 INV OFF - Out
AC INST PWR OFF - Out
WINDSHIELD PWR OFF LH - Out
WINDSHIELD PWR OFF RH - Out

(6) Momentarily depress WSHIELD OVHT TEST switch. Check the following indicating lights:

WINDSHIELD HOT LH - Illuminated
WINDSHIELD HOT RH - Illuminated

(7) Place FWD WSHIELD rotary switch to LH. Check the following indicating lights.

WINDSHIELD PWR OFF LH - Out
WINDSHIELD PWR OFF RH - Illuminated

(8) Place FWD WSHIELD rotary switch to RH. Check the following indicating lights:

WINDSHIELD PWR OFF LH - Illuminated
WINDSHIELD PWR OFF RH - Out

(9) Place FWD WSHIELD rotary switch to BOTH. Open NO. 1 INVERTER TRIP circuit breaker (2 amp). Check the following indicator lights:

#1 INV OFF - Illuminated
#2 INV OFF - Out
AC INST PWR OFF - Out
WINDSHIELD PWR OFF LH - Out
WINDSHIELD PWR OFF RH - Illuminated

NOTE: During inverter transfer the 3 inverter fail lights will illuminate for approximately 5 seconds.

(10) Open No. 2 TRIP-INVERTER circuit breaker (2 amp). Check the following indicating lights:

#1 INV OFF - Illuminated
#2 INV OFF - Illuminated
AC INST PWR OFF - Out
WINDSHIELD PWR OFF LH - Illuminated
WINDSHIELD PWR OFF RH - Illuminated

(11) Place INVERTER switch to NO. 2. Check the following indicating Lights:

#1 INV OFF - Illuminated
#2 INV OFF - Illuminated

AC INST PWR OFF - Illuminated
WINDSHIELD PWR OFF LH - Out
WINDSHIELD PWR OFF RH - Illuminated

(12) Place FWD WSHIELD rotary switch to RH. Check the following indicating lights:

WINDSHIELD PWR OFF LH - Illuminated
WINDSHIELD PWR OFF RH - Out

(13) Place INVERTER switch to OFF. Set FWD WSHIELD rotary switch to OFF.

(14) Close NO. 1 INVERTER TRIP and NO. 2 TRIP-INVERTER circuit breakers.

(15) Turn off external electrical power and disconnect if required.

WINDSHIELD WIPER SYSTEM - DESCRIPTION AND OPERATION

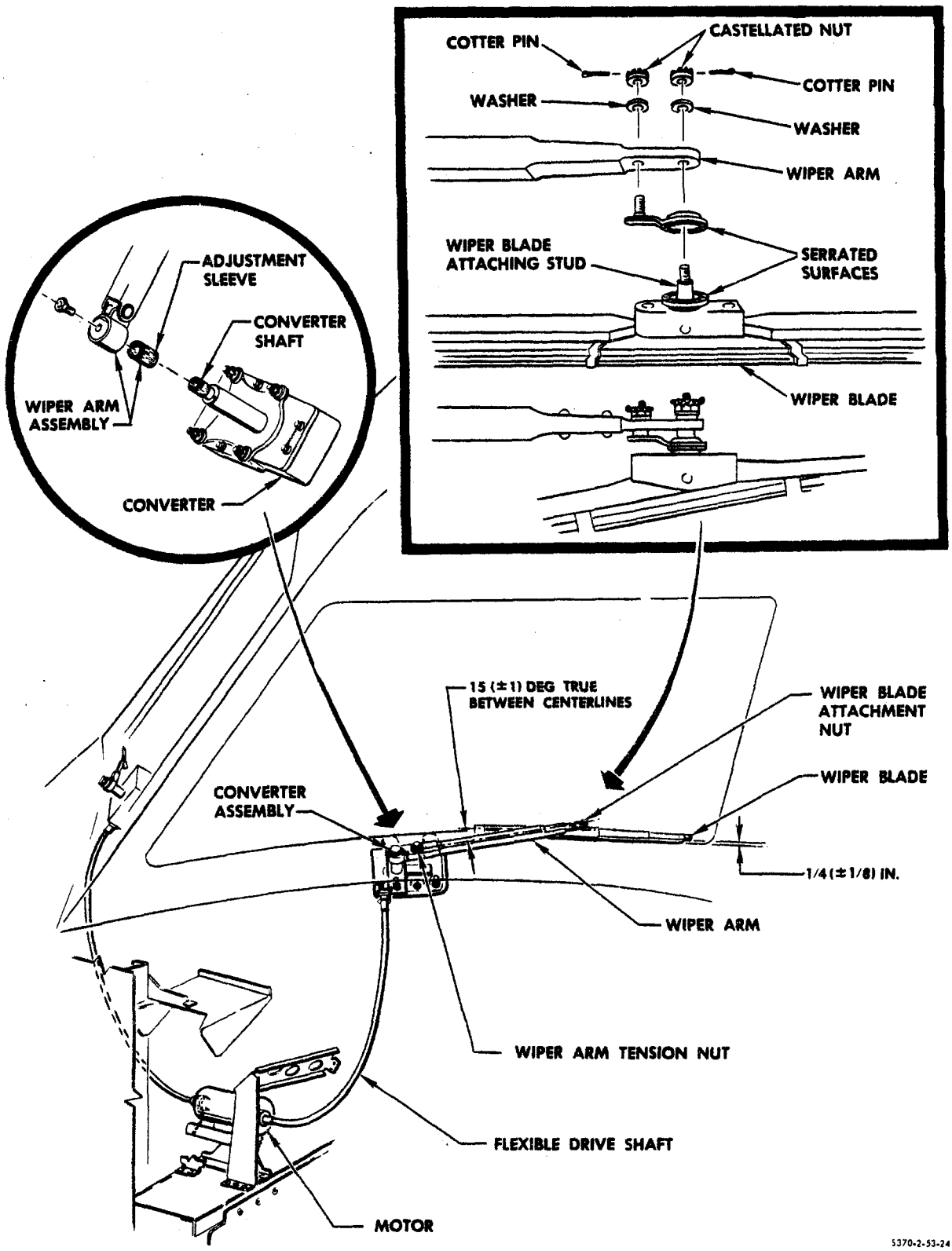
1. General

The windshield wiper system (figure 30-41-1) consists of windshield wiper arm and blade assemblies for both the pilot and copilot windshields. The system is actuated by an electric motor and controlled by a five-position switch. Flexible drive shafts connect the motor to the converter units. An automatic reset thermo-protector, inside the motor, temporarily de-energizes the motor in case of an overload. Radio noise filters are in the motor base. The motor incorporates a gear train that actuates a cam switch to automatically stow both wipers when the windshield wiper selector switch is turned to PARK.

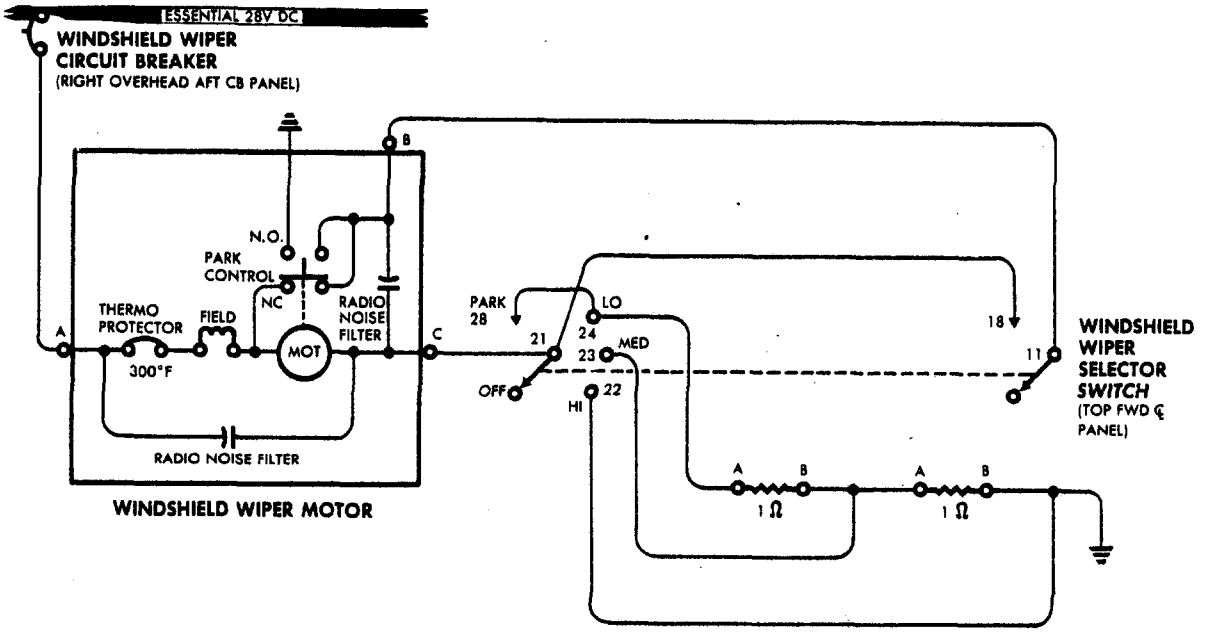
The converter units change the rotary motor power to a reciprocating power to actuate the windshield wiper arms and wiper blades. The wiper arms and flexible wiper assemblies wipe an area sufficient for visibility during taxiing and at flight speeds up to 180 knots IAS. The dual wiper installation wipes a rectangular shaped area equal to an approximate 90-degree segment of a circle. The two wiper assemblies are stowed at the bottom of the glass.

The windshield wiper system (figure 30-41-2) is controlled by a five-position switch on the overhead control panel in the cockpit. The five positions are PARK, OFF, HIGH, MED, and LOW. The system is powered by the essential dc bus with a 10-ampere circuit breaker on the cockpit circuit-breaker panel.

Turning the windshield wiper selector switch to LOW, HIGH, or MED will cause the windshield wipers to operate at low, high, or medium speed. With the selector switch at OFF position, the wipers are inoperative. Moving of the selector switch to PARK will actuate the wiper arms to move down on the windshield to a stowed position. The PARK position of the selector switch is a momentary contact. After the wiper arms are in the stowed position, releasing the selector switch allows the switch to return to the OFF position and returns the windshield wiper motor park control to the NC position.



Windshield Wiper System
Figure 30-41-1



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Windshield Wiper System Electrical Schematic
Figure 30-41-2

WINDSHIELD WIPER SYSTEM - TROUBLE SHOOTING

1. Trouble Shooting Windshield Wiper System

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. SYSTEM INOPERATIVE.		
(1) Motor inoperative.	Electric motor should have 28-volt dc on terminal "A" and, with windshield wiper selector switch in LOW, MED, or HIGH, motor terminal "B" should be grounded.	Replace motor.
	Using a multimeter, check resistance between pin "A" and pin "C." Zero resistance indicates a shorted filter condenser. Infinite resistance indicates an open field, an open armature, or an open thermo-protector, or a combination of any of the three items.	Replace motor.
B. SYSTEM INOPERATIVE IN HIGH, LOW, OR MED SPEEDS.		
(1) Wiring open circuit.	Check wiring from windshield selector switch to ground.	Correct open circuit.
(2) Defective resistor in MED or LOW speed.	Check each resistor for one-ohm resistance.	Replace defective resistor.
(3) Defective switch	Check switch operation.	Replace defective switch.

WINDSHIELD WIPER SYSTEM - MAINTENANCE PRACTICES

1. Replacement of Windshield Wiper Blade

   A. Removing Windshield Wiper Blade

      (1) See figure 30-41-1; then loosen wiper arm tension nut.

      (2) Remove cotter pin, nut, and washers from wiper blade attaching stud; then remove blade.

   B. Installing Windshield Wiper Blade

      (1) See figure 30-41-1; then, with wiper arm in park position, place wiper blade attach bolt through wiper arm mounting hole. Position wiper blade on windshield glass, parallel with windshield frame.

      (2) Make sure both serrated surfaces are properly positioned together; then tighten nut. Torque nut (17 to 25 inch-pounds as required) to align nut castellations with cotter-pin hole. Install cotter pin.

      (3) Tighten wiper arm tension nut until wiper blade is held snug against windshield glass. (Refer to check-out procedures for windshield wiper system.)

2. Replacement of Windshield Wiper Arm

   A. Removing Windshield Wiper Arm

      (1) See figure 30-41-1; then loosen wiper tension nut.

      (2) Cut lockwire and remove bolt at wiper arm hub.

      (3) Lift arm and adjustment sleeve from converter shaft.

   B. Installing and Adjusting Windshield Wiper Arm

      (1) With converter in PARK position, install adjustment sleeve in arm and place on converter shaft so that arm will be as close to desired position as possible. (See figure 30-41-1.)

      (2) Adjust wiper arm, using adjustment sleeve outer serrations to position wiper blade 1/4 (±1/8) inch from edge of glass.

         NOTE: The blade assembly should not wipe the glass beyond any portion of the full stroke or PARK position.

      (3) Install wiper arm hub bolt, and safety with MS20995N20 lockwire.

      (4) Readjust wiper blade, as required, and tighten wiper arm tension nut until wiper blade is held snug against windshield glass. (Refer to check-out procedures for windshield wiper system.)

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3. Replacement of Converter Unit

A. Removing Converter Unit

(1) Remove windshield wiper arm. (Refer to procedure for removing windshield wiper arm.)

(2) Cut lockwire and disconnect flexible drive shaft from windshield wiper motor.

(3) Remove attaching bolts and remove converter assembly.

B. Installing Converter Unit

(1) Place converter unit in place with wiper arm shaft extending outside of fuselage attaching bolts. (See figure 30-41-1.)

(2) Install drive shaft, tightening nut finger-tight. Safety drive shaft nut, using MS20995N20 lockwire.

CAUTION: FLEX DRIVE IS SUSCEPTIBLE TO KINKING. USE EXTREME CAUTION WHEN INSTALLING FLEX DRIVE, TO PREVENT THIS CONDITION. OVERHEATING WILL RESULT IF DRIVE IS KINKED.

(3) Install windshield wiper arm. (Refer to procedure for installing windshield wiper arm.)

4. Replacement of Windshield Wiper Motor

A. Removing Windshield Wiper Motor

(1) Cut lockwire on drive shafts and disconnect right and left drive shafts.

(2) Disconnect electrical plug from windshield wiper motor.

(3) Cut lockwire and remove attach bolts. Remove motor.

B. Installing Windshield Wiper Motor

(1) Place motor into position and install four attach bolts. Safety attach bolts with lockwire. (See figure 30-41-1.)

(2) Connect electrical plug.

(3) Check windshield wiper arms for equal alignment and install drive shafts, tightening nuts finger-tight. Safety drive shaft nuts, using MS20995N20 lockwire.

CAUTION: FLEX DRIVE IS SUSCEPTIBLE TO KINKING. USE EXTREME CAUTION WHEN INSTALLING FLEX DRIVE, TO PREVENT THIS CONDITION. OVERHEATING WILL RESULT IF DRIVE IS KINKED.

(4) Test operation of windshield wiper system. (Refer to procedure for checking windshield wiper system.)

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1. Operational Check-out of Windshield Wiper System

A. Check-out of Windshield Wiper System

CAUTION: DO NOT OPERATE WIPERS ON DRY WINDSHIELDS (PLASTIC OR GLASS), AS THE WINDSHIELD COULD BECOME SCRATCHED. WHEN IT IS NOT POSSIBLE TO SPRAY WATER ON WINDSHIELD PANELS, INSTALL A THIN, CLEAN PROTECTIVE COVER BETWEEN WIPER BLADE AND PANEL. TO PREVENT WIPER BLADE DAMAGE, DO NOT RUN WIPERS CONTINUOUSLY ON THIS COVER.

- DO NOT OPERATE WINDSHIELD WIPERS CONTINUOUSLY ON LOW SPEED (5 TO 10 MINUTES MAXIMUM).

(1) Connect external dc power to airplane bus system. Make sure windshield wiper circuit breaker is engaged.

(2) Move windshield wiper slector switch to LOW. Wiper arms will operate at low speed.

(3) Move windshield wiper selector switch to MED. Wiper arms will operate at a medium speed.

(4) Move windshield wiper selector switch to HIGH. Wiper arms will operate at a high speed.

(5) Move windshield wiper selector switch to OFF. Windshield wipers will stop.

(6) Hold switch to PARK position. The wiper arms will move down the windshield to stowed position.

(7) Check windshield wiper blade to see that it is snug against the windshield glass.

(8) Check movement of wiper blade for a sweep area in equal relation to windshield frame.

B. Checking and Adjusting Windshield Wiper System

CAUTION: DO NOT OPERATE WIPERS ON DRY WINDSHIELDS (PLASTIC OR GLASS), AS THE WINDSHIELD COULD BECOME SCRATCHED. WHEN IT IS NOT POSSIBLE TO SPRAY WATER ON WINDSHIELD PANELS, INSTALL A THIN, CLEAN PROTECTIVE COVER BETWEEN WIPER BLADE AND PANEL. TO PREVENT WIPER BLADE DAMAGE, DO NOT RUN WIPERS CONTINUOUSLY ON THIS COVER.

- DO NOT OPERATE WINDSHIELD WIPERS CONTINUOUSLY ON LOW SPEED (5 TO 10 MINUTES MAXIMUM).

(1) Make sure windshield is sprayed with water, or a thin, clean protective cover is between wiper blade and windshield panel.

(2) With essential bus power on, engage windshield wiper circuit breaker. Remove copilot's windshield wiper arm and wiper blade.

(3) Loosen adjusting screw at pilot's wiper arm hub until pressure against windshield is about 2 pounds.

(4) On pilot's side, watch wiper blade oscillate back and forth, noting arc of travel. Turn windshield wiper switch to OFF so that wiper blade stops at about the end of its stroke. Either end is satisfactory.

(5) Disconnect flex drive at motor and rotate shaft until wiper blade reaches end of downstroke.

NOTE: There is no blade movement during about one shaft rotation at the end of the stroke. After this one shaft rotation, wiper blade travel will reverse.

(6) Turn windshield wiper selector switch to PARK.

(7) Reconnect flex drive at motor. Safety with MS20995N20 lockwire.

(8) Adjust wiper blade to 1/4 (±1/8) inch from bottom of windshield panel, using following procedure:

(a) Measure distance that arm (at blade attach point) is from desired position.

(b) Determine required rotation of adjustment sleeve, using following table:

DISTANCE (INCHES)	NO. OF TEETH	DISTANCE (INCHES)	NO. OF TEETH
1/16	2	9/16	20
1/8	4	5/8	22
3/16	7	11/16	24
1/4	9	3/4	27
5/16	11	13/16	29
3/8	13	7/8	31
7/16	16	15/16	33
1/2	18	1	36

(c) See figure 30-41-1; then remove wiper arm from adjustment sleeve. (Do not disturb position of sleeve on converter shaft.)

(d) Rotate adjustment sleeve around converter shaft the determined number of teeth, in direction blade should move.

(e) Reinstall wiper arm.

(9) Reinstall copilot's windshield wiper arm and wiper blade. Repeat steps (3) through (8) to adjust copilot's wiper blade.

(10) Adjust pressure of each flexible wiper against glass by tightening adjusting screw at wiper arm hub. Pressure should be 7 (±1) pounds. This should be checked with a pull scale hooked under the wiper arm at the rubber wiper attachment. Read scale just as rubber lifts from panel.

(11) Operate wiper on wet glass for a minimum of 30 seconds, observing that it wipes within area desired to be cleared. (See figure 30-41-1.)

(12) Check wiper pressure again. If it has changed, repeat steps (10) and (11). Recheck wiper pressure, following this procedure until reading stabilizes.

(13) Simulating rain with a water spray, check wiper operation for satisfactory rain clearance at various speeds.

CAUTION: DO NOT OPERATE WIPERS ON DRY PANELS.

- DO NOT OPERATE WINDSHIELD WIPERS CONTINUOUSLY ON LOW SPEED (5 TO 10 MINUTES MAXIMUM).

(14) Turn windshield wiper selector switch through OFF TO PARK. On pilot's windshield, check that wiper stops in park position. The wiper should be 1/4 (±1/8) inch from the bottom of the windshield panel. (See figure 30-41-1.)

(15) If either wiper stops off park position, repeat steps (3) through (11).

(16) Safety flexible drive nut, using MS20995N20 lockwire. (Note hole in motor base for this purpose.)

(17) Complete all safety wiring. Disengage windshield wiper circuit breaker and position essential bus power switch to OFF.

AILERON CONTROL SECTORS AND SUPPORTS ANTI-ICING

MAINTENANCE PRACTICES

APPLICATION OF ICE ADHESION DEPRESSANT TO AILERON CONTROL SECTORS AND SUPPORTS

To reduce the possibility of icing during high-altitude or inclement-weather flight conditions, the aileron controls sectors (246-52314 and 246-52364) and support assemblies (265-523059 and 265-523060) should be treated with an ice adhesion depressant. An ice depressant is available for this purpose under the brand name of "ICEX II," manufactured by B.F. Goodrich Aviation Products.

Applying Ice Adhesion Depressant

NOTES

- ***Access to the sectors and supports is gained through the right landing gear wheel well, and by removing the 306-523007-3 sector guard.***
- ***Refer to Special Inspections, page 5-10-07 in Chapter 5 for application frequency.***

1. Clean sectors and supports to remove all dirt and grease. Do not clean with solvents which might damage paint or markings.
2. After all surfaces are thoroughly clean and dry, apply "ICEX II" on surfaces of sectors and supports. Application on leading edge de-icer boots (if installed) may be applied from a non-spray can by using a clean cloth.

CAUTION

Apply "ICEX II" sparingly. A heavy application will collect dust and result in reduced effectiveness.

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STALL WARNING LIFT TRANSDUCER VANE HEATING

1. General

On aircraft NA-370-2 through -6, -8 and -9, the stall warning system has two lift transducers (high- and low-altitude) recessed in the underside of the right wing. The sensing vanes on these lift transducers are exposed to outside atmospheric conditions and are electrically heated to prevent icing and to ensure proper stall warning system operation. On aircraft NA-370-1, -7, and NA-380-1 and later, two lift transducers are installed on the underside of both wings. (Refer to chapter 34.)

CHAPTER

31

INDICATING/ RECORDING SYSTEMS

CHAPTER 31

INSTRUMENTS

TABLE OF CONTENTS

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CHAPTER 31

LIST OF EFFECTIVE PAGES

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List of Effective Pages	Added	1*	Jul 6/73
31		1	Mar 1/70
31-00-0		1*	Jul 6/73
		2*	Jul 6/73
		3*	Jul 6/73
	Added	4*	Jul 6/73
31-00-1		201	Mar 1/70
		202	Mar 1/70
		203	Mar 1/70
		204	Mar 1/70

*The asterisk indicates pages revised, added or deleted by the current revision.

INSTRUMENTS - DESCRIPTION

1. General

There are five instrument panels. The pilot's outboard and inboard instrument panels, the copilot's outboard and inboard instrument panels, and the center instrument panel contain all the instruments except the stand-by compass, and the instruments found on the voltmeter panel. The stand-by compass is mounted on the windshield bow. The instruments are internally lighted.

There are two instrument panel vibrators, one for each of the two outboard instrument panels. The vibrators are attached directly to the panels and should be electrically operated at any time accurate instrument readings are desired. The vibrators prevent indicator needles from sticking and giving erroneous readings.

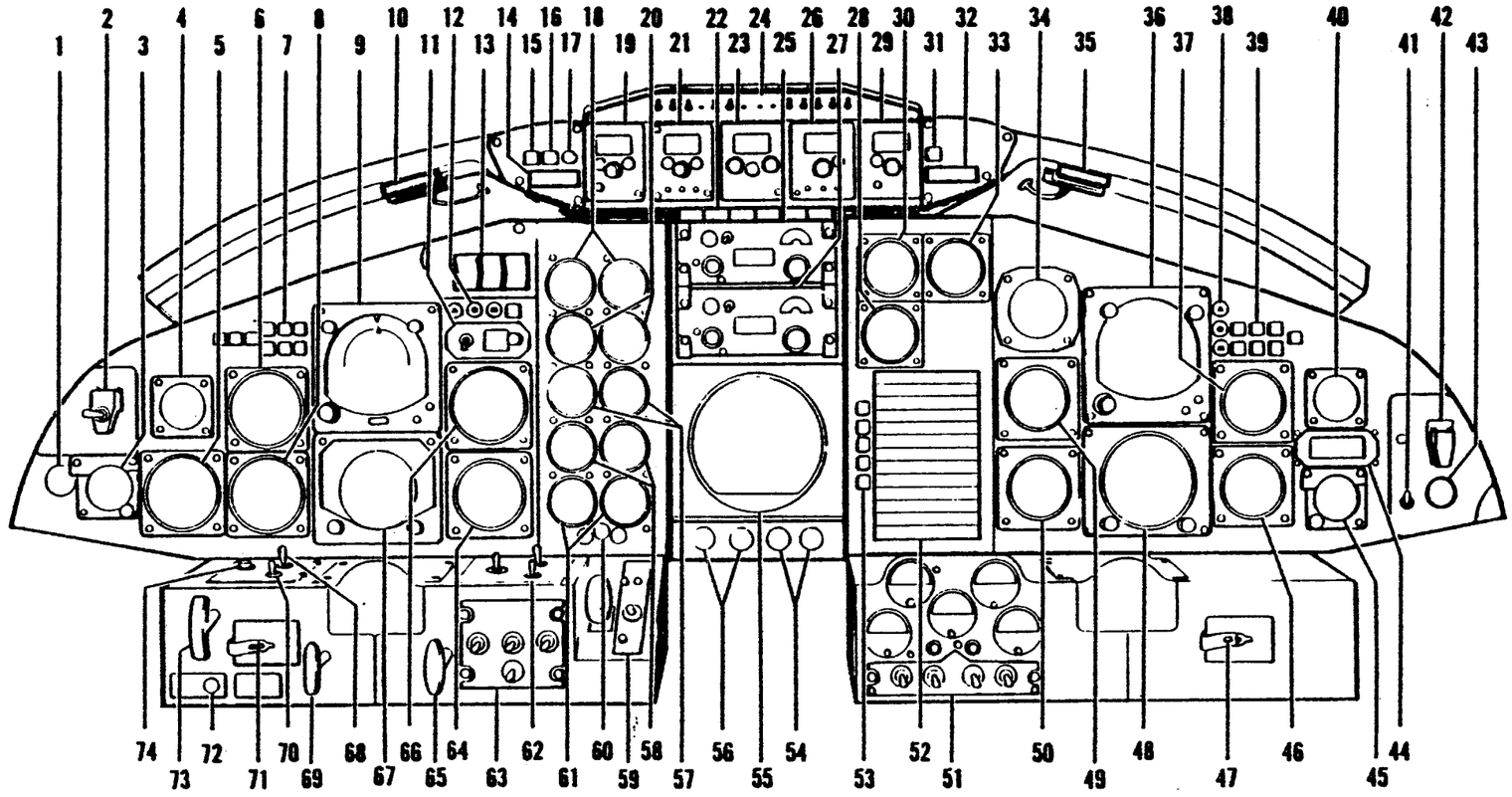
The arrangement of the instrument panels and the basic instruments is shown in figure 31-00-1. Basic instruments and their power sources are listed in the following table.

INSTRUMENT	TYPE	POWER SOURCE
ENGINE INSTRUMENTS		
Fuel flow indicator	Synchro	115-volt ac essential bus, phase A
Oil pressure indicator	Synchro	26-volt ac bus
Exhaust temperature indicator	Galvanometer	System-generated
* Tachometer - gas generator rotor speed (Ng)	Magnetic drag motor	Tachometer generator
† Tachometer - fan speed(Nf)	Magnetic drag motor	Tachometer generator
* Fuel and oil temperature indicator	Ratio Resistance	28-volt dc essential bus
Engine pressure ratio indicator		115-volt ac essential bus, phase A
† Fuel temperature indicator	Synchro	28-volt dc essential bus
FLIGHT CONTROL INSTRUMENTS		
Mach-airspeed indicator	Mechanical	Pitot-static system
Accelerometer	Mechanical	Gravity
Cabin altimeter and differential pressure indicator	Mechanical	Static system
NAVIGATIONAL INSTRUMENTS		
Stand-by compass	Mechanical-magnetic	Earth's magnetic field

* NA-370-2 Through -6, -8 and -9
† NA-370-1 and -7, NA-380-1 and Later

INSTRUMENT	TYPE	POWER SOURCE
MISCELLANEOUS INSTRUMENTS		
Fuel quantity indicator	Servo	115-volt ac essential bus
Hydraulic pressure indicator	Synchro	26-volt ac bus
Free air temperature indicator	Voltmeter	28-volt dc essential bus
Horizontal stabilizer trim position indicator	Voltmeter	28-volt dc essential bus
Rudder trim tab position indicator	Voltmeter	28-volt dc essential bus
Aileron trim tab position indicator	Voltmeter	28-volt dc essential bus
Wing flap position indicator	Voltmeter	28-volt dc essential bus
Cabin pressure rate-of-change indicator	Mechanical	
Vibrator		28-volt dc essential bus
* Battery Temperature	Electrical	28-volt dc essential bus
Frequency Meter	Electrical	28-volt dc essential bus
* Hour Meter	Electrical	28-volt dc essential bus

*NA-370-1 and -7, NA-380-1 and Later



1. PILOT'S COMPASS SLAVE INDICATOR
2. PILOT'S COMPASS SLAVING SWITCH
3. PILOT'S ACCELEROMETER
4. PILOT'S CLOCK
5. PILOT'S RADIO ALTIMETER
6. PILOT'S AIRSPEED/MACH INDICATOR
7. PILOT'S ANNUNICATOR LIGHTS
8. PILOT'S RADIO-MAGNETIC INDICATOR
9. PILOT'S APPROACH HORIZON
10. PILOT'S NOSE WHEEL STEERING-ON INDICATOR LIGHT
11. PILOT'S REPORTED ALTITUDE INDICATOR
12. PILOT'S MARKER BEACON LIGHTS
13. PILOT'S INSTRUMENT SYSTEMS SELECTOR SWITCH
14. PILOT'S MASTER CAUTION LIGHT
15. TEST RESET MONITOR LIGHT
16. PILOT'S MONITOR MASTER DISCONNECT INDICATOR LIGHT
17. LANDING LIGHTS EXTEND INDICATOR LIGHT
18. ENGINE PRESSURE RATIO GAGES
19. NO. 1 VHF COMM CONTROL PANEL
20. TACHOMETERS
21. NO. 1 VHF NAV AND DME CONTROL PANEL
22. THRUST REVERSER ADVISORY LIGHTS
23. ATC TRANSPONDER CONTROL PANEL
24. COMMUNICATION/NAVIGATION SWITCH PANEL
25. NO. 1 ADF CONTROL PANEL
26. NO. 2 VHF NAV AND DME CONTROL PANEL
27. NO. 2 ADF CONTROL PANEL
28. ENGINE SYNCHRONIZATION INDICATOR
29. NO. 2 VHF COMM CONTROL PANEL
30. WING FLAP POSITION INDICATOR
31. COPILOT'S MONITOR MASTER DISCONNECT INDICATOR LIGHT
32. COPILOT'S MASTER CAUTION LIGHT
33. FREE AIR TEMPERATURE INDICATOR
34. FUEL AND OIL TEMPERATURE INDICATOR
35. COPILOT'S NOSE WHEEL STEERING-ON INDICATOR LIGHT
36. COPILOT'S APPROACH HORIZON
37. COPILOT'S ALTIMETER
38. COPILOT'S MARKER BEACON LIGHTS
39. COPILOT'S ANNUNICATOR LIGHTS
40. COPILOT'S CLOCK
41. STEERING NEEDLE SWITCH
42. COPILOT'S COMPASS SLAVING SWITCH
43. COPILOT'S COMPASS SLAVE INDICATOR
44. COPILOT'S REPORTED ALTITUDE INDICATOR
45. COPILOT'S ACCELEROMETER
46. COPILOT'S VERTICAL VELOCITY INDICATOR
47. COPILOT'S STATIC-PRESSURE SELECTOR SWITCH
48. COPILOT'S COURSE INDICATOR
49. COPILOT'S AIRSPEED/MACH INDICATOR
50. COPILOT'S RADIO MAGNETIC INDICATOR
51. ELECTRICAL INDICATOR AND CONTROL PANEL
52. CAUTION-WARNING LIGHT PANEL
53. COMPARATOR WARNING MONITOR CONTROL LIGHTS
54. HYDRAULIC PRESSURE GAGES
55. WEATHER RADAR INDICATOR
56. OIL PRESSURE GAGES
57. EXHAUST GAS TEMPERATURE GAGES
58. FUEL FLOW INDICATORS
59. LANDING GEAR CONTROL PANEL
60. LANDING GEAR POSITION LIGHTS
61. FUEL QUANTITY GAGES
62. ALTERNATE TRIM CONTROL PANEL
63. EXTERIOR LIGHT-FUEL QUANTITY SWITCH PANEL
64. PILOT'S VERTICAL VELOCITY INDICATOR
65. PARKING BRAKE HANDLE
66. PILOT'S ALTIMETER
67. PILOT'S COURSE INDICATOR
68. SPEED BRAKE EMERGENCY DUMP SWITCH
69. GUST LOCK HANDLE
70. STALL WARNING TEST SWITCH
71. PILOT'S STATIC-PRESSURE SELECTOR SWITCH
72. LANDING GEAR ELECTRIC RESET BUTTON
73. LANDING GEAR EMERGENCY RELEASE HANDLE
74. MACH/AIRSPEED WARNING TEST BUTTON

5370-2-0-2

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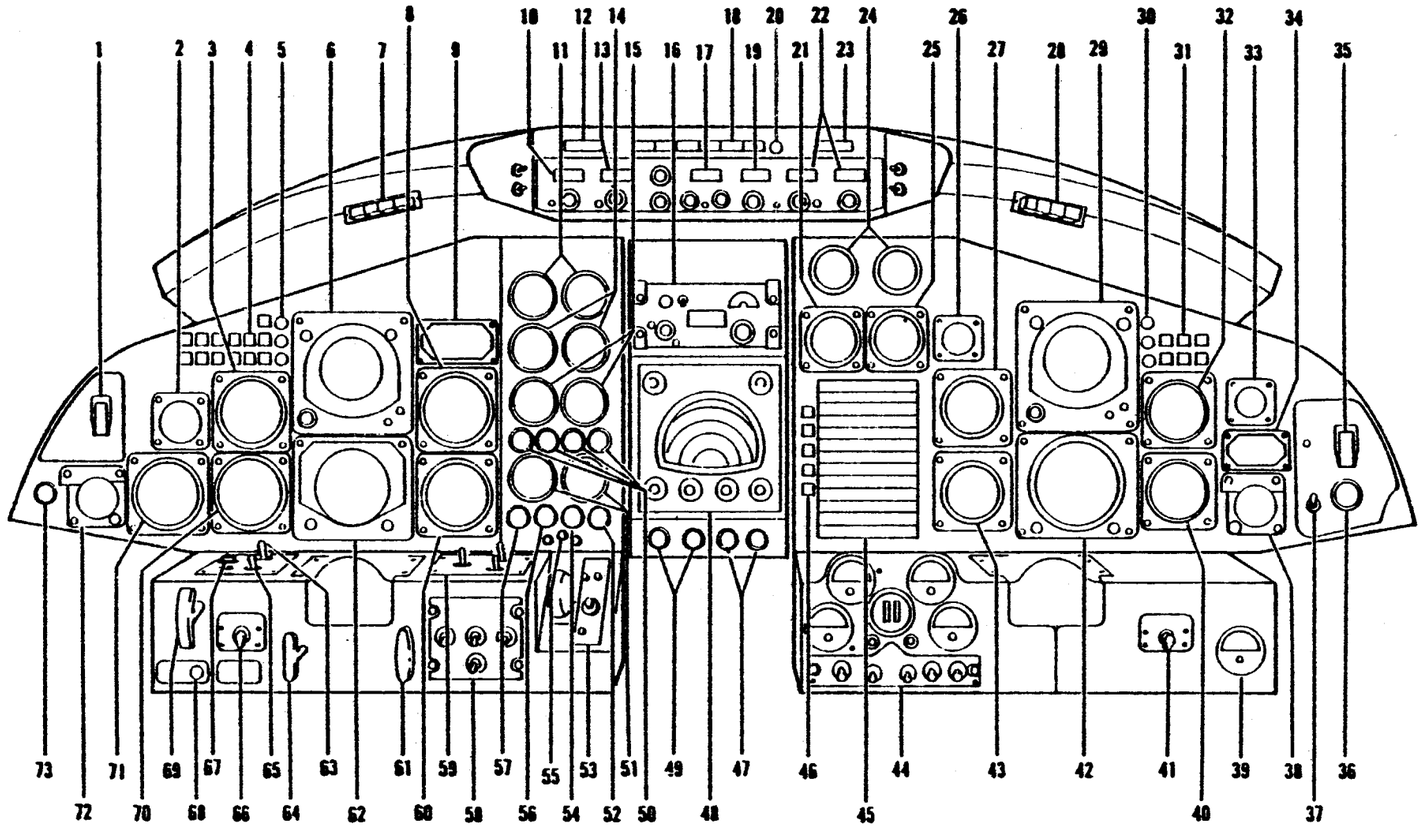
Pilot's and Copilot's Instrument Panels - NA-370-2 Through -6, -8 and -9

Figure 31-00-1



1. PILOT'S COMPASS SLAVING SWITCH
2. PILOT'S CLOCK
3. PILOT'S AIRSPEED/MACH INDICATOR
4. PILOT'S ANNUNCIATOR LIGHTS
5. PILOT'S MARKER BEACON LIGHTS
6. PILOT'S FLIGHT DIRECTOR
7. PILOT'S NOSE WHEEL STEERING-ON, ENGINE STALL & ANTISKID INDICATOR LIGHTS
8. PILOT'S ALTIMETER
9. ALTITUDE ALERTER
10. NO. 1 VHF COMM CONTROL PANEL
11. FUEL FLOW INDICATORS
12. PILOT'S MASTER CAUTION LIGHT
13. NO. 1 VHF NAV CONTROL PANEL
14. ENGINE PRESSURE RATIO GAGES
15. EXHAUST GAS TEMPERATURE GAGES
16. ADF CONTROL PANEL
17. ATC TRANSPONDER CONTROL PANEL
18. THRUST REVERSER ADVISORY LIGHTS
19. NO. 2 VHF NAV CONTROL PANEL
20. LANDING LIGHTS EXTENDED INDICATOR LIGHT
21. WING FLAP POSITION INDICATOR
22. NO. 2 VHF COMM CONTROL PANEL
23. COPILOT'S MASTER CAUTION LIGHT
24. FUEL QUANTITY GAGES
25. FREE AIR TEMPERATURE INDICATOR
26. ENGINE SYNCHRONIZATION INDICATOR
27. COPILOT'S AIRSPEED/MACH INDICATOR
28. COPILOT'S NOSE WHEEL STEERING-ON, ENGINE STALL & ANTISKID INDICATOR LIGHTS
29. COPILOT'S FLIGHT DIRECTOR
30. COPILOT'S MARKER BEACON LIGHTS
31. COPILOT'S ANNUNCIATOR LIGHTS
32. COPILOT'S ALTIMETER
33. COPILOT'S CLOCK
34. REPORTED ALTITUDE INDICATOR
35. COPILOT'S COMPASS SLAVING SWITCH
36. COPILOT'S COMPASS SLAVE INDICATOR
37. STEERING NEEDLE SWITCH
38. COPILOT'S ACCELEROMETER
39. AC FREQUENCY METER
40. COPILOT'S VERTICAL VELOCITY INDICATOR
41. COPILOT'S STATIC-PRESSURE SELECTOR VALVE
42. COPILOT'S COURSE INDICATOR
43. COPILOT'S RADIO MAGNETIC INDICATOR
44. ELECTRICAL INDICATOR & CONTROL PANEL
45. CAUTION-WARNING LIGHT PANEL
46. COMPARATOR WARNING MONITOR CONTROL LIGHTS
47. HYDRAULIC PRESSURE GAGES
48. WEATHER RADAR
49. FUEL TEMPERATURE INDICATORS
50. TACHOMETERS (FANSPEED, N_F)
51. TACHOMETERS (GAS GENERATOR ROTOR SPEED' N_G)
52. OIL TEMPERATURE INDICATOR
53. LANDING GEAR CONTROL PANEL
54. OIL PRESSURE GAGE
55. LANDING GEAR POSITION LIGHTS
56. OIL TEMPERATURE INDICATOR
57. OIL PRESSURE GAGE
58. EXTERIOR LIGHT-FUEL QUANTITY SWITCH PANEL
59. ALTERNATE TRIM CONTROL PANEL
60. PILOT'S VERTICAL VELOCITY INDICATOR
61. PARKING BRAKE HANDLE
62. PILOT'S COURSE INDICATOR
63. SPEED BRAKE EMERGENCY DUMP SWITCH
64. GUST LOCK HANDLE
65. STALL WARNING TEST SWITCH
66. PILOT'S STATIC-PRESSURE SELECTOR VALVE
67. MACH/AIRSPEED WARNING TEST BUTTON
68. LANDING GEAR ELECTRIC RESET BUTTON
69. LANDING GEAR EMERGENCY RELEASE HANDLE
70. PILOT'S RADIO MAGNETIC INDICATOR
71. RADIO ALTIMETER
72. PILOT'S ACCELEROMETER
73. PILOT'S COMPASS SLAVE INDICATOR

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Pilot's and Copilot's Instrument Panels -
NA-380-1 and Later
Figure 31-00-2

INSTRUMENTS - MAINTENANCE PRACTICES

1. Replacement of Instrument Panels

A. Removing Instrument Panels

(1) Place battery switch at OFF and disconnect external electrical power cart from airplane.

(2) Loosen Airloc fasteners that hold instrument panel to instrument panel frame.

(3) Slide instrument panel straight aft. Maintain panel in vertical position. Be careful not to scrape bottom instruments on panel frame.

(4) Disconnect pitot-static lines.

(5) Disconnect all floating leads from instrument panel.

(6) Disconnect electrical plugs from their receptacles.

CAUTION: ALL EXPOSED ELECTRICAL RECEPTACLES, PLUGS, AND PITOT-STATIC FITTINGS MUST BE COVERED TO KEEP OUT FOREIGN MATERIAL, PARTICULARLY METAL PARTICLES, THAT MAY CAUSE THE INSTRUMENTS TO BECOME INOPERATIVE.

- IF ELECTRICAL PLUGS ARE NOT MARKED AND ARE OF THE SAME TYPE, BE SURE TO MARK EACH PLUG FOR REINSTALLATION.

(7) Carefully lift panel from airplane.

B. Installing Instrument Panels

(1) Place battery switch at OFF.

(2) Disconnect external electrical power from airplane.

(3) Place instrument panel in instrument panel frame.

(4) Connect pitot-static lines.

(5) Connect all floating leads to instrument panel.

(6) Insert electrical plugs into receptacles.

(7) Safety electrical plugs with lockwire.

(8) Slide instrument panel into position against instrument panel frame. Be careful not to scrape bottom instruments on panel frame.

(9) Tighten Airloc fasteners.

(10) Check instruments for operation.

2. Replacement of Clamp-mounted Instruments

A. Removing Clamp-mounted Instruments

(1) Place battery switch at OFF if instrument is electrically operated.

(2) Disconnect external electrical power from airplane if instrument is electrically operated.

(3) Loosen round-head jackscrew.

(4) Withdraw instruments from panel, to expose connections on rear of instrument.

(5) Disconnect power source from instrument. Cover disconnected fittings.

CAUTION: ALL EXPOSED ELECTRICAL RECEPTACLES, PLUGS, AND PITOT-STATIC FITTINGS MUST BE COVERED TO KEEP OUT FOREIGN MATTER, PARTICULARLY METAL PARTICLES, THAT MAY CAUSE THE INSTRUMENTS TO BECOME INOPERATIVE.

(6) Remove instrument from panel.

(7) Remove clamp from panel by completely removing jackscrew and flush-head attaching screw.

B. Installing Clamp-mounted Instruments

See figure 31-00-201 for installation procedure.

3. Replacement of Bezel-mounted Instruments

A. Removing Bezel-mounted Instruments

(1) Place battery switch at OFF if instrument is electrically operated.

(2) Disconnect external electrical power from airplane if instrument is electrically operated.

(3) Remove mounting screws from instrument bezel.

(4) To expose power connections, pull instrument from panel.

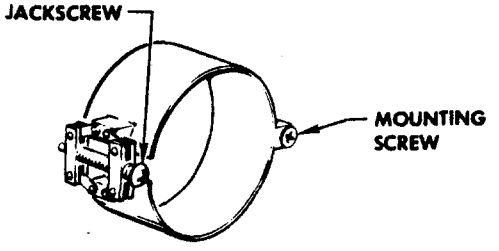
(5) Disconnect instrument power source and cover disconnected fitting.

CAUTION: ALL EXPOSED ELECTRICAL RECEPTACLES, PLUGS, AND PITOT-STATIC FITTINGS MUST BE COVERED TO KEEP OUT FOREIGN MATTER, PARTICULARLY METAL PARTICLES, THAT MAY CAUSE THE INSTRUMENTS TO BECOME INOPERATIVE.

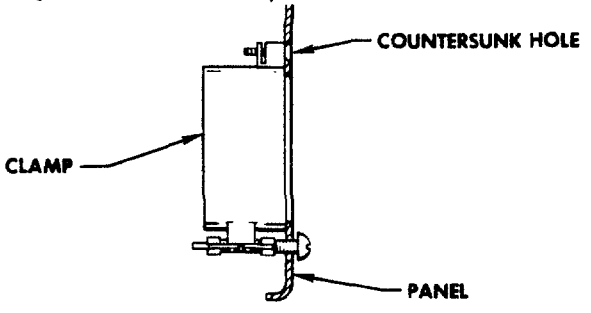
CLAMP-TYPE MOUNTING

1 Clean mounting clamp and instrument for bonding.

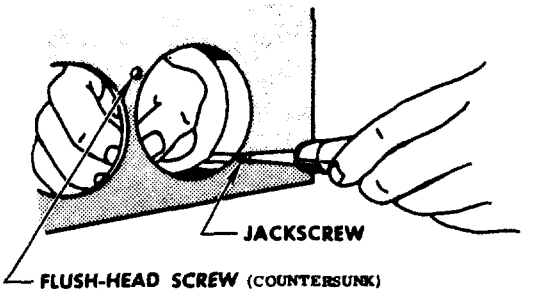
2 Remove jackscrew and mounting screw from clamp.



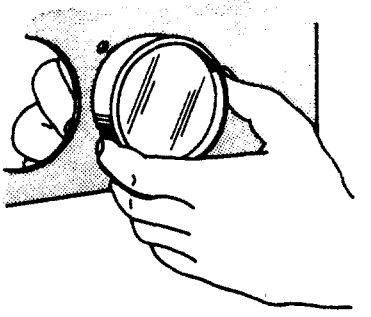
3 Position clamp on forward side of instrument panel with small lug at countersunk hole in panel.



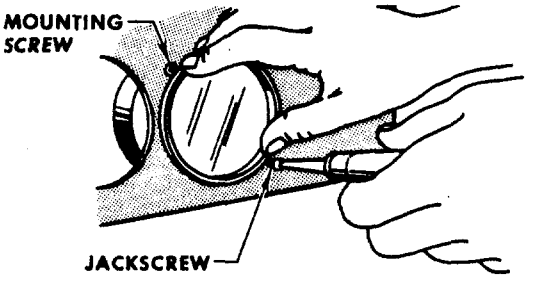
4 Install flush-head screw, but do not tighten it. Install jackscrew until clamp begins to close.



5 Connect instrument power source, and insert instrument in panel.



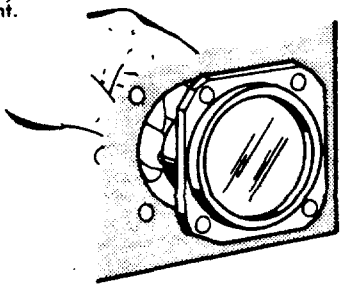
6 Hold instrument in place and tighten jackscrew and mounting screw.



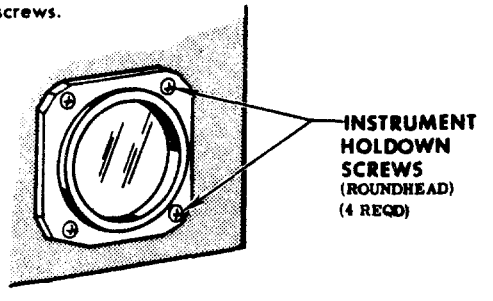
BEZEL-TYPE MOUNTING

1 Clean instrument and contact areas for bonding.

2 Connect instrument power source to instrument.



3 Install instrument in panel with four screws.



S265-1-51-7

(6) Remove instrument.

NOTE: If instrument has static fitting and is being replaced, proceed with the following steps.

(7) Note angle of static fitting.

(8) Remove static fitting from instrument being replaced. Inspect static fitting and replace if necessary.

(9) Install static fitting at correct angle in replacement instrument.

B. Installing Bezel-mounted Instruments

See figure 31-00-201 for installation procedure.

4. Electrical Bonding of Instruments

To make a good electrical contact, bond upper left mounting screw on all bezel-mounted instruments. Clamp-mounted instruments are bonded to the clamp. Remove paint and foreign matter from three fourths of the diameter of that part of instrument case that contacts mounting clamp.

CHAPTER

32

LANDING GEAR

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LANDING GEAR

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	102		Mar 1/70
	103		Mar 1/70
	104		Mar 1/70
32-40-02	301/302	*	Feb 29/96
	303/304	*	Feb 29/96
32-40-03	401		Mar 1/70
	402		Mar 1/70
	403		Oct 3/77
	404		Mar 1/70
	405		Jul 6/73
	406		Oct 3/77
	407		Jan 19/90
	408		Jan 19/90
	409		Mar 1/70
	410		Mar 1/70
	411		Mar 1/70
	412		Mar 1/70
	413		Mar 1/70
	414		Mar 1/70
	415		Jan 19/90
	416		Jan 19/90
	417		Jan 19/90
	418		Jan 19/90
	419		Oct 3/77
	420		Mar 1/70

* The asterisk indicates pages revised, added, or deleted by the current revision.

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	202		Oct 3/77
	203		Oct 3/77
	204		Oct 3/77
32-41A-00	1	*	Feb 29/96
	2	*	Feb 29/96
	DELETED 3	*	Feb 29/96
32-41A-01	101		Jul 6/73
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	204		Oct 3/77
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	5		Mar 1/70
	6		Mar 1/70
	7		Jul 6/73
	8		Oct 3/77
	9		Oct 3/77
	10		Oct 3/77
	11		Jul 6/73
	12		Oct 3/77
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	13B		Oct 3/77
	14		Oct 3/77
	14A		Oct 3/77
32-50-01	101		Jul 6/73
	102		Mar 1/70
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	104		Mar 1/70
	105		Mar 1/70
32-50-02	401		Mar 1/70
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	404		Jan 19/90
	405		Jul 6/73
	406		Mar 1/70
	407		Oct 5/92
	408		Oct 5/92
	409		Oct 5/92
	410		Oct 5/92
	410A		Jan 19/90
	410B		Jan 19/90
	411		Mar 1/70
32-50-03	501		Sep 2/88
	502		Sep 2/88
	503		Sep 2/88
	504		Sep 2/88
	505		Sep 2/88
	506		Sep 2/88
	507		Sep 2/88
	508		Sep 2/88
	509/510		Sep 2/88
	601		Sep 2/88
	602		Sep 2/88
	603/604		Sep 2/88
32-60-00	1		Jul 6/73
	2		Jul 2/73
32-60-01	401		Mar 1/70
	402		Mar 1/70
	403		Mar 1/70
	404		Mar 15/83
	405		Mar 15/83
	406		Dec 1/77
	407		Dec 1/77
	408		Mar 1/70
	409		Dec 1/77
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	411		Mar 15/83
	412		Mar 15/83
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	414		Mar 15/83
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* The asterisk indicates pages revised, added, or deleted by the current revision.

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LANDING GEAR

DESCRIPTION

GENERAL

The airplane has an electrically controlled, hydraulically and pneumatically operated, tricycle-type landing gear. (See Figure 32-00-1.) The system receives electrical power from the 28 VDC essential bus, hydraulic pressure from the hydraulic power system, and pneumatic pressure from the engines. The landing gear is fully retractable into the fuselage and wing and is faired in by nose wheel doors and main gear strut fairing doors. The landing gear system is divided into subsystems, according to their functions. The subsystems are the landing gear and landing gear door, landing gear position indicating and warning system, and landing gear emergency system. The landing gear system contains the necessary electrical wiring and mechanically actuated electrical switches to control retraction and extension of the landing gear and operation of the position indicating and warning system. All sequencing of the landing gear is accomplished with electrical switches.

The landing gear position indicating system consists of three green indicator lights, each of which comes on when the corresponding landing gear is down and locked. The landing gear warning system gives visual and audible indication of a landing gear unsafe condition. The visual warning is indicated by a red light in the landing gear control handle; the audible warning is indicated by an audible warning fed into the radio headsets. A warning horn in the cockpit also gives an audible indication of a landing gear unsafe condition. The landing gear emergency system extends the landing gear in case of hydraulic system failure. The landing gear is extended in emergency by pulling the landing gear emergency release handle in the cockpit. This mechanically unlocks the landing gear and gear doors, and opens gear-down hydraulic pressure lines to return pressure. During emergency extension, the landing gear is extended by gravity and aerodynamic forces.

The L.G. ELEC. RESET switch, on the instrument panel, is used to reset landing gear control electrical power if the emergency release handle has been pulled.

The landing gear may also be retracted, although the airplane is not airborne, by depressing a landing gear downlock override button above the gear control handle.

On S/N 370-1 thru -9 changed by S/B 73-7, and S/N 380-1 and subs, a set of relay contacts has been added in the main gear load switch circuit to isolate the cabin pressure dump relay from the load switches whenever the gear handle is in the UP position. (See Figure 32-00-3 or 32-00-4 for wiring configurations.)

On S/N 370-1, -7, 380-1 and subs and 370-2 thru -6, -8 and -9 changed by SOK 11, the auxiliary system shutoff valve is automatically opened (power removed from the valve) in the event of a hydraulic pump failure at take-off during landing gear retraction (see Figures 32-00-3, -4 and -5). In the event of hydraulic pump failure during takeoff and the landing gear system in the retraction cycle, when the normal system hydraulic pressure drops to 2000 (±100) psi, electrical power is routed through the hydraulic low pressure switch to the coil of the hydraulic low pressure monitor relay (K780), which is then energized. At the same time, electrical power is routed through the RH and LH landing gear uplock switches (S596 and S594), through the energized closed contacts of relay K780 to the coil of the landing gear valve (up) monitor relay K781. Relay K781 energizes which in turn removes electrical power from the auxiliary system shutoff valve allowing the valve to open which releases the hydraulic pressure stored in the auxiliary accumulator into the normal hydraulic system thereby completely retracting the main landing gear. As soon as the RH and LH landing gear uplocks are actuated, electrical power is removed from relay K781, which then de-energizes and applies electrical power to the auxiliary system shutoff valve. The shutoff valve is then closed and traps the remaining auxiliary system pressure in the auxiliary system accumulator.

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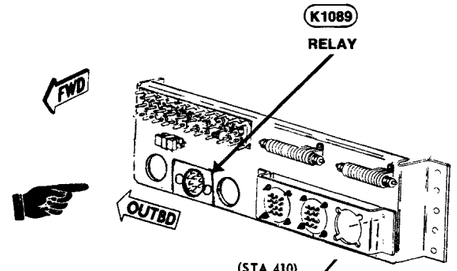
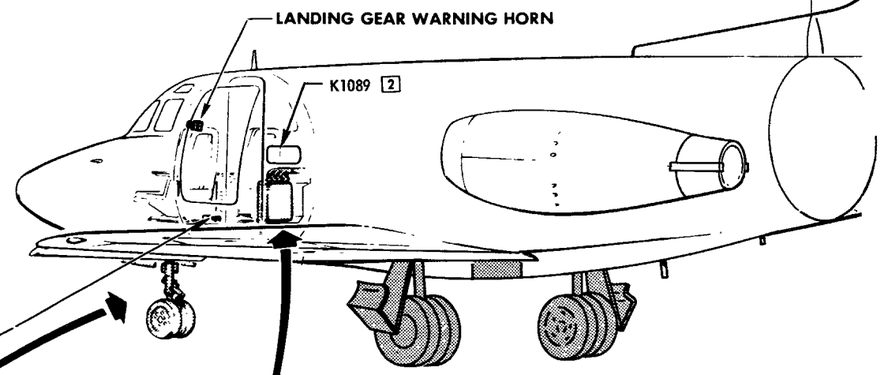
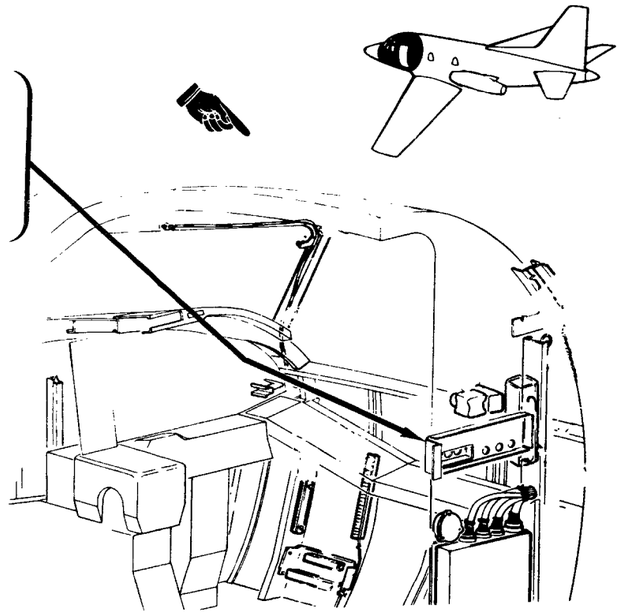
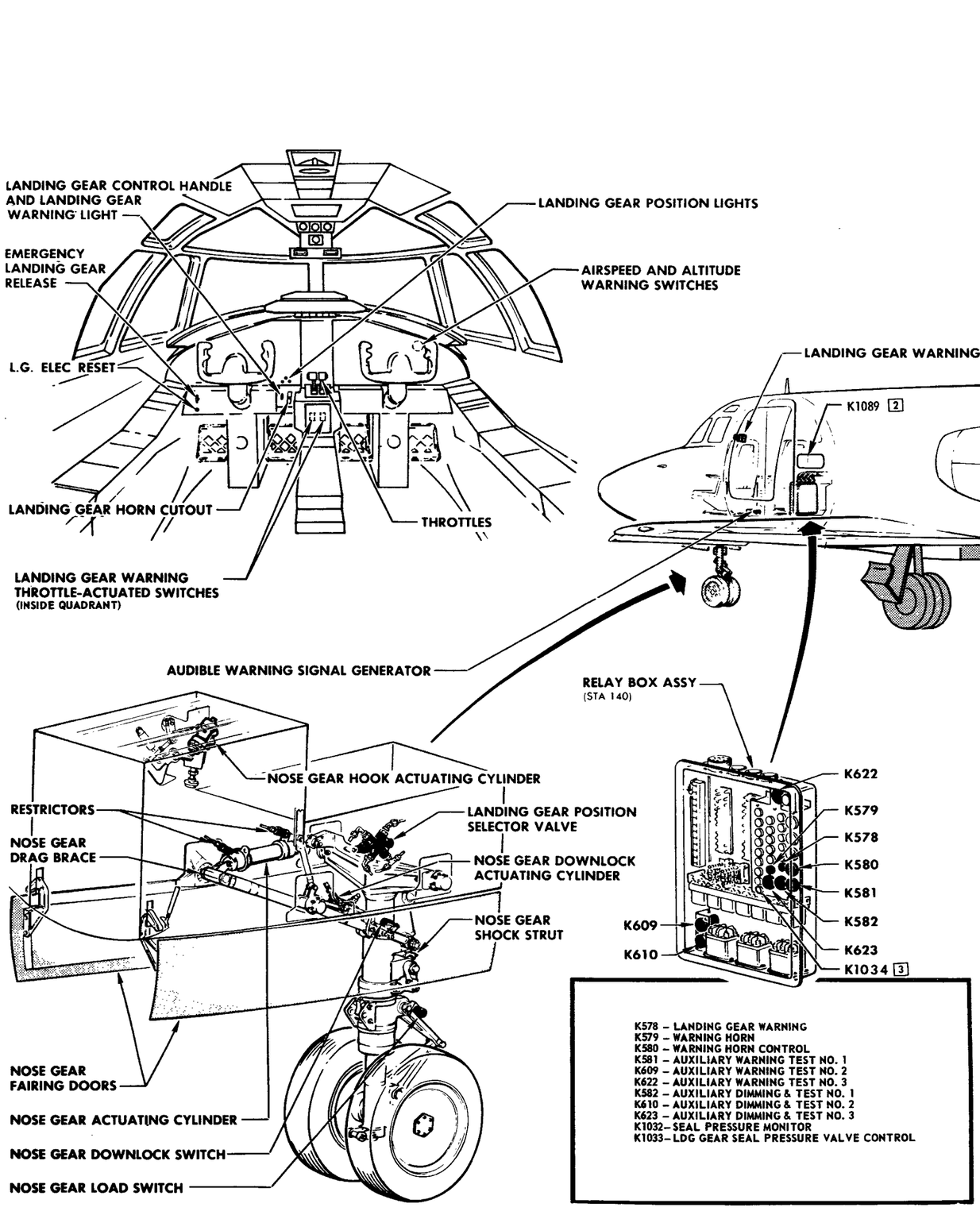
LANDING GEAR CONTROL HANDLE AND LANDING GEAR WARNING LIGHT
EMERGENCY LANDING GEAR RELEASE
L.G. ELEC RESET
LANDING GEAR HORN CUTOUT
LANDING GEAR WARNING THROTTLE-ACTUATED SWITCHES (INSIDE QUADRANT)
LANDING GEAR POSITION LIGHTS
AIRSPEED AND ALTITUDE WARNING SWITCHES
THROTTLES








AUDIBLE WARNING SIGNAL GENERATOR
NOSE GEAR HOOK ACTUATING CYLINDER
RESTRICTORS
NOSE GEAR DRAG BRACE
LANDING GEAR POSITION SELECTOR VALVE
NOSE GEAR DOWNLOCK ACTUATING CYLINDER
NOSE GEAR SHOCK STRUT
NOSE GEAR FAIRING DOORS
NOSE GEAR ACTUATING CYLINDER
NOSE GEAR DOWNLOCK SWITCH
NOSE GEAR LOAD SWITCH

K578 – LANDING GEAR WARNING
K579 – WARNING HORN
K580 – WARNING HORN CONTROL
K581 – AUXILIARY WARNING TEST NO. 1
K609 – AUXILIARY WARNING TEST NO. 2
K622 – AUXILIARY WARNING TEST NO. 3
K582 – AUXILIARY DIMMING & TEST NO. 1
K610 – AUXILIARY DIMMING & TEST NO. 2
K623 – AUXILIARY DIMMING & TEST NO. 3
K1032– SEAL PRESSURE MONITOR
K1033– LDG GEAR SEAL PRESSURE VALVE CONTROL

1 AIRPLANES SERIAL NO. NA-370-2 THROUGH -6 CHANGED BY SERVICE BULLETIN 72-6, NA-370-1 AND -7, AND NA-380-1 AND LATER

2 AIRPLANES SERIAL NO. NA-370-1 THROUGH -9 CHANGED BY SERVICE BULLETIN 73-7, AND NA-380-1 AND LATER (LOCATED ON BOTTOM TRAY OF ELECTRICAL EQUIPMENT RACK)

3 AIRPLANES SERIAL NO. S/N 370-1 THRU -9

Figure 32-00-1 — Landing Gear System — Electrical and Hydraulic (Sheet 1 of 2)

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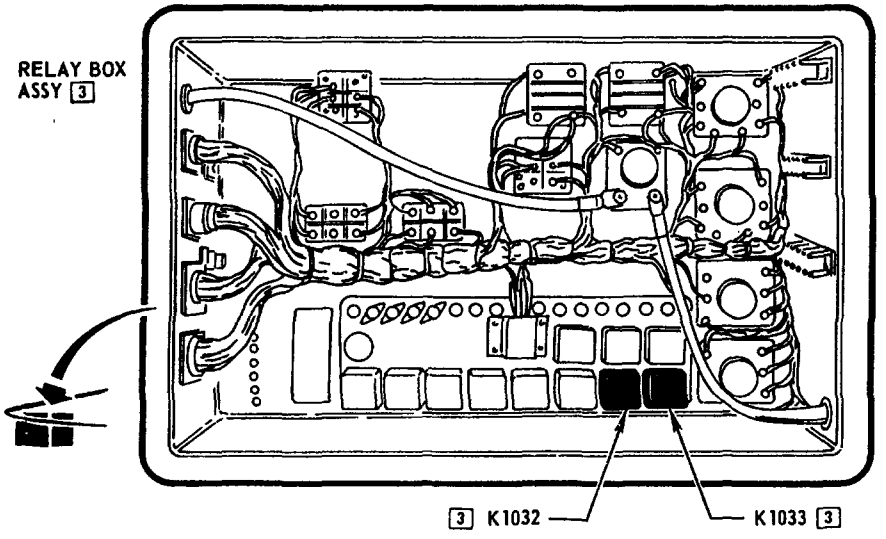

[3] AIRPLANES SERIAL NO. S/N 370-1 THRU -9

Figure 32-00-1 — Landing Gear System — Electrical and Hydraulic (Sheet 2 of 2)

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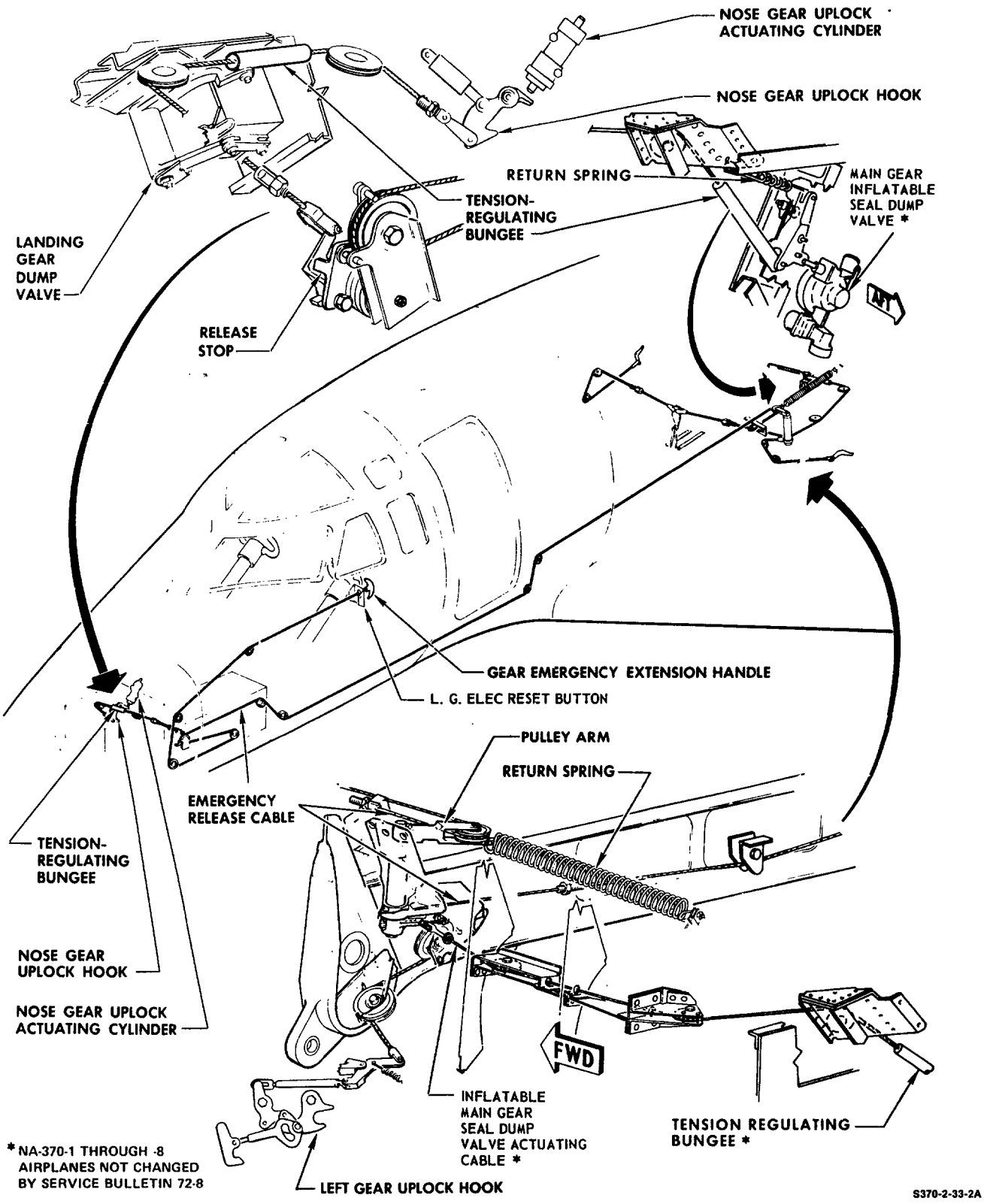
NOSE GEAR UPLOCK ACTUATING CYLINDER
NOSE GEAR UPLOCK HOOK
RETURN SPRING
MAIN GEAR INFLATABLE SEAL DUMP VALVE *
TENSION-REGULATING BUNGEE
LANDING GEAR DUMP VALVE
RELEASE STOP
GEAR EMERGENCY EXTENSION HANDLE
L. G. ELEC RESET BUTTON
PULLEY ARM
RETURN SPRING
EMERGENCY RELEASE CABLE
TENSION-REGULATING BUNGEE
NOSE GEAR UPLOCK HOOK
NOSE GEAR UPLOCK ACTUATING CYLINDER
FWD
INFLATABLE MAIN GEAR SEAL DUMP VALVE ACTUATING CABLE *
TENSION REGULATING BUNGEE *
LEFT GEAR UPLOCK HOOK
* NA-370-1 THROUGH -8 AIRPLANES NOT CHANGED BY SERVICE BULLETIN 72-8
S370-2-33-2A

Figure 32-00-2 — Landing Gear — Emergency Extension System

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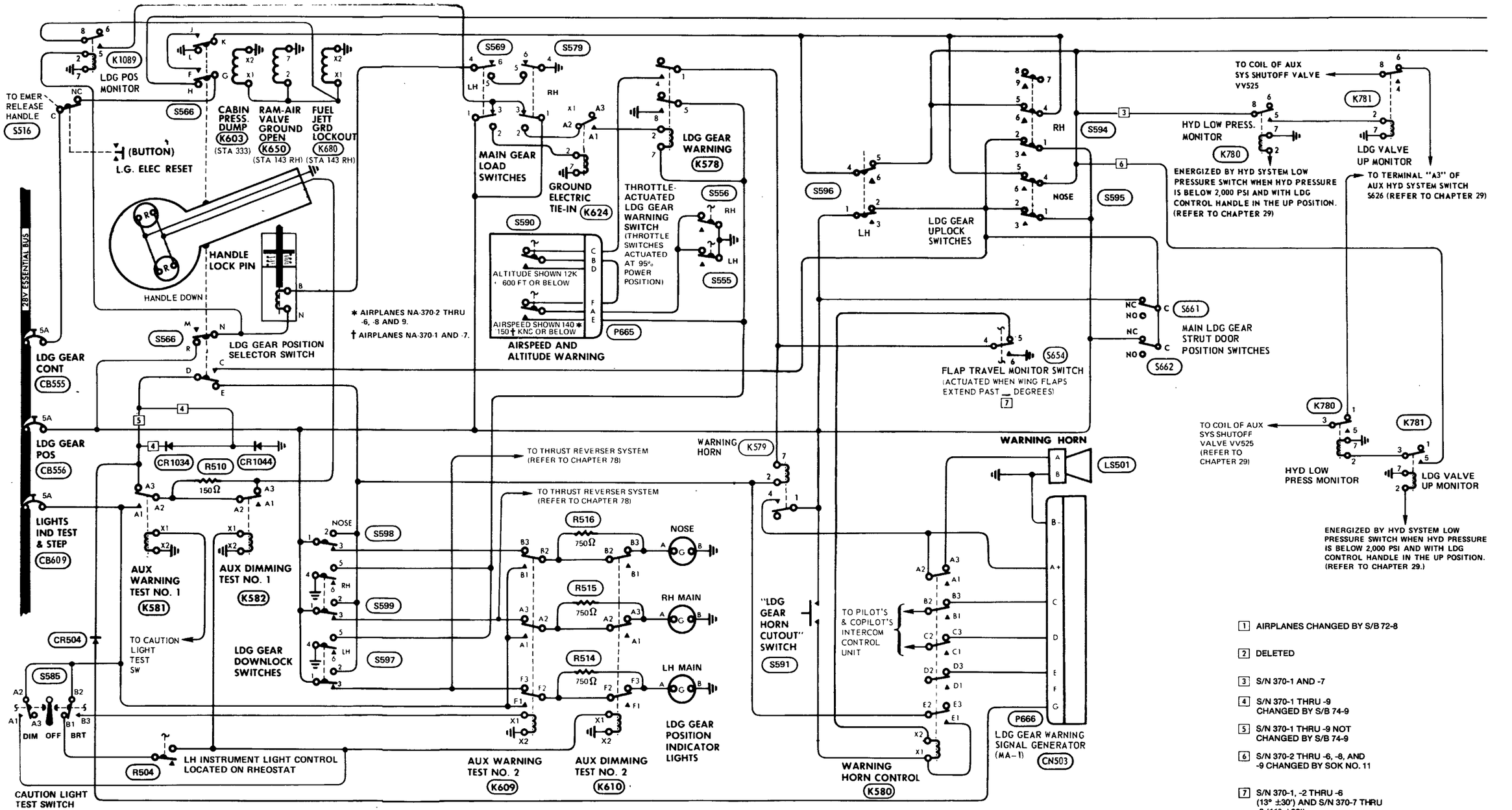
28V ESSENTIAL BUS
LDG GEAR CONT CB555
LDG GEAR POS CB556
LIGHTS IND TEST & STEP CB609
5A
K1089
LDG POS MONITOR
TO EMER RELEASE HANDLE S516
S566
(BUTTON) L.G. ELEC RESET
CABIN PRESS. DUMP K603 (STA 333)
RAM-AIR VALVE GROUND OPEN K650 (STA 143 RH)
FUEL JETT GRD LOCKOUT K680 (STA 143 RH)
HANDLE LOCK PIN
HANDLE DOWN
LDG GEAR POSITION SELECTOR SWITCH
* AIRPLANES NA-370-2 THRU -6, -8 AND 9.
† AIRPLANES NA-370-1 AND -7.
S569
S579
LH
RH
MAIN GEAR LOAD SWITCHES
GROUND ELECTRIC TIE-IN K624
LDG GEAR WARNING K578
THROTTLE-ACTUATED LDG GEAR WARNING SWITCH (THROTTLE SWITCHES ACTUATED AT 95% POWER POSITION)
S556
S555
S590
ALTITUDE SHOWN 12K - 600 FT OR BELOW
AIRSPEED SHOWN 140 * 150 † KNS OR BELOW
AIRSPEED AND ALTITUDE WARNING
P665
S596
S595
LDG GEAR UPLOCK SWITCHES
NOSE
RH
LH
TO COIL OF AUX SYS SHUTOFF VALVE VV525
K781
HYD LOW PRESS. MONITOR
K780
ENERGIZED BY HYD SYSTEM LOW PRESSURE SWITCH WHEN HYD PRESSURE IS BELOW 2,000 PSI AND WITH LDG CONTROL HANDLE IN THE UP POSITION. (REFER TO CHAPTER 29)
LDG VALVE UP MONITOR
TO TERMINAL "A3" OF AUX HYD SYSTEM SWITCH S626 (REFER TO CHAPTER 29)
S661
MAIN LDG GEAR STRUT DOOR POSITION SWITCHES
S662
S654
FLAP TRAVEL MONITOR SWITCH (ACTUATED WHEN WING FLAPS EXTEND PAST __ DEGREES) 7
TO THRUST REVERSER SYSTEM (REFER TO CHAPTER 78)
WARNING HORN K579
WARNING HORN
LS501
CR1034
R510
150Ω
CR1044
S598
S599
S597
AUX WARNING TEST NO. 1 K581
AUX DIMMING TEST NO. 1 K582
LDG GEAR DOWNLOCK SWITCHES
TO CAUTION LIGHT TEST SW
CR504
S585
DIM OFF BRT
LH INSTRUMENT LIGHT CONTROL LOCATED ON RHEOSTAT
R504
CAUTION LIGHT TEST SWITCH
R516
R515
R514
750Ω
RH MAIN
LH MAIN
LDG GEAR POSITION INDICATOR LIGHTS
AUX WARNING TEST NO. 2 K609
AUX DIMMING TEST NO. 2 K610
"LDG GEAR HORN CUTOUT" SWITCH S591
TO PILOT'S & COPILOT'S INTERCOM CONTROL UNIT
WARNING HORN CONTROL K580
P666
LDG GEAR WARNING SIGNAL GENERATOR (MA-1) CN503
TO COIL OF AUX SYS SHUTOFF VALVE VV525 (REFER TO CHAPTER 29)
HYD LOW PRESS MONITOR
LDG VALVE UP MONITOR
ENERGIZED BY HYD SYSTEM LOW PRESSURE SWITCH WHEN HYD PRESSURE IS BELOW 2,000 PSI AND WITH LDG CONTROL HANDLE IN THE UP POSITION. (REFER TO CHAPTER 29.)
[1] AIRPLANES CHANGED BY S/B 72-8
[2] DELETED
[3] S/N 370-1 AND -7
[4] S/N 370-1 THRU -9 CHANGED BY S/B 74-9
[5] S/N 370-1 THRU -9 NOT CHANGED BY S/B 74-9
[6] S/N 370-2 THRU -6, -8, AND -9 CHANGED BY SOK NO. 11
[7] S/N 370-1, -2 THRU -6 (13° ±30') AND S/N 370-7 THRU -9 (11° ±30')
S370-2-33-3B

Figure 32-00-3 — Landing Gear System — S/N 370-1 Thru -9 (Sheet 1 of 2)

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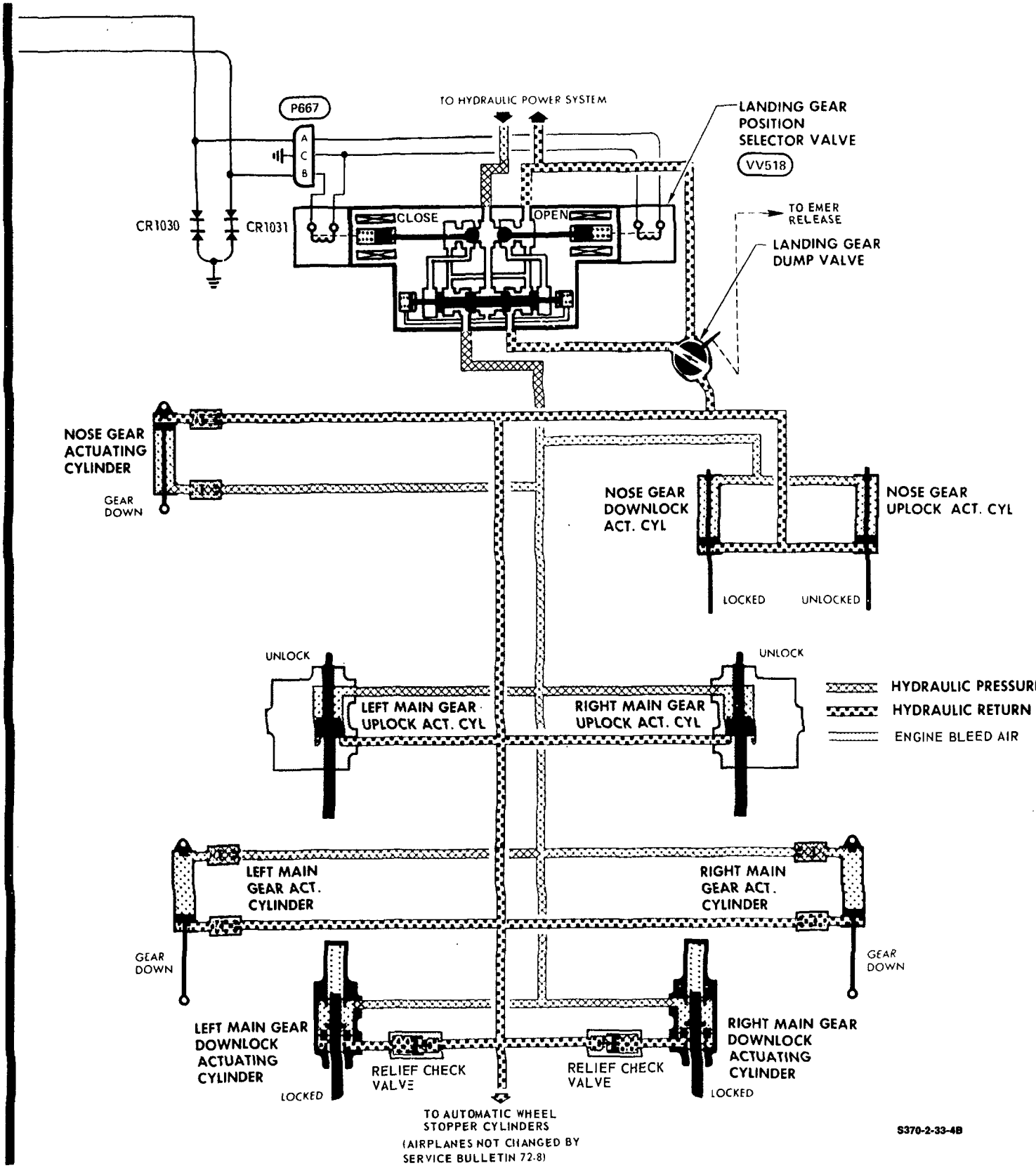

Figure 32-00-3 — Landing Gear System — S/N 370-1 Thru -9 (Sheet 2 of 2)

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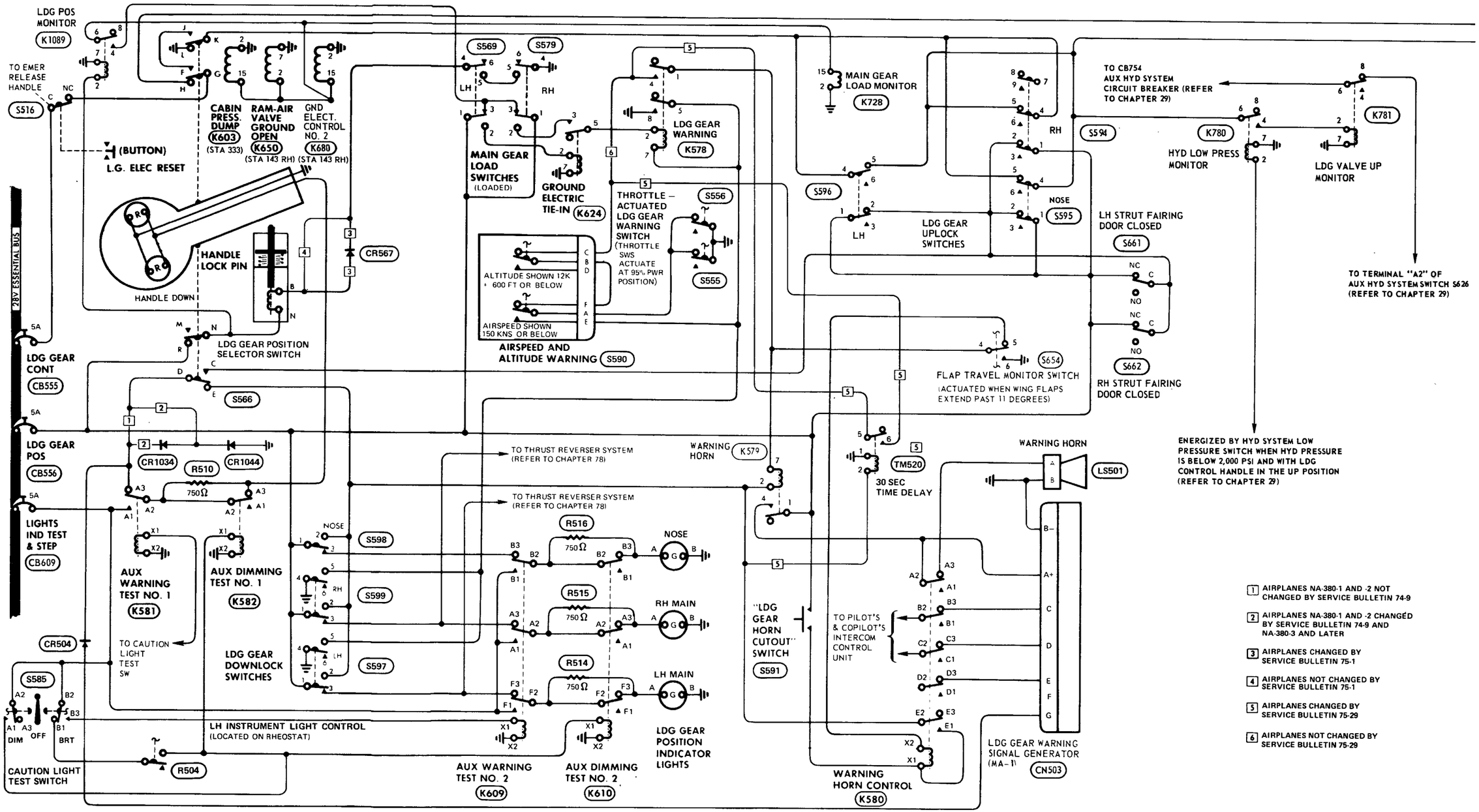
LDG POS MONITOR
K1089
TO EMER RELEASE HANDLE
S516
(BUTTON) L.G. ELEC RESET
CABIN PRESS. DUMP K603 (STA 333)
RAM-AIR VALVE GROUND OPEN K650 (STA 143 RH)
GND ELECT. CONTROL NO. 2 K680 (STA 143 RH)
HANDLE LOCK PIN
HANDLE DOWN
CR567
28V ESSENTIAL BUS
5A
LDG GEAR CONT CB555
LDG GEAR POSITION SELECTOR SWITCH
S566
LDG GEAR POS CB556
CR1034
R510
750Ω
CR1044
LIGHTS IND TEST & STEP CB609
AUX WARNING TEST NO. 1 K581
AUX DIMMING TEST NO. 1 K582
TO CAUTION LIGHT TEST SW
CR504
S585
DIM OFF BRT
CAUTION LIGHT TEST SWITCH
R504
LDG GEAR DOWNLOCK SWITCHES
NOSE S598
RH S599
LH S597
LH INSTRUMENT LIGHT CONTROL (LOCATED ON RHEOSTAT)
AUX WARNING TEST NO. 2 K609
AUX DIMMING TEST NO. 2 K610
S569
S579
LH
RH
MAIN GEAR LOAD SWITCHES (LOADED)
GROUND ELECTRIC TIE-IN K624
LDG GEAR WARNING K578
THROTTLE – ACTUATED LDG GEAR WARNING SWITCH (THROTTLE SWS ACTUATE AT 95% PWR POSITION)
S556
S555
ALTITUDE SHOWN 12K + 600 FT OR BELOW
AIRSPEED SHOWN 150 KNS OR BELOW
AIRSPEED AND ALTITUDE WARNING S590
TO THRUST REVERSER SYSTEM (REFER TO CHAPTER 78)
R516
R515
R514
NOSE
RH MAIN
LH MAIN
LDG GEAR POSITION INDICATOR LIGHTS
MAIN GEAR LOAD MONITOR K728
S596
LDG GEAR UPLOCK SWITCHES
S594
S595
NOSE
RH
LH
FLAP TRAVEL MONITOR SWITCH S654
(ACTUATED WHEN WING FLAPS EXTEND PAST 11 DEGREES)
"LDG GEAR HORN CUTOUT" SWITCH S591
WARNING HORN K579
TM520
30 SEC TIME DELAY
TO PILOT'S & COPILOT'S INTERCOM CONTROL UNIT
WARNING HORN CONTROL K580
WARNING HORN LS501
LDG GEAR WARNING SIGNAL GENERATOR (MA-1) CN503
TO CB754 AUX HYD SYSTEM CIRCUIT BREAKER (REFER TO CHAPTER 29)
K780
HYD LOW PRESS MONITOR
K781
LDG VALVE UP MONITOR
LH STRUT FAIRING DOOR CLOSED S661
RH STRUT FAIRING DOOR CLOSED S662
TO TERMINAL "A2" OF AUX HYD SYSTEM SWITCH S626 (REFER TO CHAPTER 29)
ENERGIZED BY HYD SYSTEM LOW PRESSURE SWITCH WHEN HYD PRESSURE IS BELOW 2,000 PSI AND WITH LDG CONTROL HANDLE IN THE UP POSITION (REFER TO CHAPTER 29)
[1] AIRPLANES NA-380-1 AND -2 NOT CHANGED BY SERVICE BULLETIN 74-9
[2] AIRPLANES NA-380-1 AND -2 CHANGED BY SERVICE BULLETIN 74-9 AND NA-380-3 AND LATER
[3] AIRPLANES CHANGED BY SERVICE BULLETIN 75-1
[4] AIRPLANES NOT CHANGED BY SERVICE BULLETIN 75-1
[5] AIRPLANES CHANGED BY SERVICE BULLETIN 75-29
[6] AIRPLANES NOT CHANGED BY SERVICE BULLETIN 75-29
S370-2-33-1B

Figure 32-00-4 — Landing Gear System — S/N 380-1 Thru -4, -6 Thru -9 and -11 (Sheet 1 of 2)

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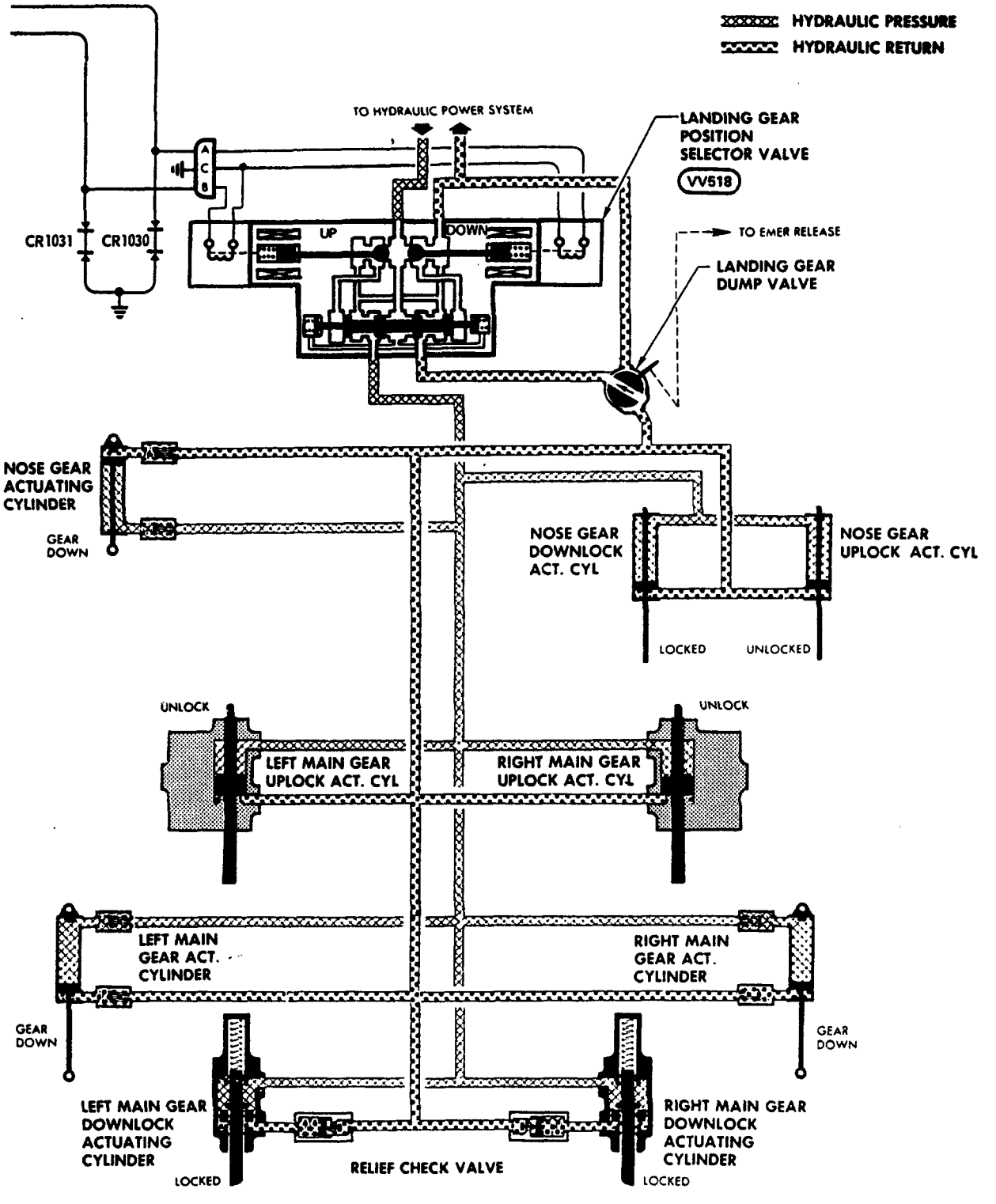

Figure 32-00-4 — Landing Gear System — S/N 380-1 Thru -4, -6 Thru -9 and -11 (Sheet 2 of 2)

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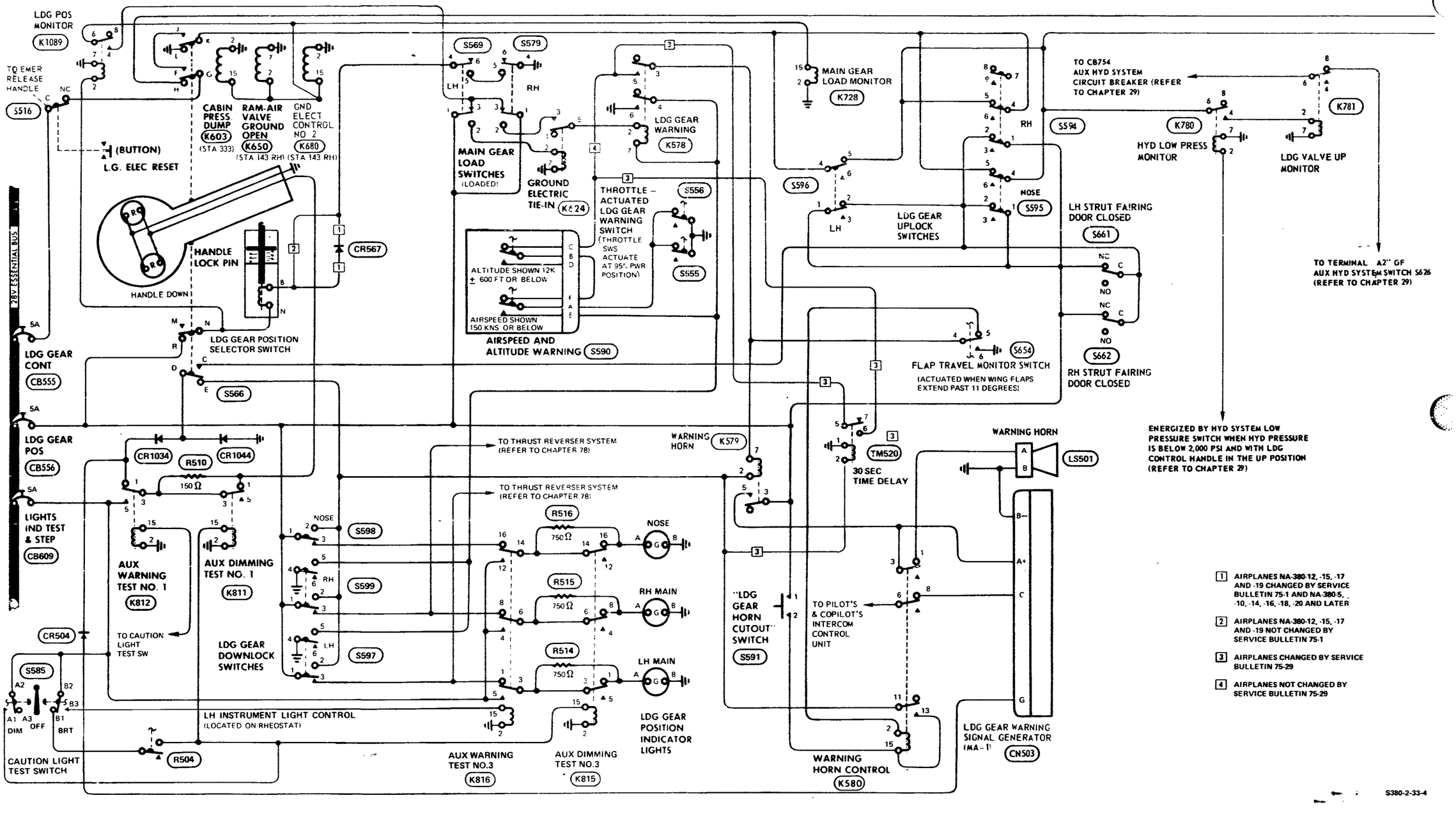

Figure 32-00-5 — Landing Gear System — S/N 380-5, -10, -12 and Subs (Sheet 1 of 2)

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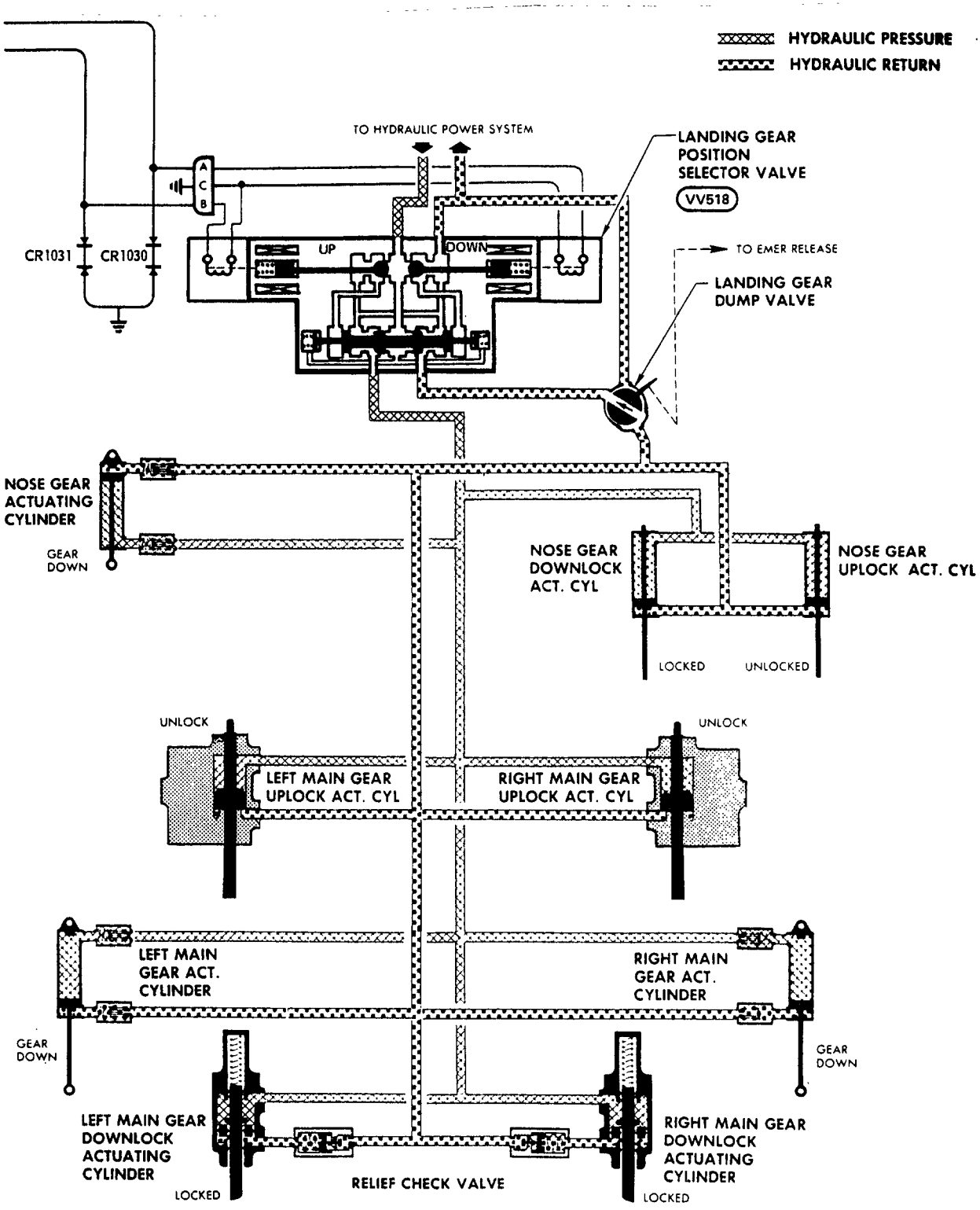

Figure 32-00-5 — Landing Gear System — S/N 380-5, -10, -12 and Subs (Sheet 2 of 2)

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LANDING GEAR - TROUBLE SHOOTING

1. Trouble Shooting Landing Gear System

The following equipment is needed to perform these procedures:

- Hydraulic test stand capable of delivering 4 gpm at 3000 psi
- 28-volt dc external electrical power
- Voltmeter capable of measuring 30 volts
- Continuity checker or equivalent
- Air Source for inflatable wheel seal - NA-370 airplanes not changed by Service Bulletin 72-8

NOTE: In the following trouble shooting procedure, the sequence of troubles may or may not occur as listed. It is suggested that frequent reference be made to the landing gear schematics when trouble shooting the system. This will aid in isolating a specific trouble.

- If a system unit is replaced, repaired, or adjusted, an operational check-out of the system is required.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. LANDING GEAR CONTROL HANDLE UP - GEAR WILL NOT RETRACT.		
(1) "LANDING GEAR - CONT" circuit breaker out (open).	Check that circuit breaker is pushed in (closed). If circuit breaker is out, electrical circuit is open.	Push in circuit breaker.
(2) "L.G. ELEC. RESET" switch open.	Check that "L.G. ELEC RESET" switch is reset. If switch is not reset, electrical circuit is open to landing gear position selector switch.	Reset "L.G. ELEC. RESET" switch.
(3) Faulty landing gear position selector valve.	Discharge hydraulic pressure. Disconnect landing gear position selector valve electrical plug. Check for voltage at pin "B" of electrical plug. If voltage is available, check continuity between pin "C" of electrical plug and ground. If there is no continuity, electrical circuit is open to ground. If there is continuity, electrical circuit to and from landing gear position	Repair electrical wiring (refer to Wiring Diagram Manual), or replace landing gear position selector valve.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
	selector valve is satisfactory and position selector valve is faulty.	
B. LANDING GEAR CONTROL HANDLE UP - GEAR UP - NOSE WHEEL FAIRING DOOR NOT LOCKED.		
(1) Nose wheel fairing door lock roller improperly adjusted.	Check adjustment of nose wheel fairing door lock roller. If door lock roller is improperly adjusted, door lock cannot engage door lock roller.	Adjust nose wheel fairing door lock roller.
(2) Voltage not available at pin "A" of landing gear position selector valve electrical disconnect plug, open circuit from pin "C" of electrical plug and ground, or faulty landing gear position selector valve.	Discharge hydraulic pressure. Pull landing gear emergency release. Disconnect landing gear position selector valve electrical plug. Place landing gear control handle at down. Check for voltage at pin "A" of electrical plug. If voltage is available, check continuity from pin "C" of electrical plug to ground. If there is no continuity, the electrical circuit is open to ground. If there is continuity, the electrical circuit is complete to and from the selector valve; therefore, the selector valve is faulty.	Repair electrical wiring (refer to Wiring Diagram Manual), or replace landing gear position selector valve.
C. LANDING GEAR HANDLE DOWN - SEALS DEFLATED (NA-370 AIRPLANES NOT CHANGED BY SERVICE BULLETIN 72-8) - GEAR WILL NOT EXTEND.		
(1) Faulty landing gear position selector valve.	Discharge hydraulic pressure. Disconnect electrical plug from valve. Check for voltage at pin "A" of plug. If voltage is available, check continuity between pin "C" and ground. If there is no continuity, circuit is open to ground. If there is continuity, circuit to and from landing gear position selector valve is satisfactory and valve is faulty.	Replace landing gear position selector valve.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
(2) Wheel seal pressure switch faulty. (NA-370 airplanes not changed by Service Bulletin 72-8)	If landing gear wheel seal fail light is on, pressure switch is faulty. If light is not on, go to probable cause (3).	Replace switch.
(3) Relay K1032 faulty (contacts stuck). (NA-370 airplanes not changed by Service Bulletin 72-8)	Disconnect plugs P527 and P528 from rear fuselage relay box and check for continuity between pin "B" of receptacle J512 and pin "C" of receptacle J513. If continuity does not exist, relay K1032 is faulty.	Replace relay.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
D. LANDING GEAR CONTROL HANDLE DOWN - SEALS DEFLATED (NA-370 AIRPLANES NOT CHANGED BY SERVICE BULLETIN 72-8) - NOSE GEAR NOT DOWN AND LOCKED.		
(1) Binding or incorrectly adjusted nose gear actuating cylinder.	Discharge, hydraulic pressure. Pull "LANDING GEAR - CONT" circuit breaker. Disconnect nose gear actuating cylinder and check adjustment. Disconnect hydraulic lines at both ends of actuating cylinder. Check actuating cylinder for binding by pushing piston in and out of actuating cylinder. If actuating cylinder is correctly adjusted and is not binding, proceed to next probable cause. An incorrectly adjusted or binding actuating cylinder could prevent correct operation of gear.	Adjust or replace nose gear actuating cylinder.
(2) Interference between nose gear downlock pins and lower drag brace.	Visually check for interference between nose gear downlock pins and lower drag brace. If there is no evidence of interference, proceed to next probable cause. Interference could prevent locking nose gear down.	Replace nose gear drag brace.
(3) Interference between downlock pin and downlock ramp.	Check clearance between downlock pins and downlock ramp. If there is correct clearance, proceed to next probable cause. Incorrect clearance will prevent nose gear from locking down.	Adjust clearance between lower surface of downlock pin and downlock ramp.
(4) Binding downlock pin, broken spring in downlock assembly.	Pull "LANDING GEAR - CONT" circuit breaker. Using a suitable tool, manually push downlock pin into lock assembly, and release pin. Downlock pin should snap back to down-and-locked position. If pin does not return, backup spring in	Replace nose gear actuating cylinder and upper drag brace assembly

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
	lock assembly is broken. If pin extends part way, pin is galled and will bind.	
E. LANDING GEAR CONTROL HANDLE DOWN - MAIN GEAR NOT DOWN AND LOCKED.		
(1) Binding or incorrectly adjusted main gear actuating cylinder.	Discharge hydraulic pressure. Pull "LANDING GEAR - CONT" circuit breaker. Disconnect main gear actuating cylinder and check adjustment. Disconnect hydraulic lines at both ends of actuating cylinder. Check actuating cylinder for binding by pushing piston in and out of actuating cylinder. If actuating cylinder is correctly adjusted and is not binding, proceed to next probable cause. An incorrectly adjusted or binding actuating cylinder could prevent correct operation of the gear.	Adjust or replace main gear actuating cylinder.
(2) Interference between main gear downlock pin and main gear side brace.	Visually check for interference and incorrect clearance between main gear downlock pin and main gear side brace. If there is no interference and clearance is correct, proceed to next probable cause. Interference and incorrect clearance will prevent main gear from locking down.	Adjust clearance between main gear downlock pin and main gear side brace.
(3) Interference between main gear side brace assembly and downlock assembly casting.	Check for interference or incorrect clearance between main gear side brace and downlock assembly casting. If there is no interference and clearance is correct, proceed to next probable cause. If there is interference, the main gear will be prevented from locking down.	Adjust clearance between main gear side brace assembly and downlock assembly casting.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
(4) Binding downlock pin or broken downlock pin backup spring.	Pull "LANDING GEAR - CONT" circuit breaker. Disconnect side brace hinged fairing door. Manually push downlock pin into downlock assembly casting, and release pin. If downlock pin is easily retracted and does not quickly extend, the downlock pin backup spring is broken. If the pin only partially extends or slowly extends, the pin is binding.	Replace downlock pin assembly.
F. LANDING GEAR CONTROL HANDLE DOWN - GEAR DOWN - POSITION INDICATORS SHOW ALL GEARS ARE NOT DOWN AND LOCKED.		
(1) "LANDING GEAR - POS" circuit breaker out.	Check position of "LANDING GEAR - POS" circuit breaker. If circuit breaker is not pushed in (closed), the electrical circuit to all landing gear position lights will open.	Push in "LANDING GEAR - POS" circuit breaker.
G. LANDING GEAR CONTROL HANDLE DOWN - GEAR DOWN - NOSE GEAR POSITION INDICATOR SHOWS NOSE GEAR NOT DOWN AND LOCKED.		
(1) Indicator lamp burned out.	Check indicator lamp bulb.	Replace bulb.
(2) Nose gear downlock switch incorrectly adjusted or faulty.	Check adjustment of nose gear downlock switch. If switch is incorrectly adjusted, the circuit will be open to nose gear position indicator light. If nose gear downlock switch is correctly adjusted, the switch is faulty.	Adjust or replace nose gear downlock switch.
H. LANDING GEAR CONTROL HANDLE DOWN - GEAR DOWN - LEFT OR RIGHT MAIN GEAR INDICATOR POSITION LIGHTS INDICATE MAIN GEAR NOT DOWN AND LOCKED.		
(1) Indicator lamp burned out.	Check indicator lamp bulb.	Replace bulb.
(2) Main gear downlock switch incorrectly adjusted or faulty.	Check adjustment of main gear downlock switch. If switch is incorrectly adjusted, electrical circuit to main	Adjust or replace main gear downlock switch.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
	gear position indicator light will be open. If switch is correctly adjusted, switch is faulty.	
I. RED CAUTION LIGHT IN LANDING GEAR CONTROL HANDLE DOES NOT GO OUT - NA-370-2 THROUGH -6 AIRPLANES CHANGED BY SERVICE BULLETIN 72-6; NA-370-1, 7, 8, AND 9 AND NA-380-1 AND LATER AIRPLANES.		
(1) Faulty main landing gear strut door position switches.	With landing gear retracted, check for 28-volt dc at common terminal of S661 and S662. If voltage is present switch(es) is faulty.	Replace switch(es).
J. WHEEL SEALS WILL NOT INFLATE WITH GEAR UP AND LOCKED - NA-370 AIRPLANES NOT CHANGED BY SERVICE BULLETIN 72-8.		
(1) Faulty seal pressure regulator valve.	Check for 28-volt dc at pin "A" of valve plug P887, and check for continuity to ground at pin "B" of P887. If voltage and/or continuity exist, valve is faulty.	Replace valve.
(2) Faulty control relay, K1033.	Perform voltage and continuity check of relay K1033 circuitry.	Replace relay K1033.
(3) Faulty or improperly adjusted gear uplock switches.	If gear position indicator lights are working properly, adjustment of switches is O.K. and switch is faulty. Check adjustment. (See figure 32-60-403 and 32-60-405.)	Adjust or replace gear uplock switch.
K. LANDING GEAR WHEEL SEAL FAIL CAUTION LIGHT STAYS ON - LANDING GEAR DOWN - WHEEL SEALS DEFLATED - NA-370 AIRPLANES NOT CHANGED BY SERVICE BULLETIN 72-8.		
(1) Faulty seal pressure switch.	Perform continuity check at pins "A" and "B" of switch electrical receptacle. If continuity exists, switch is faulty.	Replace switch.
(2) Time-delay relay TM502 faulty.	Disconnect plugs P527 and P528 from rear fuselage relay box and check for continuity between pin "C" of J512 receptacle and pin "D" of J513 receptacle. If continuity exists, relay is faulty.	Replace relay.

LANDING GEAR

MAINTENANCE PRACTICES

WARNING

On S/N 370-2 thru -6, -8, -9 and 380-1 and subs, before applying landing gear hydraulic pressure for checks, adjustments, etc., deplete normal and auxiliary accumulator pressure. This will prevent the auxiliary accumulator power assist retraction system from operating and prevent serious injury to personnel and/or damage to the airplane.

GENERAL

In some cases, replacement hydraulic parts or units will not come equipped with the various AN plumbing fittings required for their installation. It is suggested that, before removal of these fittings from the unit to be replaced, the location and position of each be recorded. This aids in correctly positioning fittings when they are installed on replacement parts.

BLEEDING LANDING GEAR SYSTEM

If landing gear components have been replaced, system lines have been disconnected or the gear has been moved by means other than hydraulic pressure, the system must be bled prior to operation with full pressure.

CAUTION

Air in the main landing gear actuators can cause failure of the cylinders if actuated under full pressure. Bleeding air from the system must be accomplished at the minimum operating pressure, to prevent damage.

1. Check fluid level of hydraulic system reservoir.
2. Deplete normal and auxiliary accumulator hydraulic pressures, before applying hydraulic pressure.
3. Bleed air from the system by operation of the landing gear at pressures not to exceed 1800 psi for a minimum of five complete cycles.
4. Service hydraulic system reservoir.

SERVICING LANDING GEAR SHOCK STRUTS

NOTE

The nose gear shock strut is serviced from the nose gear wheel well; the main gear shock struts are serviced from the main gear wheel wells. The main gear shock struts should be serviced in pairs. At the top of each gear strut is a high-pressure valve assembly, which is used to inflate and deflate the shock strut. Care must be used when the struts are deflated, to prevent injury to personnel or damage to the airplane by the sudden collapse of the struts. Before servicing, strut remove any obstructions from under fuselage and wings that might cause damage to those parts when airplane is lowered.

Filling Landing Gear Shock Struts

1. Place airplane on jacks and raise enough to permit retraction.
2. Remove strut air filler valve cap, and slowly loosen 3/4-inch hexnut until air is released at desired rate.
3. Remove complete air filler valve assembly.
4. With strut in normal ground position and fully compressed, fill strut with hydraulic fluid (Military Specification MIL-H-5606) to level of filler hole.
5. Extend and fully compress strut three times to free air that may be trapped in strut. Check fluid level and, if necessary, repeat step 4.

NOTE

Proper fluid level is achieved with strut fully compressed.

6. Replace air filler valve assembly. Tighten large hexnut to 100-110 inch-pounds. Safety air filler valve hexnut with lockwire.
7. Inflate strut to correct pressure.

Inflating Landing Gear Shock Struts

WARNING

To prevent possible explosion that could result in serious injury or death to personnel, do not inflate landing gear shock struts with oxygen.

1. Place airplane on jacks and raise enough to permit retraction.
2. Remove air filler valve cap, located on top of strut.
3. Attach air filler chuck to air filler valve.

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MAINTENANCE PRACTICES (Continued)

Inflating Landing Gear Shock Struts (Cont' d)

4. Loosen small hexnut one to two turns counterclockwise.

NOTE

It is recommended that one nitrogen bottle regulated to 459 psi be used to simultaneously supply nitrogen to both main gear struts utilizing two air filler chucks joined to a "T" line to the nitrogen regulator. If this recommended procedure is not followed, an unbalanced strut inflation condition may persist.

5. With nose wheel clear of ground and strut fully extended, inflate strut to 180 psi. With main gear wheels clear of ground and struts fully extended, inflate struts to 459 psi.
6. Tighten hexnut to 50-70 inch-pounds. Shut off air filler valve and remove air filler chuck.
7. Check valve for leaks, and install and tighten valve cap.
8. Lower aircraft & remove jacks.

NOTE

If unbalanced condition still occurs check servicing of struts fwd & aft.

LUBRICATING LANDING GEAR SYSTEM

Proper lubrication is as important to the operation of the landing gear system as proper adjustment. If a component is replaced and requires lubrication, it should be lubricated upon installation. Many components are permanently damaged if they are not properly lubricated.

1. For nose landing gear system lubrication, refer to chapter 12.
2. For main landing gear system lubrication, refer to chapter 12.

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INSPECTION/CHECK

OPERATIONAL CHECKOUT OF LANDING GEAR SYSTEM

Equipment/Tools

- Hydraulic test stand capable of delivering 4 gpm at 3000 psi (if available)
- 28 VDC external electrical power source
- Stop watch or equivalent
- Jacking and mooring kit
- Set of jacks (three)

Operational checkout of the landing gear system is divided into three checkout procedures:

- Checking operation of landing gear
- Checking operation of landing gear position indicating and warning system
- Checking operation of landing gear emergency system

Checking operation of landing gear consists of making a normal landing gear retraction and extension check. Checking operation of the landing gear position indicating and warning system consists of an operational check of the landing gear position light and an operational check of the warning light in the gear control handle and the audible warning system. (Refer to procedures for checking operation of landing gear position indicating and warning system in 32-60-02.) Checking the operation of the landing gear emergency system consists of making an emergency extension and retraction check of the landing gear. (Refer to procedures for checking operation of landing gear emergency system in 32-30-01.)

NOTE

An operational check of the landing gear system must be performed if a unit has been replaced, if hydraulic lines have been disconnected, or if an adjustment has been made on any unit. The checkout or checking procedure must be thorough enough to ensure that the system is free of hydraulic leaks and that no malfunction could occur because of improper maintenance.

General Procedure Before Landing Gear System Operation

1. Position jacks under airplane and raise airplane to permit landing gear retraction.

WARNING

Deplete normal and auxiliary accumulator hydraulic pressures when through using hydraulic pressure. This will ensure that the auxiliary accumulator power assist retraction system will not operate when the gear handle is placed in the up position. This will prevent serious injury to personnel and/or damage to airplane.

2. If a hydraulic test stand is to be used, open hydraulic system ground test panel access door at left side of airplane, aft of wing trailing edge, and connect hydraulic test stand to hydraulic system.

NOTE

If several operational sequences are required, a hydraulic test stand should be used to prevent overheating of hydraulic pump. The cycle time limits of the landing gear are based on the use of a hydraulic test stand.

3. Check fluid level of hydraulic system reservoir.
4. Remove landing gear maintenance safety locks.
5. Connect 28 VDC external electrical power.

CAUTION

To prevent possible damage to the landing gear audible warning signal generator, do not allow generator to operate more than 2 minutes.

NOTE

When warning horn is not being checked, push throttles forward to prevent horn operation.

Checking Operation of Landing Gear

1. Push in LANDING GEAR — CONT and LANDING GEAR — POS circuit breakers.
2. Position CABIN AIR switch to OFF.
3. Start hydraulic test stand and regulate flow to 4 gpm at a maximum pressure of 1850 psi.
4. Move LDG GEAR handle UP.
5. The nose gear should retract and lock. Check that nose gear strut is fully extended, wheels centered, and gear properly retracted into wheel well. The main gear should retract until the main gear uplock rollers contact the uplock hooks.
6. With the main gear in contact with the uplock hooks, test stand pressure may be increased until the main gear locks in the uplock hooks. Hydraulic pressure required to lock the main gear in the uplock hooks should not exceed 2300 psi.

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INSPECTION/CHECK (Continued)

Checking Operation of Landing Gear (Cont'd)

7. Move landing gear handle down. The landing gear should extend and lock.
8. Using a stop watch, check time required to retract and extend landing gear as follows:
   A. Regulate test stand to 4 gpm at 3000 psi.
   B. Move LDG GEAR handle UP. When gear retracts and landing gear warning light goes out, move landing gear handle down. A complete retraction or extension cycle should take no longer than 11 (±2) seconds. A complete cycle should take no longer than 18 (±3) seconds.
9. Cycle landing gear and visually check landing gear warning lights in cockpit. Visually check operation of landing gear.
10. Extend landing gear.
11. On S/N 370-1, -7, and 370-2 thru -6, -8 and -9 changed by SOK 11, and 380-1 and subs, proceed as follows:
   A. Pressurize normal and auxiliary system to normal operating pressure; then, turn off test stand or the aircraft hydraulic pump.
   B. Move LDG GEAR handle to UP. Check that landing gear retracts and locks, and note pressure drop on auxiliary accumulator pressure gauge.
   C. Move LDG GEAR handle to DOWN and pressurize system as required. Ensure that main landing gear is down and locked.

Checking Operation of Landing Gear Control Handle

NOTE

This procedure should be performed subsequent to the landing gear operational check. To check the landing gear control handle uplock and downlock devices for excessive wear and/or damage, proceed as follows:

1. Ensure that electrical and hydraulic power are *not* connected to airplane.

WARNING

Before applying landing gear hydraulic pressure for checks, adjustments, etc., deplete the normal and auxiliary accumulator pressure. This will prevent the auxiliary accumulator power assist retraction system from operating and prevent serious injury to personnel and/or damage to the airplane.

WARNING

Maintenance safety pins will not ensure a safe gear condition.

2. Check that landing gear maintenance safety pins are installed.
3. Pull out LANDING GEAR — CONT circuit breaker.
4. With LDG GEAR handle DOWN, apply a force, not to exceed 4 pounds, in the up direction. Landing gear control handle movement should not exceed 1/4 inch, measured from the center of the landing gear control handle knob.
5. Position LDG GEAR handle to UP, using downlock manual override button on landing gear control panel.
6. With landing gear control handle up and locked, apply a force, not to exceed 4 pounds, in the down direction. Landing gear control handle movement, measured from the center of the landing gear control handle knob, should not exceed 1/2 inch.
7. Position LDG GEAR handle to DOWN.
8. Push in LANDING GEAR — CONT circuit breaker.

General Procedure After Landing Gear System Operation

1. Shut down hydraulic test stand and check landing gear system for evidence of hydraulic leaks.
2. Disconnect hydraulic test stand and external electrical power.
3. Deleted
4. Install maintenance safety locks.
5. Lower airplane and remove jacks.
6. Service hydraulic system reservoir.
7. Push in all circuit breakers that were pulled.
8. Install all access doors.

LANDING GEAR POSITION AND WARNING SYSTEM — OPERATIONAL TEST (FLIGHT)

1. Descend through 12,000 (±600) feet at partial throttle with landing gear up and airspeed less than 140 [1] /150 [2] knots indicated airspeed.
2. Check that audible warning sounds at 12,000 (±600) feet.
3. Deleted
4. Resume normal flight.

[1] S/N 370-2 thru -6, -8 and -9
[2] S/N 370-1, -7, 380-1 and subs

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INSPECTION/CHECK (Continued)

INTERIOR INSPECTION OF MLG SHOCK STRUT PISTON

1. Jack the airplane to permit landing gear retraction. Refer to chapter 7.

CAUTION

Movement of the landing gear by means other than with hydraulic pressure may result in partial evacuation of hydraulic fluid from the actuator. Movement by other means may result in damage to the actuator during subsequent landing gear operations.

2. Remove the lock ring securing the hubcap to the wheel and remove the hubcap.
3. Remove the two bolts retaining the anti-skid generator to the axle, pull out the generator and disconnect the plug.
4. Remove the dust seal cap nut, washer and bolt underneath the strut and carefully remove the anti-skid wiring and plug from the axle. Tie-up (stow) the electrical wiring and the plug.
5. Place the HYDRAULIC PUMP switch to OFF.
6. Dump all normal and auxiliary hydraulic accumulator pressure.
7. Apply electrical power to the airplane and place the LDG GEAR control handle to the UP position.

WARNING

Before applying landing gear hydraulic pressure for checks, adjustments, etc., deplete normal and auxiliary accumulator pressure. This will prevent the auxiliary accumulator power assist retraction system from operating and prevent serious injury to personnel and/or damage to the airplane.

8. Cycle the HYDRAULIC PUMP switch as necessary to partially retract the main landing gear to approximately a 30-degree angle. Place a suitable support or chock under the main landing gears to support the gears.
9. With the HYDRAULIC PUMP switch at OFF, dump all accumulator pressures (see step 6).

NOTE

When the LDG GEAR control handle is DOWN, the nose gear will slowly extend and possibly lock in the down position.

10. Place the LDG GEAR handle DOWN. The main gears will settle against the chock or support installed in step 8.
11. Remove electrical power from the airplane and open (pull out) the LDG GEAR-CONT circuit breaker.
12. Clean and inspect the interior of the strut piston for corrosion.
13. Remove any corrosion using aluminum or stainless steel wool or sandpaper attached to a rod and operated by a low speed motor.
14. Inspect the area for cracks. Paint the reworked surface with one coat of wash primer, two coats zinc chromate, and one coat of aluminized lacquer paint.
15. Install the anti-skid generator into the axle, install the two retaining bolts through the bottom of the strut, and connect the plug to the generator. Lockwire the plug to the generator.
16. Position the anti-skid generator into the axle and install two retaining bolts through the wheel nut, axle and generator.
17. Install the dust seal cap underneath the strut (bolt, washer, and nut).
18. Close the LDG GEAR-CONT circuit breaker.
19. Move the LDG GEAR control handle to UP. Apply external electrical power to the airplane.
20. Cycle the HYDRAULIC PUMP control switch as required to partially retract the landing gear sufficiently to remove the chock or support from the main landing gear.

CAUTION

To prevent possible damage to retract cylinders, do not move cylinders by any means other than hydraulic power.

21. Remove the chock or support from the main landing gear.
22. Move the LDG GEAR control handle to DOWN and position the HYDRAULIC PUMP switch to AUTO or ON. Check that gear extends and locks (indicated by 3 green gear position lights coming on). Allow the hydraulic system to build up to maximum pressure. Then, position the HYDRAULIC PUMP switch to OFF.

CAUTION

Make certain that anti-rotation pin between the hubcap and the wheel is correctly aligned.

23. Perform operational check of the landing gear system.
24. Lower airplane to the ground and remove jacks.

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INSPECTION/CHECK (Continued)

INTERIOR INSPECTION OF MLG SHOCK STRUT PISTON (Cont'd)

25. Perform operational check of the anti-skid system and install the hubcaps.

NOTE

When installing hubcaps, ensure that cap engages anti-skid generator properly, and dowel pin on the rim of cap engages aligning hole properly.

26. Secure airplane as desired.

Inspect and Clean MLG Piston Interior Metering Pin Seat

1. Disassemble the main landing gear shock strut and piston. Refer to "Replacement of Main Gear Shock Strut Packing (O-rings) in this chapter.
2. Clean and dry the interior and exterior of the main landing gear piston using a suitable solvent. Refer to chapter 12.
3. Visually inspect the area of the piston where the metering pin seats for corrosion, rust, and/or other foreign material. If none is found, proceed to step 9.
4. Insert P/N 465-810010-001 MLG piston interior cleaning tool into the upper end of the lower piston bore and secure with P/N 127L013 piston nut.

NOTE

The MLG piston interior cleaning tool will deflect the glass beads onto the discrepant area and shield the remainder of the piston bore.

5. Bead blast the discrepant area using a nozzle of 1/2 inch diameter, glass bead per MIL-G-9954A, amendment no. 2. Size of bead to be DT-8, at an air pressure of 150 ±5 psi. Duration of bead blast to be 30 to 60 seconds as required. Rotate the MLG piston interior cleaning tool 1/2 turn and repeat.
6. Remove the MLG piston interior cleaning tool from the piston. Clean and dry the piston and ensure it is free of all bead blasting material and other contamination.
7. Inspect for notches, cuts, or scratches in the side of the barrel or excessive pitting on the land or in the radius. See Figure A for rework limits and the use of the measuring tool to determine thickness, size, and location of any discrepancies.
8. Apply a light coat of MIL-C-16173D, Grade 1 compound to the area cleaned by bead blast, and the balance of the inside bore of the lower piston below the metering pin seat.
9. Assemble MLG shock strut and piston. Refer to "Replacement of Main Gear Shock Strut Packing (O-rings)" in this chapter.

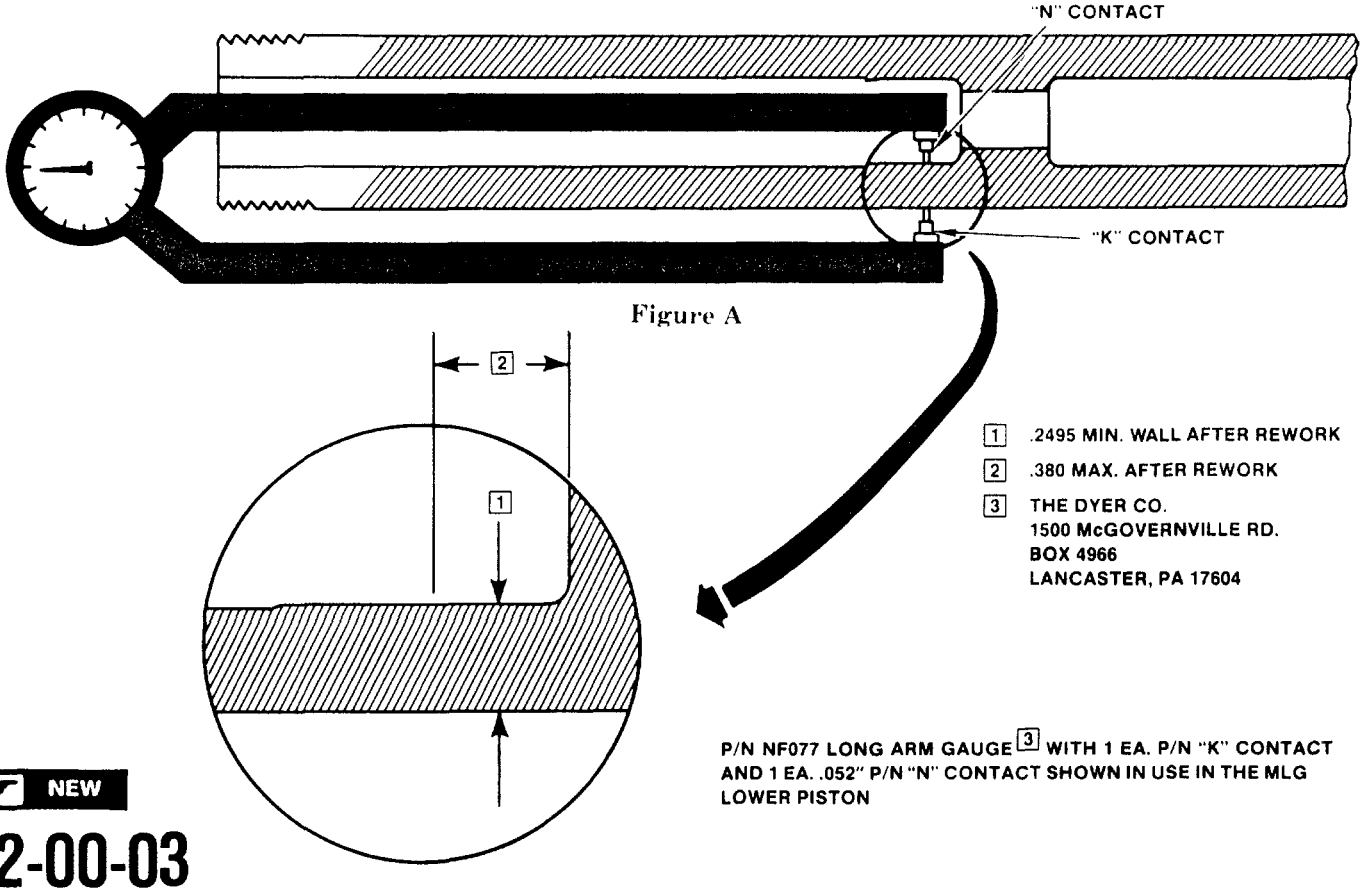
"N" CONTACT
"K" CONTACT
2
1
1 .2495 MIN. WALL AFTER REWORK
2 .380 MAX. AFTER REWORK
3 THE DYER CO.
1500 McGOVERNVILLE RD.
BOX 4966
LANCASTER, PA 17604
P/N NF077 LONG ARM GAUGE 3 WITH 1 EA. P/N "K" CONTACT AND 1 EA. .052" P/N "N" CONTACT SHOWN IN USE IN THE MLG LOWER PISTON

Figure A

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INSPECTION/CHECK (Continued)

INSPECT INTERIOR OF NOSE LANDING GEAR SHOCK STRUT PISTON

1. Jack airplane to permit landing gear retraction.

CAUTION

Movement of landing gear by means other than hydraulic pressure may result in partial evacuation of hydraulic fluid from the actuator. Movement by other means may result in damage to the actuator during subsequent landing gear operations.

2. Remove nose wheels, uplock fitting and balance weight bracket (bottom of strut).
3. Remove nose wheel axle and bearings.
4. Place HYDRAULIC PUMP switch to OFF position.
5. Dump all hydraulic accumulator pressure (normal and [1] auxiliary).
6. Apply electrical power to airplane. Move LDG GEAR handle to UP.
7. Cycle HYDRAULIC PUMP switch as necessary to partially retract (approximately one-half retracted) nose landing gear. Place suitable support or chock under nose landing gear to support it in a one-half retraction position.
8. With the HYDRAULIC PUMP switch at OFF, dump all hydraulic accumulators pressure. Refer to step 5.

NOTE

When the LDG GEAR handle is placed in the DOWN position, the main gear will slowly extend and possibly lock in the down position.

9. Place LDG GEAR handle to DOWN. The nose landing gear will settle against chock or support installation in step 7. The main landing gear will extend and possibly lock in the down position.
10. Remove electrical power from airplane and open (pull-out) LDG GEAR — CONT circuit breaker.
11. Clean and inspect interior of strut piston for corrosion.
12. Remove corrosion using aluminum or stainless steel wool or sandpaper attached to a rod and operated by a low speed motor.
13. Inspect area for cracks; then paint reworked surface with one coat wash primer and two coats zinc chromate and one coat of aluminized lacquer paint.
14. Close LDG GEAR — CONT circuit breakers.
15. Move LDG GEAR handle to UP. Apply external electrical power to airplane.
16. Cycle HYDRAULIC PUMP switch as required to partially retract landing gear sufficiently to remove chock or support from nose landing gear.

CAUTION

To prevent possible damage to retract cylinders, do not move landing gear by any means other than hydraulic power.

17. Remove chock or support from nose landing gear.
18. Move LDG GEAR handle to DOWN and position HYDRAULIC PUMP switch to AUTO or ON. Check that gear extends and locks (indicated by 3 green gear position lights being illuminated). Allow hydraulic system to build up to maximum. Position HYDRAULIC PUMP switch to OFF.
19. Install nose wheel axle, bearings, uplock fitting and balance weight bracket.
20. Install nose wheels.
21. Perform operation check of landing gear.
22. Lower airplane and remove jacks.

INSPECT NOSE LANDING GEAR AND DOORS

Inspect the following:

1. Nose gear doors for security of attachment; skin for cracks, buckles, dents, abrasions, wrinkles, corrosion, and loose rivets. Seal for security of attachment and cuts or tears.
2. Nose gear strut and trunnion for cracks, corrosion, damage, leakage, and specified extension.
3. Drag brace and attaching linkage for wear, cracks, corrosion and security.
4. Nose gear hydraulic actuating cylinder for security and leakage.
5. Steering potentiometers for security and cleanliness.
6. Nose wheel steering cylinder for leakage, security, and damage. Lines for chafing.
7. Steering linkage for security and wear.
8. Upper steering collar for cracks in radius of latch cut-out.
9. Lower steering collar for security of tow pin bushings; steering latch for security.
10. Upper and lower torque links for security; nose wheel retaining nuts for proper torque and safety.
11. Tires for wear, cuts, blisters, and specified inflation.

[1] S/N 370-1, -7, 380-1 and subs, or S/N 370-2 thru -6, -8 and -9 changed by SOK 11

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INSPECTION/CHECK (Continued)

INSPECT NOSE LANDING GEAR AND DOORS (Cont'd)

12. Wheels for damage, tie bolts for looseness and tie bolt nuts for cracks.
13. Nose gear balance weight for security; attaching bracket for security and cracks.
14. Switches and wire harnesses for condition, security and proper routing.

INSPECT MAIN LANDING GEAR AND DOORS

Inspect main landing gear and doors as follows:

1. MLG shock strut for leakage and specified inflation; torque links for security.
2. MLG doors (movable and fixed) for security of attachment; skin for cracks, buckles, dents, corrosion, and loose rivets; seal for cuts and tears.
3. MLG hydraulic actuating cylinders and hoses for condition, leakage, bent piston rods or ends, and security of attachment.
4. Uplock and downlock cylinders for leakage; uplocks, linkage for condition and security.
5. MLG wiring and operating linkages for condition and security; wiring for evidence of failure at switch housings and in areas subjected to flexing and bending.
6. Accessible emergency landing gear cables for corrosion, fraying and proper routing, turnbuckles for proper safety, accessible pulleys and brackets for damage.
7. Wheel for damage, cracks, loose tie bolts, and evidence of overheating in areas adjacent to brakes.
8. Tires for wear beyond acceptable limits, cuts, punctures and specified inflation.
9. Static ground wire for wear and ground contact.
10. Brake assembly, shuttle valve, hose and line connectors for condition, leakage, chafing and security of attachment.
11. Brake liner discs for excessive wear, brake pressure applied (any brake return pin which recedes 0.125 inch or more into spring housing indicates excessive wear).

INSPECT NOSE LANDING GEAR AXLE — ALL MODELS (See Figure 32-00-605)

NOTES

- *This inspection incorporates S/B 73-4, affecting S/N 370-1 thru -9.*
- *This inspection is terminated for all models when a -5 axle is installed.*

1. Raise nose gear wheels off the ground using the jack point on nose gear strut.
2. Remove nose wheels, axle and bearings.
3. Using adequate solvent, clean area of axle to be inspected.
4. Perform a Magnetic Particle (Magna-Glo) Inspection for cracks.
5. Install axle, bearings and nose wheels.
6. Check for proper installation and safeties.
7. Lower airplane and remove jack.

INSPECTION OF MAIN GEAR DOWNLOCK SUPPORT FORGING AND DOWNLOCK ASSEMBLY. AIRPLANES NOT CHANGED BY S/B 83-10, PART 2.

1. Remove all electrical power from the airplane.
2. Jack airplane per chapter 7 of Sabreliner Maintenance Manual.
3. Deplete all hydraulic system pressure.
4. Remove wing trailing edge access panels both main landing gear downlock areas.
5. Remove both main landing gear downlock assemblies from downlock support forgings per chapter 32 of Sabreliner Maintenance Manual.

WARNING

The main landing gear downlock assembly is spring loaded. Exercise extreme care to prevent injury when disassembling.

6. Disassemble, clean and inspect per steps A thru K, and figure 32-00-601.

NOTE

The flat sides of the downlock pin may be clamped in a padded vise to hold the assembly while removing the retainer nut. Hold downward pressure on the retainer nut until it is released from downlock body, then slowly release pressure until springs are relaxed.

A. Cut lockwire and remove 306-335004-3 upper retainer using tool shown in figure 32-00-601A or spanner wrench.
B. Clean 306-335002-9 (or -10) downlock body.
C. Inspect downlock for corrosion and allowable repair limits per figure 32-00-601B.

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INSPECTION/CHECK (Continued)

INSPECTION OF MAIN GEAR DOWNLOCK SUPPORT FORGING AND DOWNLOCK ASSEMBLY. AIRPLANES NOT CHANGED BY S/B 83-10, PART 2. (Cont'd)

D. Cut lockwire and remove 1N7-5A nut.

E. Depress 306-335513-11 pins, remove 306-335516-3 nut and slowly release pressure until spring is relaxed.

F. Clean and inspect piston and pin interiors for water and corrosion.

G. Using inside snap ring pliers, remove MS16625-1218 snap ring from 306-335004-3 retainer nut. Retain parts.

H. Remove and retain 7R4-29D retainer and 2W1AL105-69-40 washer.

J. Remove and discard MS28776-18 scraper and 2W12-29 felt washer.

K. Clean and inspect upper retainer P/N 306-335004-3 for corrosion and allowable repair limits per figure 32-00-601C.

NOTES

- ***If downlock assembly passes inspection requirements of steps C,F, and K, proceed to step 7 for reassembly.***
- ***If downlock assembly does not meet above inspection requirements, replace unacceptable part(s) with parts that meet inspection requirements or replace downlock assembly.***

7. Reassemble downlock assembly per steps A thru J and figure 32-00-601.

NOTE

Downlock assembly must be reassembled using new packings, retainers, scraper, and felt washer.

A. Lubricate all packings and felt washer with MIL-H-5606 hydraulic fluid.

B. Lightly lubricate center spring with MIL-G-81827 (AS) grease.

C. Lubricate 306-335513-11 pin at needle fitting with MIL-G-81827 (AS) grease before assembly.

D. Reassemble downlock assembly using figure 32-00-601 as a guide.

E. The 306-335515 sleeve must be installed per note 1 in Figure 32-00-601.

F. Torque 306-335516-3 nut to 190-210 inch-pounds.

G. Torque 306-335004-3 upper retainer to 200-300 inch-pounds.

H. Safety 306-335516-3 and 1N7-5A nuts to 306-335514-3 pin bolt using MS20995C41 safety wire.

J. Safety 306-335004-3 retainer nut to downlock body using MS20995C41 safety wire.

8. Install downlock assemblies per steps A thru C.

A. Coat bearing surface of downlock assemblies with MIL-C-16173D (Grade 1) corrosion preventative compound per figure 32-00-601D.

B. Install downlock assemblies in airplane per chapter 32 of Sabreliner Maintenance Manual.

C. Install wing trailing edge access panels.

9. Pressurize hydraulic system and leak check downlock assemblies.

10. Perform landing gear retraction check per chapter 32 of Sabreliner Maintenance Manual.

11. Lower airplane and remove jacks per chapter 7 of Sabreliner Maintenance Manual.

12. Remove electrical and hydraulic power from airplane.

INSPECTION OF MAIN GEAR DOWNLOCK SUPPORT FORGING AND DOWNLOCK ASSEMBLY.

1. Remove downlock assembly from downlock support forging as instructed in "Replacement of Main Gear Downlock Assembly" in this chapter.

2. Visually inspect downlock assembly for corrosion (rust deposits) and condition of paint. If corrosion is noted on downlock assembly, proceed to "Inspection of Main Gear Downlock Assembly" in this chapter, and refinish per Application of Corrosion Preventive Treatment to Downlock Support Forging and Downlock Assembly in this chapter.

3. Clean and visually inspect downlock support forging for evidence of corrosion and cracking. If evidence of corrosion and cracking is noted, proceed to "Inspection of Main Gear Downlock Support Forging" this chapter and remove, strip and further inspect the downlock support forging. If no corrosion or cracking is detected, proceed to step 4.

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INSPECTION/CHECK (Continued)

INSPECTION OF MAIN GEAR DOWNLOCK SUPPORT FORGING AND DOWNLOCK ASSEMBLY.

4. Touch up painted surface on downlock assembly and downlock support forging as necessary per "Application of Corrosion Preventive Treatment to Downlock Support Forging and Downlock Assembly" in this chapter.
5. Lightly coat all faying surfaces of the downlock assembly with corrosion preventative compound, (MIL-C-16173D (Grade 1)). See Figure 32-10-601D.
6. Reinstall downlock assembly in downlock support forging as instructed in Installing and Adjusting Main Gear Downlock Assembly, Figure 32-10-403.

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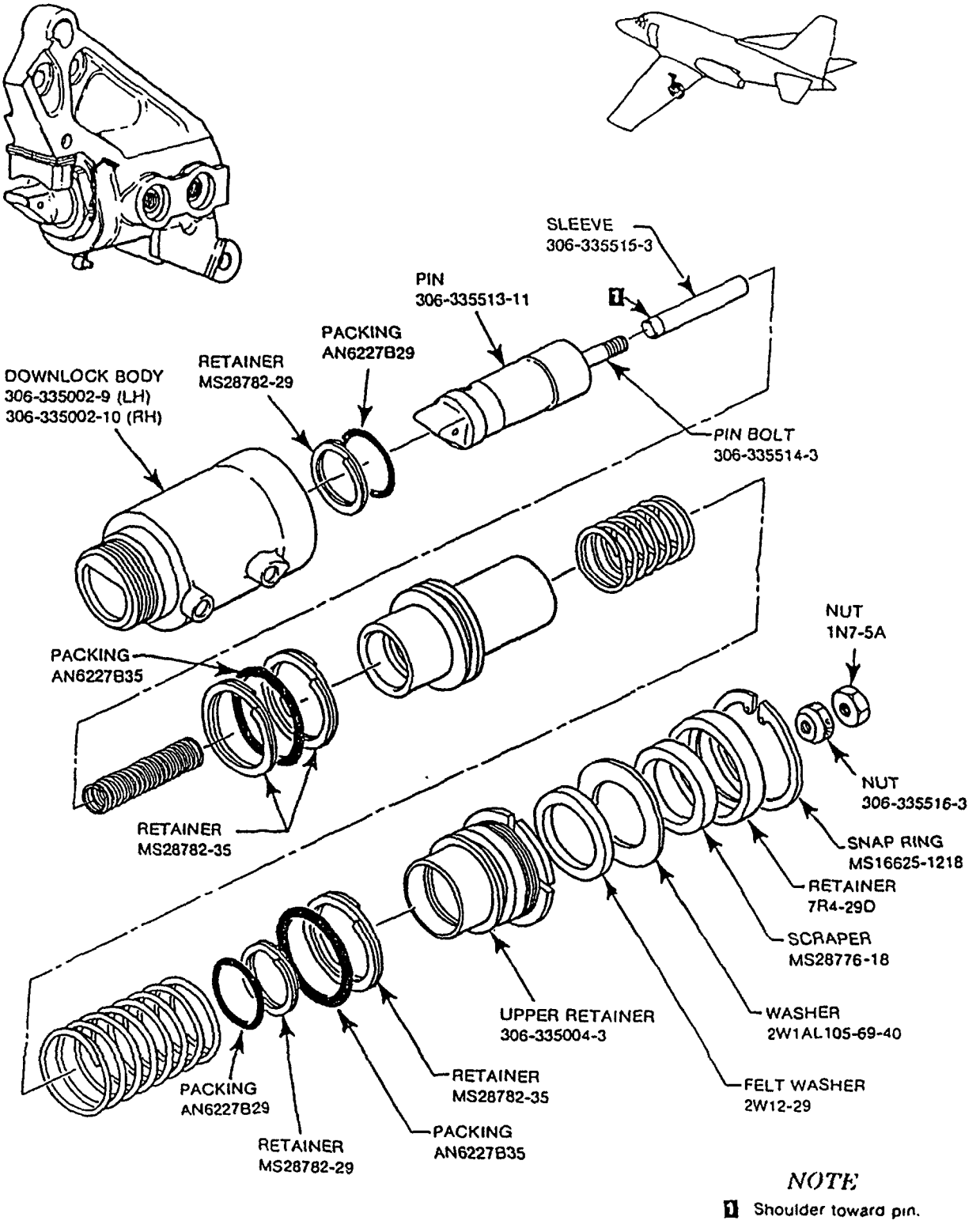

Figure 32-00-601 — Downlock Assembly Internal Breakdown

NEW

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NOTES

1. Radius .50 inch wide x .25 inch deep for clearance
2. Angles to be welded at intersection. Weld ½ inch drive adapter center back of intersections or 1 inch dia X 24 inch long pipe across back side of either angle.

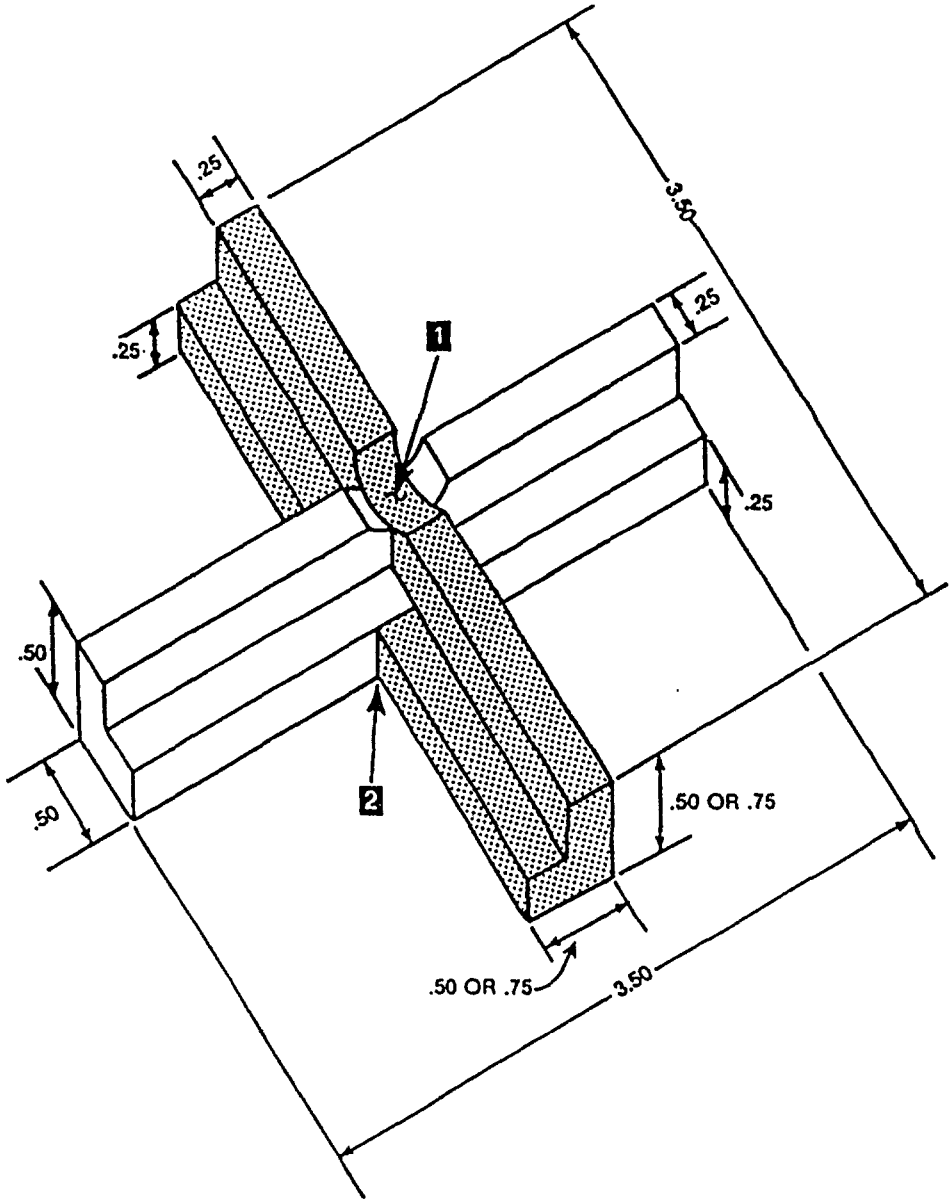

Figure 32-00-601A — Downlock Upper Retainer Removal Tool

NEW

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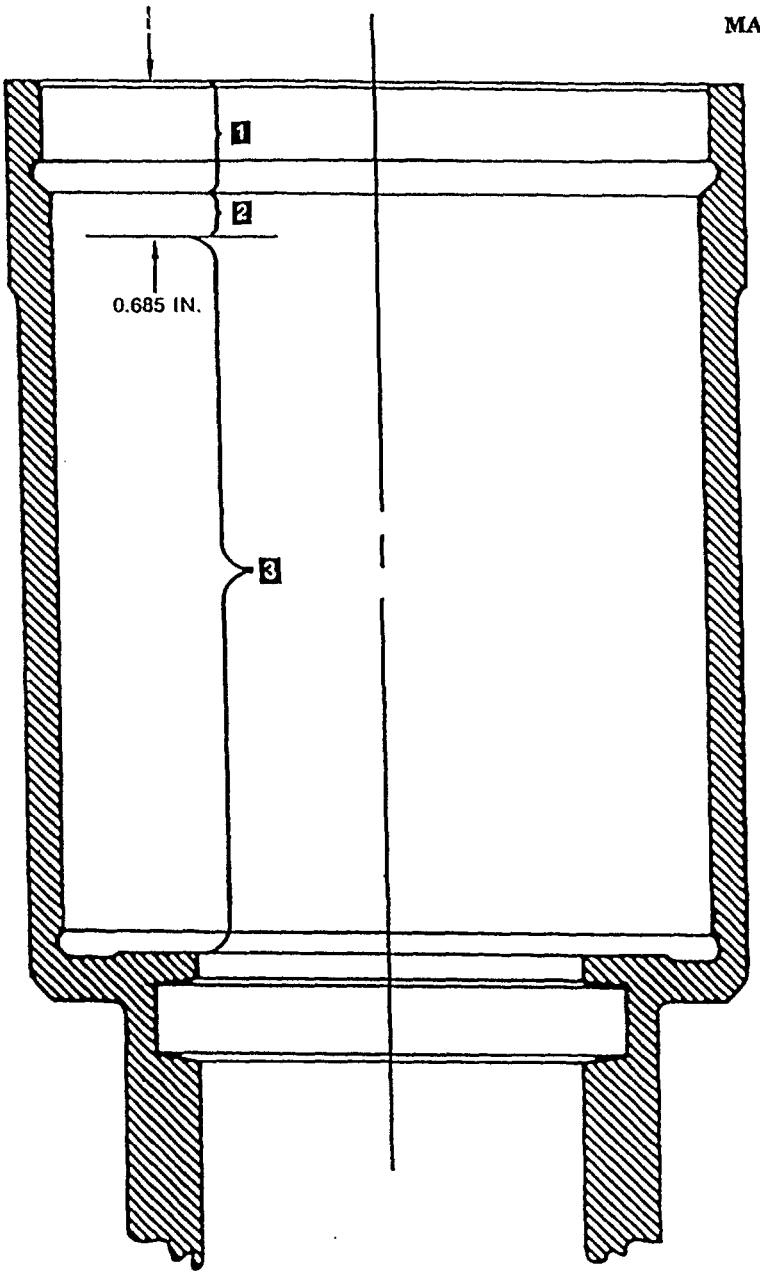

	LOCATION	CORROSION REPAIR LIMITS
1	UPPER THREAD AND THREAD RELIEF	NO CORROSION ALLOWED
2	BETWEEN LOWER END OF THREAD RELIEF AND 0.685 IN. DIMENSION FROM TOP OF BODY	MINIMUM WALL THICKNESS AFTER REPAIR — 0.209 IN.
3	BALANCE OF BODY BORE AND LOWER UNDERCUT	NO CORROSION ALLOWED

Figure 32-00-601B — Downlock Body Maximum Corrosion Repair Limits

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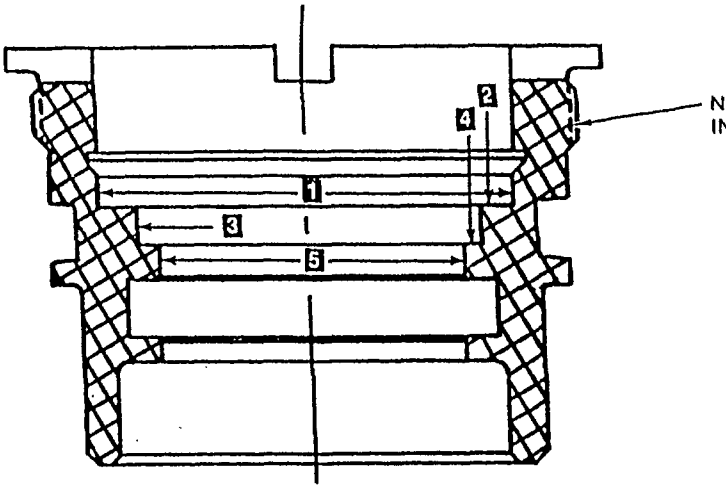
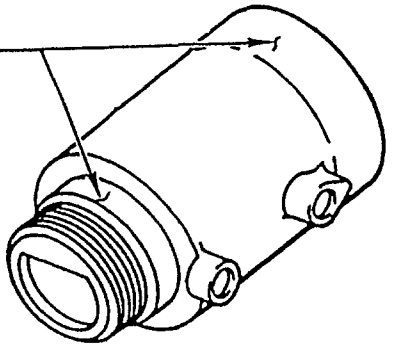


CORROSION
REPAIR LIMIT

1. 2.200 IN. DIA. MAX. AFTER REPAIR
2. MAX. MATERIAL REMOVAL: 0.030 IN. DEEP LOCALLY
3. MAX. MATERIAL REMOVAL: 0.030 IN. DEEP LOCALLY
4. MAX. MATERIAL REMOVAL: 0.010 IN. DEEP
5. 1.638 IN. DIA. MAX AFTER REPAIR

Figure 32-00-601C — Upper Retainer P/N 306-335004-3 Maximum Corrosion Repair Allowance

COAT BEARING
SURFACES WITH
MIL-C-16173D (GRADE 1)
CORROSION PREVENTATIVE
COMPOUND



CAUTION

Corrosion preventative compound not allowed in thread area.

Figure 32-00-601D — Application of Corrosion Preventative Compound

NEW

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INSPECTION/CHECK (Continued)

Inspecting Main Gear Downlock Support Forging

WARNING

When working in or around wheel well areas, use ground maintenance locks on MLG doors and MLG actuators and pin the nose gear down to prevent inadvertent retraction. Failure to do this could result in serious injury or death.

NOTE

Save all attaching hardware, fittings and components removed for reinstallation except the three EWSB26-9-A30 mounting bolts noted.

1. Remove bolt securing side brace idler link to downlock support forging. Note number and location of washers and spacers used.
2. Remove bolt securing strut door actuating rod to side brace clevis.
3. With strut door held or supported to prevent it from dropping, remove two bolts through strut door hinge and strut door hinge support. Remove strut door.
4. Remove downlock support forging from rear spar by removing three EWSB2-9-A30 bolts attaching the strut door hinge support and the downlock support forging to the rear spar.
5. Remove grease fitting from downlock support forging.
6. Remove side brace stop. Note number of shims installed under side brace stop.
7. Remove identification plate from downlock support forging by loosening adhesive with methyl ethyl ketone (MEK). Retain identification plate for reinstallation.
8. Remove all paint, grease and foreign material from surfaces of downlock support forging using a non-acid type of paint stripper (Turco 5351, or equivalent) and cleaning solvent.
9. Visually inspect all surfaces and bolt helicoil insert of the downlock support forging for evidence of cracks, corrosion pits, or deposits. Refer to Figure 32-00-601E for corrosion damage limits.
10. Remove all corrosion deposits from the downlock support forging using aluminum wool and/or 240 grit silicone carbide paper. Use caution not to exceed corrosion damage limits during corrosion removal procedure. If damage limits are exceeded, the downlock support forging should be replaced.
11. Dye penetrant inspect all exterior surfaces of the downlock support forging for evidence of cracks, paying particular attention to the hydraulic fitting attachment area. Any indications of cracking will require that the downlock support forging be replaced.

Inspecting Main Gear Downlock Assembly

1. Remove identification plate from downlock assembly by loosening adhesive with methyl ethyl ketone (MEK). Retain identification plate for reinstallation.
2. Using a non-acid type paint stripper (Turco 5351, or equivalent) and cleaning solvent, remove all paint, grease and foreign material from the downlock assembly.
3. Visually inspect exterior surface of the downlock assembly for evidence of rust.

NOTE

If rust is detected and has penetrated more than 0.005 inch in depth over an area exceeding one square inch, the downlock assembly should be replaced.

4. Remove all evidence of rust from downlock assembly using 240 grit silicone carbide paper.
5. Clean downlock assembly with cleaning solvent, then dry thoroughly.

Application of Corrosion Preventive Treatment to Downlock Support Forging and Downlock Assembly

1. Treat all bare surfaces of the downlock support forging with Alodine 1200 or 1201 (or equivalent), air dry for 15 minutes, then rinse with clean water and blow dry. The downlock assembly is steel and should not be anodized.
2. Mask all internal machined faying surfaces of the downlock support forging and all machined surfaces of the downlock assembly.
3. Paint all unmasked areas of the downlock support forging and the downlock assembly with epoxy polyamide primer. Allow primer to dry for 12 hours at an ambient temperature of between 60-90°F. Epoxy polyamide primer drying time may be accelerated by allowing primer to air dry for 30 minutes and then exposing to an elevated temperature of 130°F for one hour.
4. Paint all exterior surfaces of the downlock support forging with a finish coat of polyurethane top coat. Allow top coat to dry for 6 hours at an ambient temperature of between 60-90°F. Drying time for top coating may be accelerated by exposing the downlock support forging to an elevated temperature of 160°F for one hour, or 140°F for two hours or at 120°F for three hours.
5. Remove all masking tape and clean all previously masked surfaces.

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INSPECTION/CHECK (Continued)

Installing Main Gear Downlock Support Forging on Rear Spar

1. Secure identification plates to the downlock support forging and the downlock assembly using EC776 adhesive (or equivalent).
2. Reinstall side brace stop and shims. Make certain that the same number of shims are reinstalled as were removed. Torque screws to 50-70 inch pounds.
3. Reinstall grease fitting in downlock support forging.
4. Secure downlock support forging and strut door hinge fitting to rear spar using three new EWSB26-9-A30 bolts. Snug bolts up, do not tighten.
5. Install side brace idler link to downlock support forging. Make certain that the same number of washers are installed in the same locations as they were before removal. Do not tighten nut.
6. Tighten the three new EWSB26-9-A30 bolts securing the downlock support forging and the strut door hinge fitting to the rear spar. Torque bolts to 70 to 90 foot-pounds.
7. Install strut door to strut door fitting using the same bolts, washers, and nuts as removed. Tighten nuts until 0.005 to 0.010 inch end play is obtained on each attaching bolt.
8. Connect strut door attaching rod to side brace clevis. Torque nut to 290-410 inch-pounds. Make certain there is a large washer (LD153-0033-006) under the bolt head.
9. Tighten bolt securing side brace idler link to downlock support forging. Tighten nut to obtain 0.005 to 0.015 inch clearance between idler link washer and downlock support forging.
10. Lightly coat all faying surfaces of the downlock support forging and downlock assembly with corrosive preventative compound (MIL-C-16173D, Grade 1 or equivalent), see figure 32-00-601D, and reinstall downlock assembly in downlock support forging as instructed in "Installing and Adjusting Main Gear Downlock Assembly" on Figure 32-10-403.

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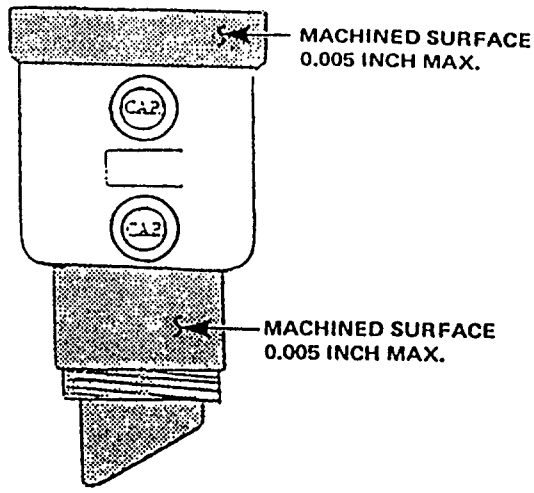
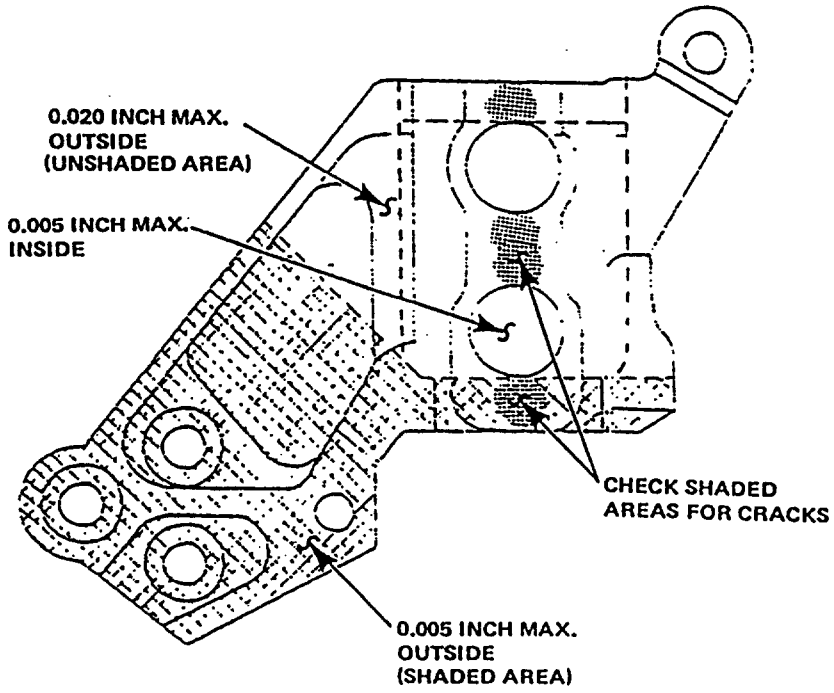


DOWNLOCK ASSEMBLY



DOWNLOCK CASTING

Figure 32-00-601E — Main Landing Gear Downlock Corrosion Damage Limits

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INSPECTION/CHECK (Continued)

INSPECTION OF MAIN LANDING GEAR ACTUATING CYLINDER ROD END AND CLEVIS PIN

Equipment/Tools

- Set of Jacks
- Jacking and Mooring Kit
- Hydraulic Test Stand Capable of Delivering 4 gpm at 3000 psi
- 28 VDC External Power

Inspect main landing gear and actuating cylinder rod end and clevis pin as follows, (typical for both LH and RH):

CAUTION

To prevent possible damage to retract cylinders, do not move cylinders by any means other than hydraulic power.

NOTES

- *An operational check of the landing gear system must be performed if a unit has been replaced, if hydraulic lines have been disconnected, or if an adjustment has been made on any unit. The operational check must be thorough enough to ensure that the system is free of hydraulic leaks and malfunctions that could occur because of improper maintenance.*
- *The following procedures may be performed with the normal hydraulic system electric pump; however, if several consecutive cycles are required for check, an external hydraulic test stand should be connected to the airplane.*
- *Landing gear must be in a half retracted position supported by a wheel chock or other suitable stop, to remove or install bolt attaching actuator cylinder rod end to clevis pin or strut.*
- *It is suggested that this inspection be performed concurrently with the landing gear system operational check.*

1. Check proper servicing of the normal system hydraulic accumulator, hydraulic reservoir and correct air pressure applied to the reservoir.
2. Position jacks under airplane and raise airplane to permit landing gear retraction.
3. Remove landing gear maintenance locks.
4. Pull the following circuit breakers:

   LH/RH ING-BOOST ..... LH Outbd Overhead
   INBD/OTBD VANE HEATER ..... LH Inbd Fwd Overhead
   LH PITOT HEAT ..... LH Console
   EPR-PITOT-STALL WARN ..... RH Outbd Overhead
   HTR CONTROL INBD/OTBD VANE HEATERS ..... RH Outbd Overhead
   RH PITOT HEAT ..... RH Console

   If systems installed, also pull out following breakers:

   AOA HEATER
   TAI/VSI/ALT ALERT

5. Push in the HYD PUMP CONT circuit breaker on the LH outbd overhead panel, and the LDG GEAR POS, LDG GEAR CONT circuit breakers on RH outbd overhead panel.
6. Connect external 28 VDC electrical power to airplane.

CAUTION

To prevent possible damage to the landing gear audible warning signal generator (MA-1), do not allow generator to operate more than 2 minutes.

NOTE

When warning horn is not being checked, push throttles fwd to prevent horn operation.

7. Check that wing flaps are in the retracted position.
8. Partially retract landing gear and support strut with a wheel chock, or other suitable means. Turn off hydraulic pressure and disconnect actuating cylinder rod end from strut attach point, then allow strut to extend into the down and locked position.
9. Loosen rod end nut, anti-rotation washer, and remove rod end from the piston rod.
10. Remove all paint from rod end.
11. Clean and inspect threads and zerk fittings boss areas of rod end using magnetic inspection method. Inspect carefully for cracks at the zerk and thread areas. Replace rod end assembly if suspected damage is noted.

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INSPECTION/CHECK (Continued)

INSPECTION OF MAIN LANDING GEAR ACTUATING CYLINDER ROD END AND CLEVIS PIN (Cont'd)

12. Inspect each flanged bushing P/N 4B20B7-006 for excessive circumferential abrasion wear and for ID of 0.4375 ±0.0005 in. If flanged bushings exhibit excessive circumferential abrasion wear or if ID is greater than 0.4380 in., replace entire rod end assembly (with new bushings installed) or install new bushings in existing rod end. Bushings are installed wet (zinc chromate primer) with an interference press fit into the rod end. Check ID of new bushings after installation or in a new rod end assembly. If new bushing ID is less than 0.4370 in., ream to 0.4375 ±0.0005 in.
13. Inspect actuating cylinder attach and attachment structure for visible damage. Check cylinder attach bolt, washer, and nut for 0.015 to 0.030 in. clearance between washer and face of fitting when washer is held against nut. Check that cylinder is free to swivel at cylinder attach end.
14. Install anti-rotation washer, nut, and rod end assembly on piston rod. Do not tighten nut or connect rod end to strut at this time.
15. Remove clevis pin from strut outer cylinder and perform a magnetic inspection. If clevis pin must be replaced, P/N 1127L003-1 is the preferred replacement part.

NOTES

- ***P/N 1127L003-1 clevis pin may be installed as a replacement for P/N 1127L003 clevis pin using the existing hardware; P/N 1127L026 thrust bearing, P/N 1127L004 sleeve bushing, and P/N 1127L005 flanged bushing. Ensure that cupped side of thrust bearing is facing toward the radius runout at head of clevis pin.***
- ***For main landing gear struts with P/N 1127L004-1 and 1127L005 flanged bushings installed into the upper cylinder, P/N 1127L026 thrust bearing is not required.***

16. Install thrust bearing (if required) on clevis pin and install clevis pin into shock strut.
17. Install lockwire washer (ensure washer flats engage clevis pin flats), and nut on clevis pin (torque nut to 100 in.-lbs). Verify that the clevis pin will rotate freely in the flanged and sleeve bushings with minimum axial travel. If clevis pin is not free to rotate, reduce torque on nut until free rotation is obtained. Max axial movement of clevis pin should not exceed 0.005 in. Safetywire nut to washer.

CAUTION

When installing an actuator and prior to connecting the actuator to the strut (connected in strut well only), position the actuator so that the rod end will clear adjacent structures. Hydraulically extend and retract (slowly with low pressure) the actuator several cycles to eliminate any trapped air. To prevent damage to the system, use a low hydraulic pressure during the cycle.

18. Position strut in the down and locked position. Apply hydraulic pressure and fully extend actuating cylinder. Adjust actuating cylinder rod end to obtain a slip-fit of actuating cylinder attaching pin when actuating cylinder is connected to strut clevis pin.

NOTES

- ***This is the initial adjustment of the actuating cylinder. The cylinder rod end does not need to be safetied at this time.***
- ***Disconnect the strut door link near inboard end of side brace.***
- ***Upstop bumper must be raised to prevent damage to upstop when step 23 is accomplished.***

CAUTION

Steps 23 through 26 (final adjustment) must be accomplished to prevent serious damage to retract cylinder and landing gear system.

19. Loosen lower jam nuts on upstop bumper by several turns of the nut. Then, tighten upper jam nut on bumper to raise bumper away from strut when it is retracted.

WARNING

Before applying landing gear hydraulic pressure for checks, adjustments, etc., deplete the normal and auxiliary accumulator pressure. This will prevent the auxiliary accumulator power assist retraction system from operating and prevent serious injury to personnel and/or damage to the airplane.

20. Install nose gear ground safety pin. This prevents nose gear from retracting while allowing 1800 psi to be maintained on the airplane for subsequent procedures.
21. Remove the uplock roller.

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INSPECTION/CHECK (Continued)

INSPECTION OF MAIN LANDING GEAR ACTUATING CYLINDER ROD END AND CLEVIS PIN (Cont'd)

NOTES

- ***While making rod end adjustments, verify that shaft is not bent by ensuring that clearance changes consistently with no gross increases or decreases. Half-turn movement should result in small, incremental variances in clearance. Large incremental variances indicates bent retract cylinder piston or shaft.***
- ***If rod end adjustment cannot be made (rod end bottoms in the cylinder piston rod), remove cylinder attach bolt, fully extend cylinder and check for 28.85 inches from center of inboard attach bolt to end of cylinder piston rod.***

22. Move the LDG GEAR handle UP. Slowly increase hydraulic pressure up to (and maintain) 1800 psi and retract gear.
23. With gear up and locked and 1800 psi hydraulic pressure, check clearance between the top of main gear outer cylinder and wing panel skin. Clearance should be 9/16 inch unless 1127L001-5 or -6 upper cylinder is installed, then clearance is 1/2 inch. (In order to make this measurement, it is necessary to remove a section of the fairing from the top of the wing where it joins the fuselage.) Shorten (or lengthen) main gear actuating cylinder rod end to obtain noted dimension when gear is retracted and actuating cylinder has 1800 psi pressure applied. (Rod end adjustment may be made with gear down and locked. Dump hydraulic pressure and rotate cylinder piston rod using a wrench on wrench flats at outboard end of piston rod.)
24. With hydraulic pressure off, disconnect landing gear actuating cylinder from strut. Position landing gear down and locked. Do not change actuating cylinder rod end adjustment. Apply hydraulic pressure to fully extend actuating cylinder, and check that there is at least 0.050 inch of actuator overstroke available. Check also that 1-5/8-inch dimension is not exceeded.

NOTE

Landing gear has to be in the half-retracted position against the wheel chocks or other suitable stops to remove or install bolt attaching actuator cylinder rod end to clevis pin on strut.

25. Turn off hydraulic pressure and reconnect actuating cylinder to strut. Tighten attach bolt nut as necessary to maintain a clearance of 0.005 inch minimum between actuator rod end bushing and clevis. Maximum torque is 100 inch-pounds. Safety nut with cotter pin.

NOTES

- ***If actuator clevis attach bolt was inserted upside down for rigging, re-install head-up.***
- ***Washers under nut may be removed if necessary to obtain 0.005-inch clearance.***

26. Tighten actuating cylinder rod end jam nut and safety to the anti-rotation washer with lockwire.
27. Install the uplock roller.
28. Check clearance between gear and uplock roller and throat of uplock hook. Clearance should be 1/16 inch. Add or remove shims between uplock roller support fitting and gear strut to obtain specified clearance when gear is up and locked.
29. With main gear up and locked and actuating cylinder pressurized to 1800 psi, check clearance between the bottom of gear uplock roller and uplock hook. Clearance must be 0.005 to 0.035 inch. Adjust gear uplock roller support bracket up or down on serrations to obtain specified clearance when gear is up and locked and hydraulic pressure on actuating cylinder is 1800 psi. Tighten roller support bracket attach bolts, and safety bolts with lockwire.
30. Loosen upper jam nut on upstop and allow bumper to rest, centered on top of strut. Tighten lower jam nut on upstop until nut surface contacts the lower surface of upstop support bracket. Tighten nut one more full turn (compressing the rubber bumper approximately 3/64 inch) while maintaining 1800 psi. Then, tighten upper jam nut to lock the adjustment.
31. Check the adjustment of uplock switch. Refer to Figure 32-60-403.
32. Extend landing gear. Safety jam nuts to each other and to upstop bumper with lockwire.
33. Remove nose gear ground safety pin.
34. Retract landing gear and shut off hydraulic pressure. Check that hydraulic lines on actuating cylinder clear all the adjacent structure.
35. Extend gear and check actuating cylinder for evidence of hydraulic leaks.
36. Connect and safety cannon plug on door selector valve in the left-hand wheel well, and remove ground safety locks.

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INSPECTION/CHECK (Continued)

INSPECTION OF MAIN LANDING GEAR ACTUATING CYLINDER ROD END AND CLEVIS PIN (Cont'd)

37. Run complete check of the landing gear. Check indicating and warning system, sequencing, and emergency operation.
38. With landing gear extended, check landing gear for installation of all safeties.
39. Remove electrical and hydraulic power.
40. Lower airplane and remove jacks.

INSPECT MLG SHOCK STRUT OUTER CYLINDER

1. Remove LH and RH MLG shock struts. Refer to Replacement of Main Gear Shock Struts, 32-10-01.
2. Disassemble both MLG shock struts (ref 32-10-01).
3. Clean all parts and repair or replace parts as described in the disassembly instructions.
4. Strip the outer cylinders using Turco 5351 or equivalent (MIL-R-25134B) per Sabreliner Corrosion Control Manual, SR-76-023.
5. Perform a magnetic particle (magna-glo) inspection per MIL-I-6868 on the entire exterior surface of the cylinder.
   A. If a crack is detected, refer to Structural Repair Manual, NA-66-1032, chapter 32-30-10 for repair limits and procedures.
6. Inspect the exterior surface of the cylinder and inner surface of the trunnion for corrosion damage.
   A. Corrosion can be removed to a maximum depth of .003 in. from the exterior surface of the cylinder and inner trunnion area. Do not remove material from machined areas except where specified by this manual and the Structural Repair Manual.
7. Inspect the inside surfaces of the side brace boss for corrosion damage and oversize.
   A. Sidebrace bosses with pits .005 in. deep or less and not more than 15% of the surface area may be cleaned by glass bead blasting and used provided the ID is within tolerance (see Figure 32-00-606). The dimensional difference between the ID of the boss and OD of the pin is not to exceed .0030 inches.
   B. If corrosion exceeds the limits, contact Sabreliner Corporation to have the bosses bushed back to acceptable dimensions.
8. Inspect the inside of the cylinder for corrosion and wear (see Figure 32-00-606).
   A. If corrosion is found between the top of the threads and 2.5 inches above the top of the threads, then the corrosion can be removed to a maximum depth of .005 inches. See Figure 32-00-606 for corrosion removal procedures.
9. Finish all cylinders where material was removed in the following way:
   A. Apply brush cadmium plate to polished area per MIL-STD-865.
   B. Bake cylinder at 375±25° for 23 hours.
   C. Phosphate treat exterior per specification TT-C-490 Type I.
   D. Mask bearing surfaces.
   E. Apply two coats of epoxy polymide primer (MIL-P-23377) to the exterior surface.
   F. Apply two coats of polyurethane. See Corrosion Control Manual, SR-76-023, for application of primer and top coats.
10. Reassemble, reinstall, and rig MLG shock strut assemblies per instructions in this chapter.
11. Perform Operational Check of MLG (ref 32-00-03).

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INSPECTION/CHECK (Continued)

INSPECTION OF MAIN LANDING GEAR DOWNLOCK SIDEBRACE IDLER ROD END

Equipment/Tools

- Set of Jacks
- Jacking and Mooring Kit
- Hydraulic Test Stand Capable of Delivering 4 gpm at 3,000 psi.
- 28 VDC External Power

CAUTION

To prevent possible damage to retract cylinders, do not move cylinders by any means other than hydraulic power.

NOTES

- ***An operational check of the landing gear system must be performed if a unit has been replaced, if hydraulic lines have been disconnected, or if an adjustment has been made on any unit. The operational check must be thorough enough to ensure that the system is free of hydraulic leaks and malfunctions that could occur because of improper maintenance.***
- ***The following procedures may be performed with the normal hydraulic system electrical pump; however, if several consecutive cycles are required for check, an external hydraulic test stand should be connected to the airplane.***

1. If main landing gear downlock sidebrace idler lower rod end is permanently identified with the numbers "81-7" on the lower outer surface, this inspection is not required (rod ends so marked are solid shank rod ends and do not require periodic inspection for cracks). (See Figure 32-00-604.)
2. Check proper servicing of the normal system hydraulic accumulator, hydraulic reservoir, and correct air pressure applied to the reservoir.
3. Position jacks under airplane and raise airplane to permit landing gear retraction.
4. Remove landing gear maintenance locks.
5. Pull out the following circuit breakers: LH and RH IGN BOOST CONTROL; OUTBOARD and INBOARD STALL WARN VANE HEATER; LEFT and RIGHT HEATERS; EPR-PITOT-STALL WARN CONTROL; RAM AIR INLETS: Also pull out the AOA HEATER, and the TAI/VSI/ALT ALERT circuit breakers if these systems are installed in airplane.
6. Push in the HYD PUMP CONT circuit breaker and the LDG GEAR POS, LDG GEAR CONT circuit breakers.
7. Connect external 28 VDC electrical power to airplane.

CAUTION

To prevent possible damage to the landing gear audible warning signal generator (MA-1), do not allow the generator to operate more than 2 minutes. Momentarily push in the landing gear warning horn cutout button when warning system is not being used. Before resuming operation, allow the generator to cool a minimum of 8 minutes.

8. Check that wing flaps are in the retracted position.
9. Place landing gear fairing door ground control switch in left main gear strut well at OPEN. Open fairing doors and install door ground safety locks.
10. Partially retract landing gear and support strut with a wheel chock or other suitable means.
11. Turn hydraulic power off.
12. Turn off battery and electric master switches.
13. Disconnect external electrical power.
14. Deplete hydraulic system pressure per chapter 12.
15. Remove main landing gear downlock sidebrace idler lower attach bolt (see Figure 32-00-604).
16. Using outside caliper or similar device, measure distance from center of rod end hole to sidebrace idler (see Figure 32-00-604.)
17. Loosen rod end nut and remove rod end from main landing gear sidebrace idler.
18. Inspect rod end as follows:
    A. If rod end shank is hollow or brass filled, magnaflux shank area for cracks. If any cracks are detected, rod end must be replaced. Solid shank rod ends are the preferred replacement part.
    B. If rod end shank is solid, inspection is not required. If not already so identified, permanently identify these rod ends as shown in Figure 32-00-604 to preclude unnecessary reinspection.
19. Install serviceable rod end and nut in main landing gear downlock sidebrace idler. Set rod end so that distance from the center of rod end hole to sidebrace idler is the same as measured in step 16 before rod end was removed. Dry torque jam nut to 75-90 in.-lbs.
20. Install downlock sidebrace idler to MLG downlock sidebrace using sidebrace idler lower attach bolt (see Figure 32-00-604).

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INSPECTION/CHECK (Continued)

INSPECTION OF MAIN LANDING GEAR DOWNLOCK SIDEBRACE IDLER ROD END (Cont'd)

21. Turn electrical power OFF. Placing LDG GEAR handle to UP. Turn electrical power ON. Cycle HYDRAULIC PUMP switch to partially retract strut to remove chock, or stop, holding strut in a partially retracted position.
22. Extend landing gear. Ensure that all gears are down and locked.
23. With MLG down and locked, ensure that downlock pin extends 25/32 in. into downlock sidebrace (see Figure 32-00-604). If this dimension is not met, re-rig MLG per chapter 32.
24. Remove fairing door ground safety locks and set door ground control switch at CLOSED.
25. Perform landing gear retract and extension tests.
26. With landing gear down and locked, shut off airplane electrical and hydraulic power.
27. Lower and remove jacks from the airplane.
28. Push in circuit breakers and secure airplane as required.

LANDING GEAR ALIGNMENT

A complete landing gear alignment check, using common hand tools, can be accomplished using procedures in the Sabreliner Structural Repair Manual, Report No. NA-66-1032.

INSPECTION — MAIN LANDING GEAR STRUT DOOR FOR CORROSION (Ref: S/B 93-2)

Equipment/Tools

- Borescope with 90° Viewing Probe

Consumables

- Primer MIL-P-23377
- For Part II, NAS1097DD6 Rivets

Procedure

NOTE

Compliance with S/B 93-2 is required prior to this inspection.

Part I

1. Inspect MLG Strut Door (P/N 370-333351) close-out areas, LH and RH, with Borescope using 4 inspection holes provided. See Figure 32-00-607 (Sheet 1 of 2).
2. If corrosion is found, proceed with Part II.
3. If no corrosion is found, touch up hole areas, using MIL-P-23377 Primer, and top coat as required.

Part II

To remove corrosion and/or add additional drain holes, proceed as follows:

1. Remove landing gear strut door assembly in accordance with Maintenance Manual instructions.
2. Remove close-out skin and inspect for corrosion on skin and in close-out area. For corrosion removal, refer to Sabreliner Corrosion Control Manual, SR-76-023. For reassembly, and/or additional drain hole installation, on P/N 370-333351 door assembly, refer to Figure 32-00-607 (Sheet 2 of 2).
3. Reinstall, rig and perform operational landing gear system check in accordance with the Sabreliner Maintenance Manual.

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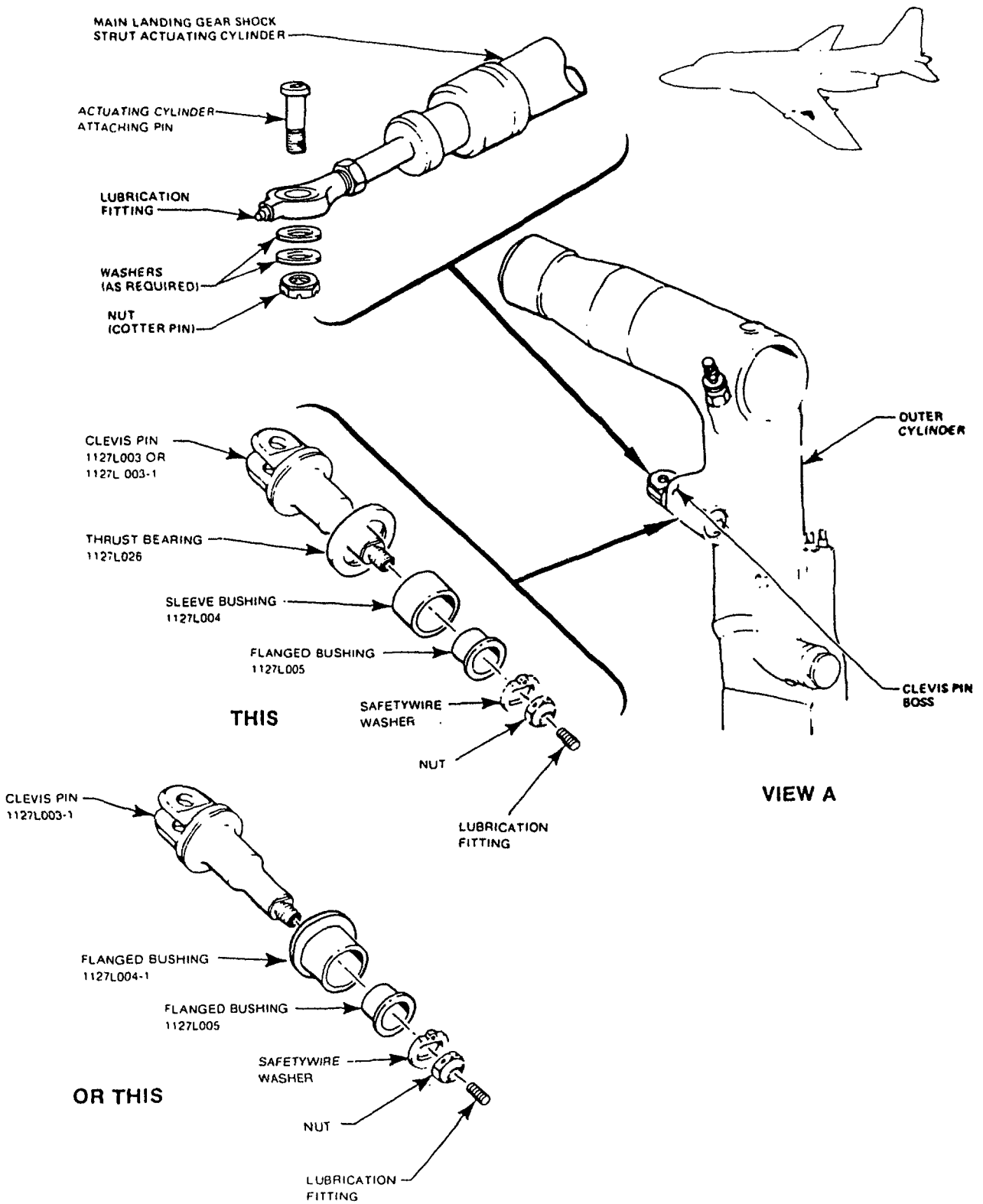
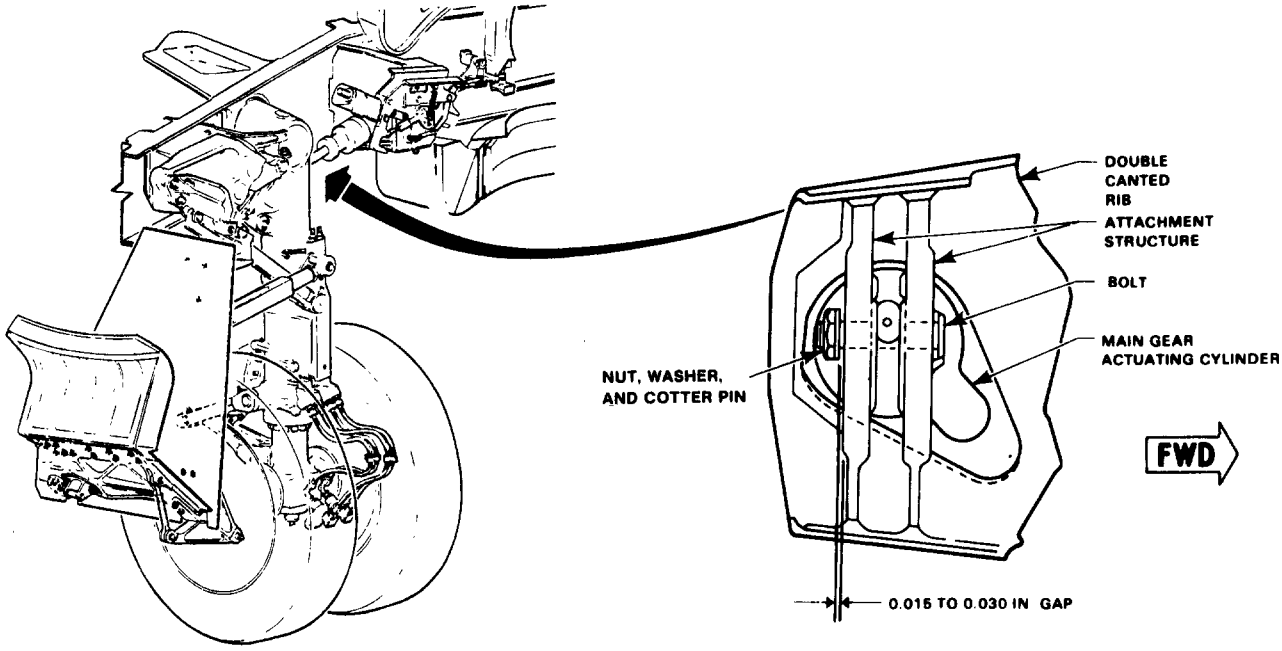
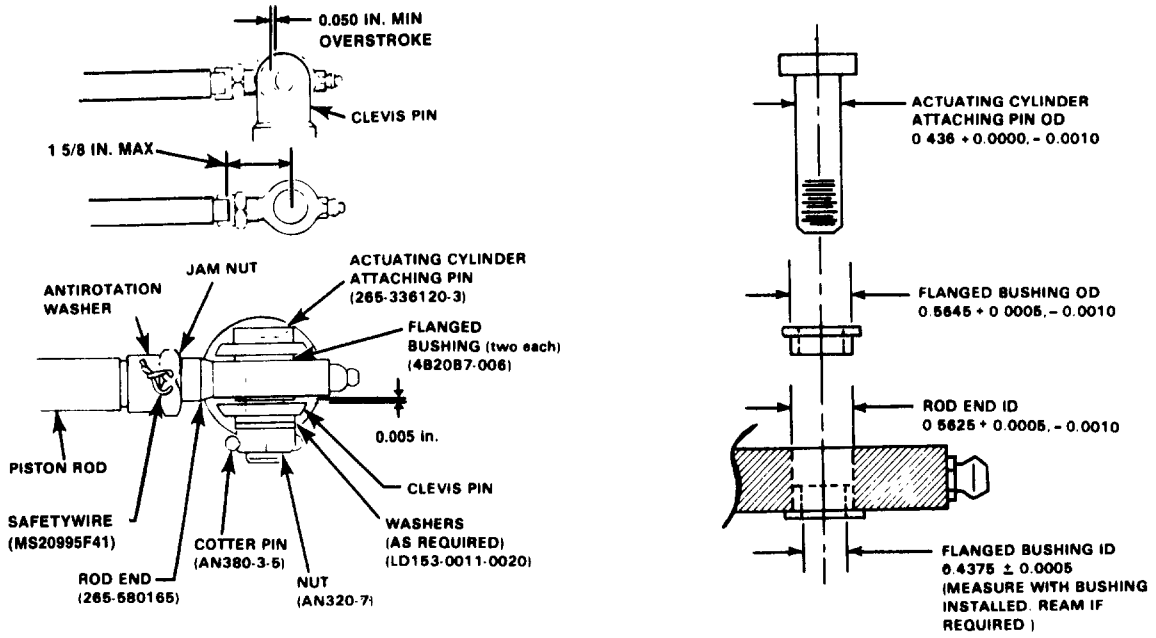


Figure 32-00-602 — Main Landing Gear Actuating Cylinder Rod End Inspection (Sheet 1 of 2)



VIEW B


0.050 IN. MIN OVERSTROKE
CLEVIS PIN
1 5/8 IN. MAX
JAM NUT
ACTUATING CYLINDER ATTACHING PIN (265-336120-3)
ANTIROTATION WASHER
FLANGED BUSHING (two each) (4B20B7-006)
0.005 in.
PISTON ROD
CLEVIS PIN
SAFETYWIRE (MS20995F41)
COTTER PIN (AN380-3-5)
WASHERS (AS REQUIRED) (LD153-0011-0020)
ROD END (265-580165)
NUT (AN320-7)
ACTUATING CYLINDER ATTACHING PIN OD 0.436 +0.0000, - 0.0010
FLANGED BUSHING OD 0.5645 + 0.0005, - 0.0010
ROD END ID 0.5625 + 0.0005, - 0.0010
FLANGED BUSHING ID 0.4375 ± 0.0005 (MEASURE WITH BUSHING INSTALLED. REAM IF REQUIRED.)

VIEW C

Figure 32-00-602 — Main Landing Gear Actuating Cylinder Rod End Inspection (Sheet 2 of 2)

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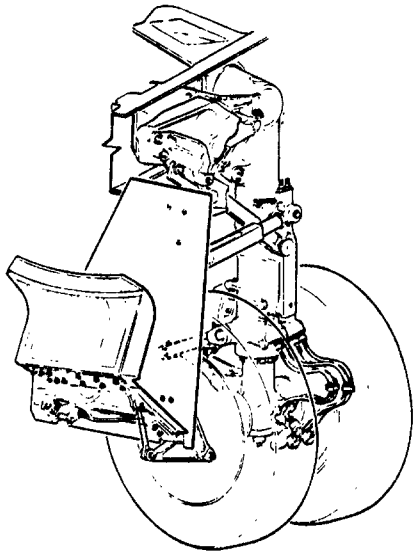
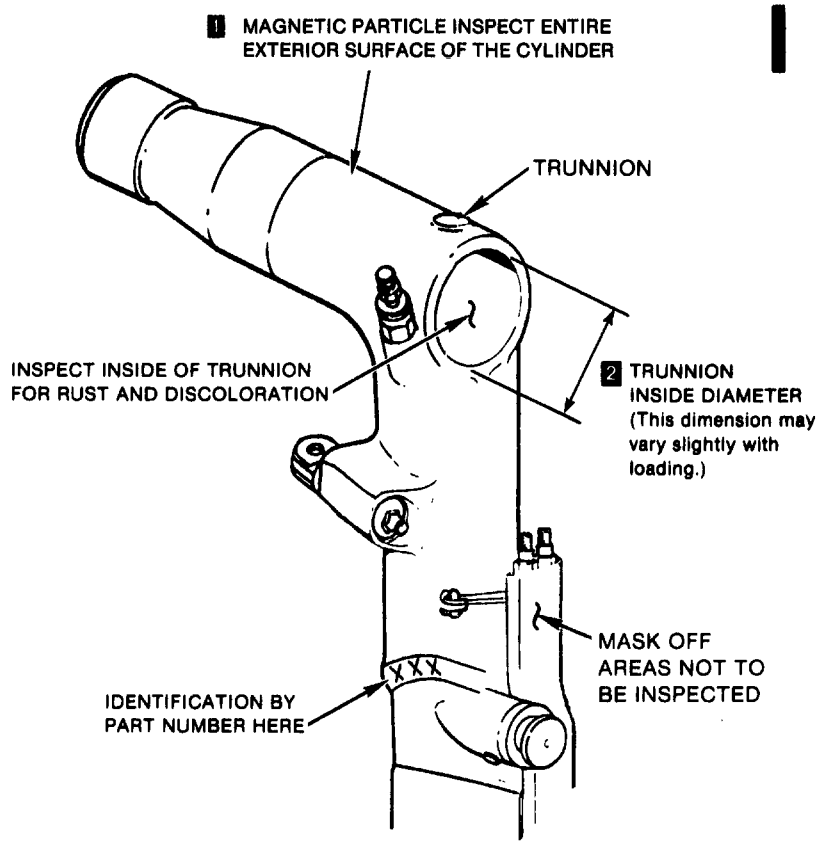


LEFT MAIN LANDING GEAR STRUT ASSEMBLY


1 MAGNETIC PARTICLE INSPECT ENTIRE EXTERIOR SURFACE OF THE CYLINDER
TRUNNION
INSPECT INSIDE OF TRUNNION FOR RUST AND DISCOLORATION
2 TRUNNION INSIDE DIAMETER (This dimension may vary slightly with loading.)
MASK OFF AREAS NOT TO BE INSPECTED
IDENTIFICATION BY PART NUMBER HERE
XXX

LEFT MAIN LANDING GEAR OUTER CYLINDER

LOUD P/N 1127L001-3 (LH)
1127L001-4 (RH)
1127L001-5 (LH)
1127L001-6 (RH)

NOTES

1. Refer to NA-66-1032 Sabreliner Structural Repair Manual (Chapter 32) for crack repair limits and procedures.
2. Inside diameter is: 2.938 ± 0.010 in for 1127L001-3, -4, -5, and -6 cylinders.

Figure 32-00-603 — Inspecting Main Landing Gear Strut Outer Cylinder

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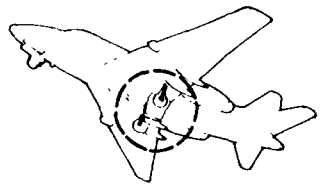

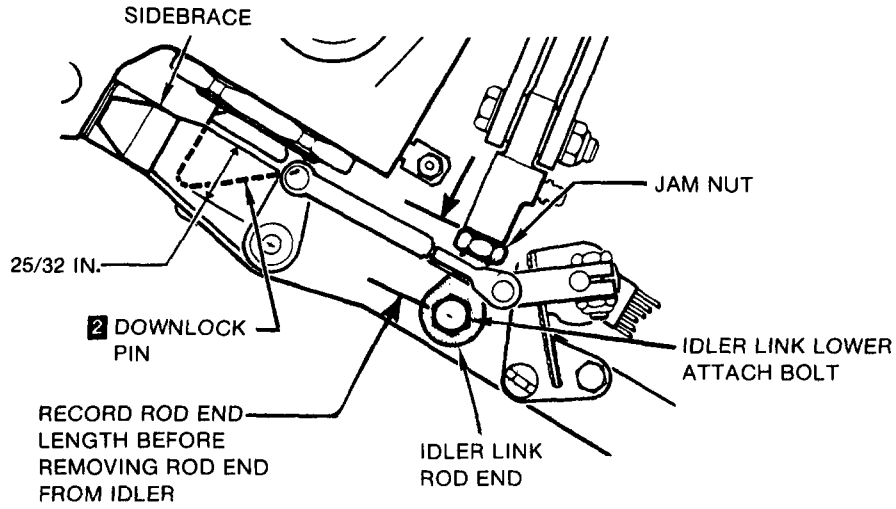

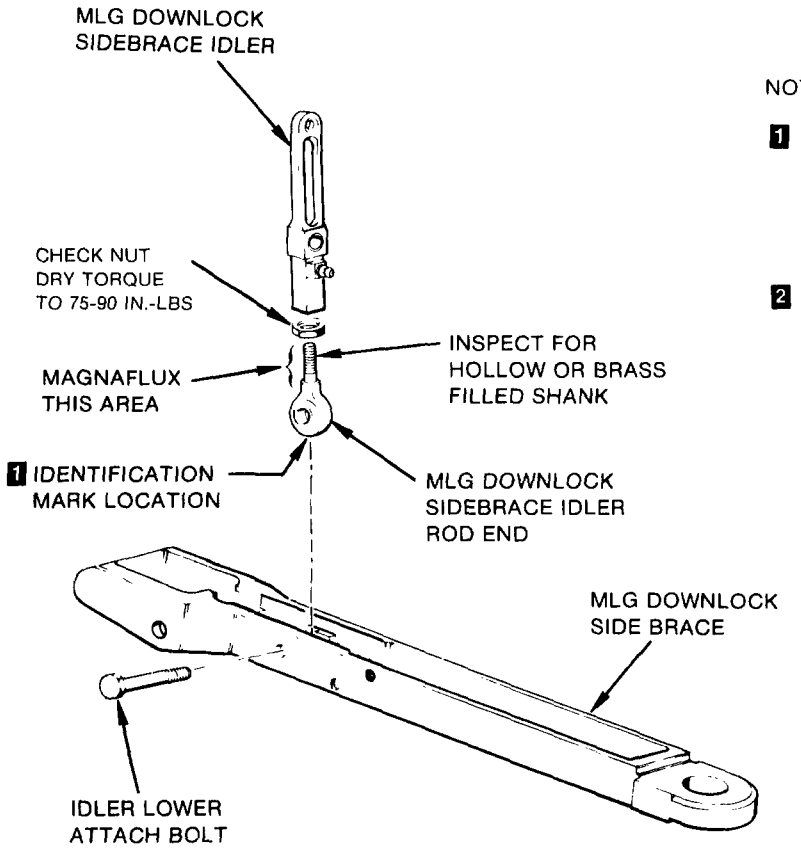
MLG DOWNLOCK SIDEBRACE IDLER
CHECK NUT DRY TORQUE TO 75-90 IN.-LBS
MAGNAFLUX THIS AREA
INSPECT FOR HOLLOW OR BRASS FILLED SHANK
1 IDENTIFICATION MARK LOCATION
MLG DOWNLOCK SIDEBRACE IDLER ROD END
MLG DOWNLOCK SIDE BRACE
IDLER LOWER ATTACH BOLT

NOTES

1 Solid shank rod ends are identified by a permanent method such as electro-chem etch or electric vibrating pencil (not impression stamp) with the numbers 81-7 on the lower outer surface.

2 Check adjustment of downlock pin.

Figure 32-00-604 — MLG Sidebrace Rod End Inspection

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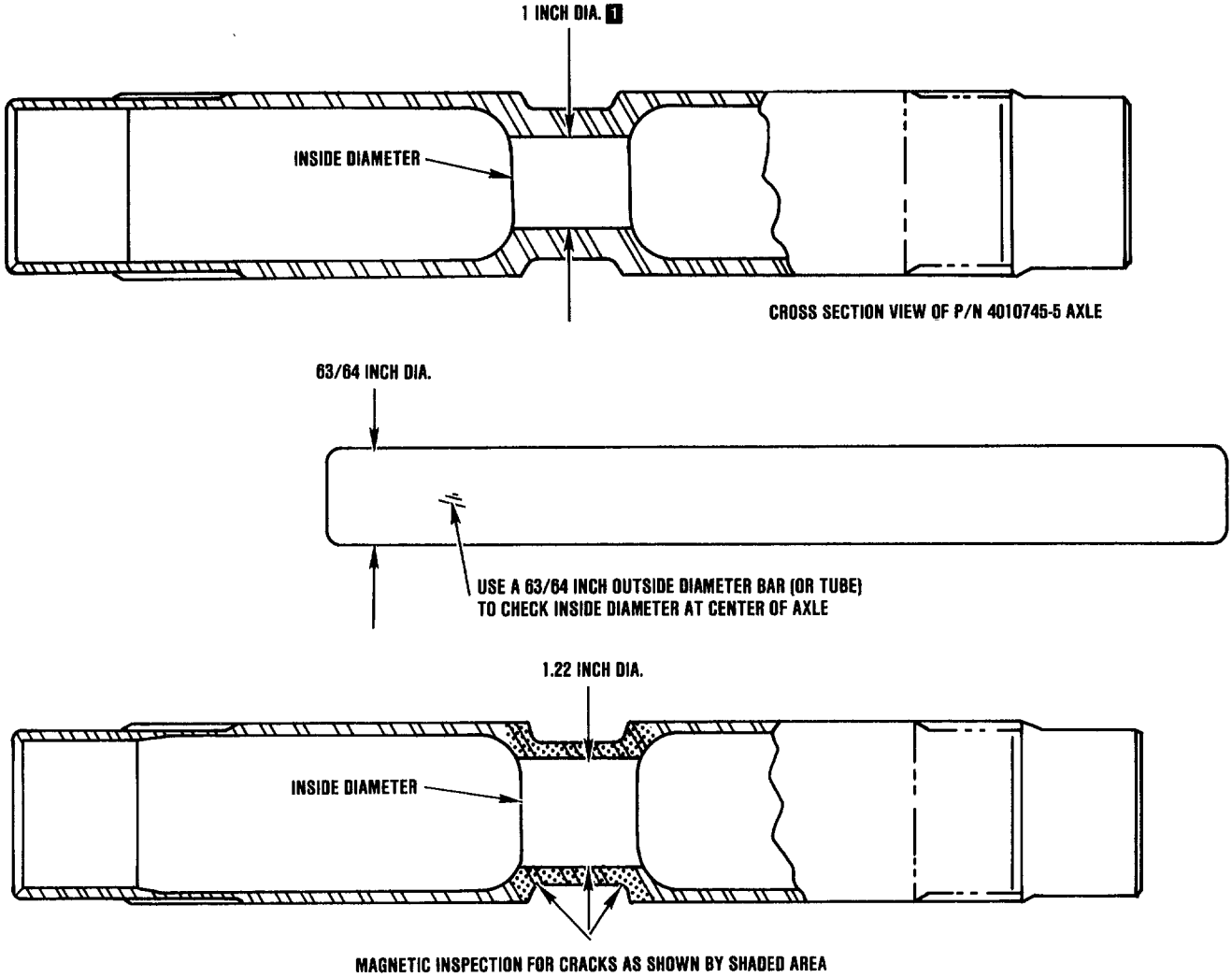
1 INCH DIA. 1
INSIDE DIAMETER
CROSS SECTION VIEW OF P/N 4010745-5 AXLE
63/64 INCH DIA.
USE A 63/64 INCH OUTSIDE DIAMETER BAR (OR TUBE)
TO CHECK INSIDE DIAMETER AT CENTER OF AXLE
1.22 INCH DIA.
INSIDE DIAMETER
MAGNETIC INSPECTION FOR CRACKS AS SHOWN BY SHADED AREA
CROSS SECTION VIEW OF P/N 4010745-3 AXLE

1 MAGNETIC INSPECTION NOT REQUIRED ON P/N 4010745-5 AXLE (HEAVY DUTY)

NEW

Figure 32-00-605 — Inspection of Nose Gear Axle

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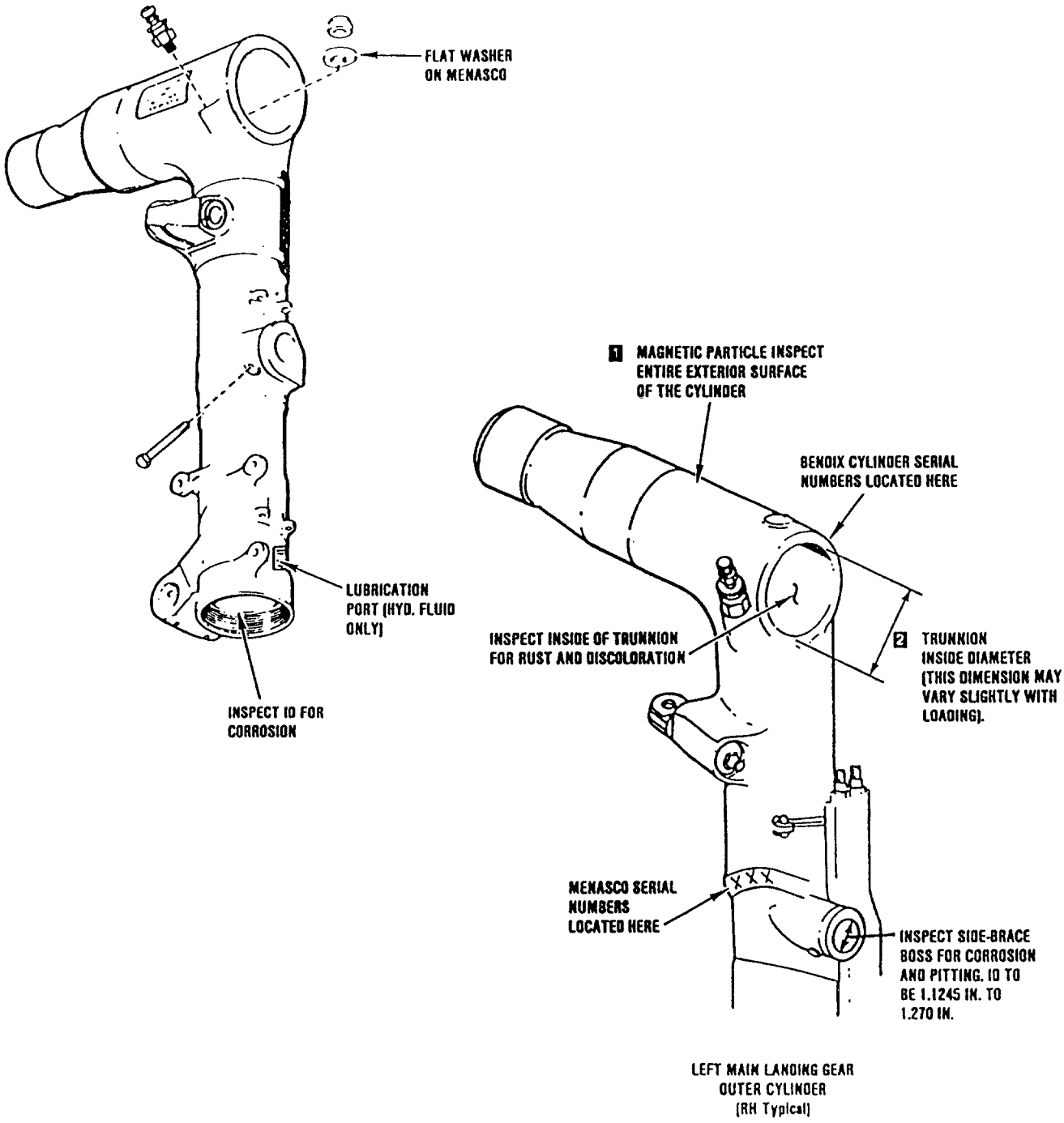
FLAT WASHER ON MENASCO
LUBRICATION PORT (HYD. FLUID ONLY)
INSPECT ID FOR CORROSION
1 MAGNETIC PARTICLE INSPECT ENTIRE EXTERIOR SURFACE OF THE CYLINDER
BENDIX CYLINDER SERIAL NUMBERS LOCATED HERE
INSPECT INSIDE OF TRUNNION FOR RUST AND DISCOLORATION
2 TRUNNION INSIDE DIAMETER (THIS DIMENSION MAY VARY SLIGHTLY WITH LOADING).
MENASCO SERIAL NUMBERS LOCATED HERE
XXX
INSPECT SIDE-BRACE BOSS FOR CORROSION AND PITTING. ID TO BE 1.1245 IN. TO 1.270 IN.
LEFT MAIN LANDING GEAR OUTER CYLINDER (RH Typical)

1 Refer to Sabreliner Structural Repair Manual, NA-66-1032 (Chapter 32) For Loud P/N 1127L001-3, -4, -5, and -6 Crack Repair Limits.

2 Inside diameter is: 3.050 ±0.010 inch for 1127L001-1 and -2 and Bendix cylinders. Inside diamter is 2.938 ±0.010 inch for 1127L001-3, -4, -5, and -6 cylinders.

Figure 32-00-606 — MLG Outer Cylinder Inspection Areas

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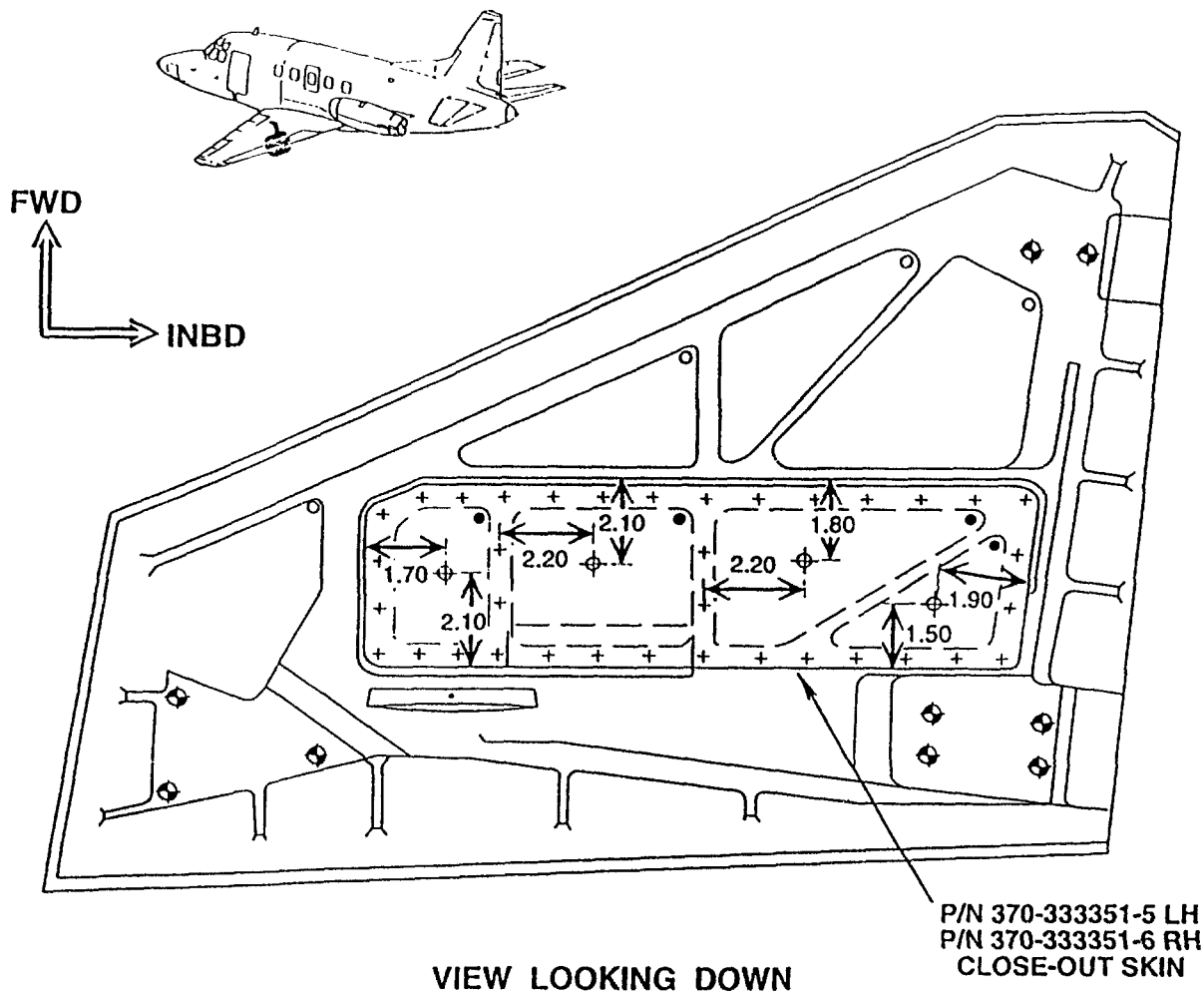

VIEW LOOKING DOWN
-70 and -80 MODELS

+ = Existing fastener location

◈ = Door hinge and roller attach holes

○ = Existing drain holes

● = Existing drain holes under close-out skin

⊕ = Added inspection holes through P/N 370-333351-5/-6 close-out skin only (4 holes at .359 dia).

Main Landing Gear Strut Door Assembly
P/N 370-333351 (LH door assy shown, RH door assy opposite)

NEW

Figure 32-00-607 — Inspection — MLG Strut Door For Corrosion
(Sheet 1 of 2) Ref: S/B 93-2

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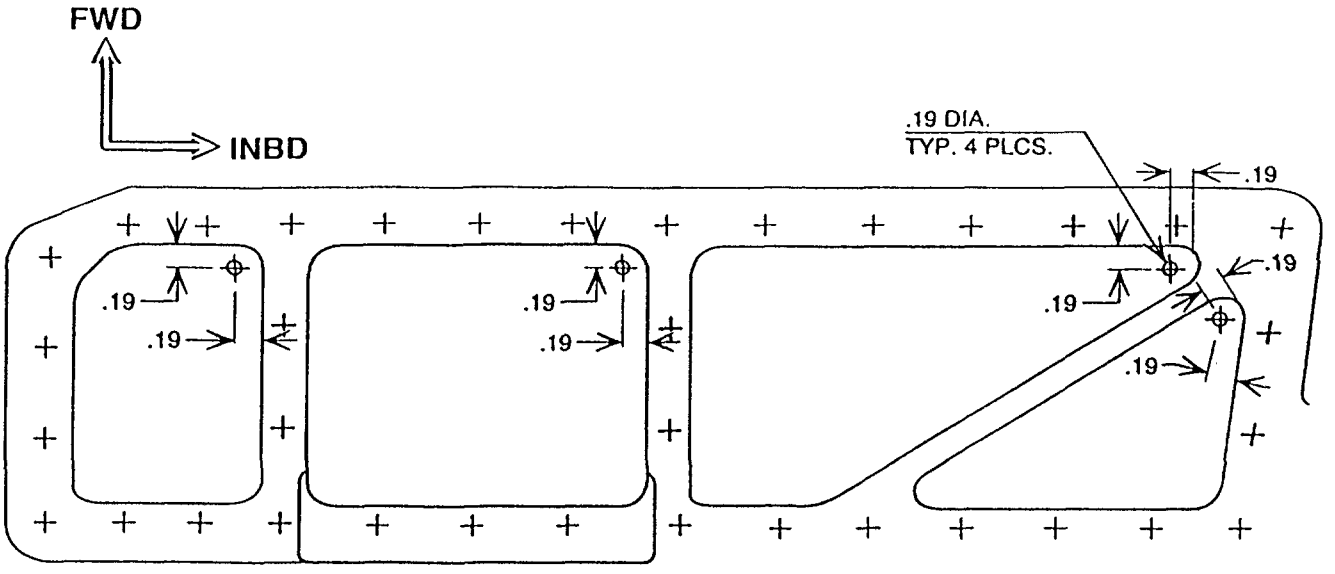

P/N 370-333351 DOOR WITH 370-333351-5/-6 CLOSE-OUT SKIN REMOVED (View Looking Down)

+ = NAS1097DD6-14 Rivets (36 each)
(optional NAS1097KE6-14)
⊕ = Added drain holes (.19 dia 4 each)

1. Remove 370-333351-5/-6 close-out skin.
2. Install drain holes as shown (4 each).
3. Prime areas with MIL-P-23377 Primer. Top coat as required.
4. Reinstall 370-333351-5/-6 close-out skin.

Drain Hole Installation (Part II)

Figure 32-00-607 — Inspection — MLG Strut Door For Corrosion
(Sheet 2 of 2) Ref: S/B 93-2

NEW

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MAIN GEAR AND DOORS

DESCRIPTION AND OPERATION

MAIN GEAR SHOCK STRUTS

The main gear shock struts are of the air-and-oil, shock-absorber type, containing a specific amount of compressed air and hydraulic fluid. The strut consists of an outer cylinder assembly at the upper end and an inner cylinder assembly (commonly referred to as the "oleo") at the lower end. The outer cylinder assembly and the inner cylinder assembly are connected by torque links at the lower end of the complete strut assembly.

The inner cylinder assembly consists of lower torque links and the axle. The cylinder consists of a piston and a metering rod, extending through the orifice in the outer cylinder assembly. A rebound snubber valve with a small orifice extending from the wall of the inner cylinder into a chamber formed between the inner cylinder assembly and outer cylinder assembly is the filler valve at the top of the outer cylinder. The upper portion of the inner cylinder is filled with hydraulic fluid. The outer cylinder contains both air and hydraulic fluid. The proportion of air to oil is determined by the extension of the shock strut.

MAIN GEAR SHOCK STRUT FAIRING DOORS

The main gear shock strut fairing doors are a pair of hinged doors mounted on the outboard side of the left and right main gear shock struts. The upper strut door is attached to a hinged fitting at the upper end of the door and is connected to the wing structure at the outboard end of the main gear side brace. The lower end of the door is hinged to the lower strut door. Mechanical linkage connecting the two doors folds the lower door upward when the gear is extended to provide ground clearance. The doors are locked in the up position by a hook and roller combination. The hook is connected to, and operated by, the main gear uplock mechanism. The main gear doors are adjusted to close flush with the fuselage and wing surface. To accomplish this, either serrated fittings, adjustable rod ends, or shims are used.

MAIN GEAR ACTUATING CYLINDERS

The main gear actuating cylinders retract and extend the right and left gear. Each cylinder is a hydraulically operated, unbalanced-type unit rated at 3000 psi. The actuating cylinder is in the main gear strut well. It is connected to the airplane structure by the cylinder end of the cylinder and to the landing gear by a clevis fitting at the piston rod end. The cylinder is extended to lower the gear and retracted to raise the gear. The actual upstop of the main landing gear occurs when the piston in the actuating cylinder bottoms out during retraction.

CAUTION

At no time should the piston in the actuating cylinder bottom out during extension of the main gear. Bottoming of the piston during main gear extension will destroy the actuating cylinder.

The piston rod can be varied in length with an adjustable rod end. End attachments are lubricated through grease fittings.

MAIN GEAR DOWNLOCK ASSEMBLY

The main gear downlock cylinder locks the main gear side brace in position when the gears are down, and withdraws the downlock pin when the gears are to be retracted. It is a hydraulically operated, balanced, floating-piston type unit rated at 3000 psi. The unit is attached to the airplane structure at the outboard end of the main gear strut well. The downlock pin and piston are not integral and are hydraulically operated. The downlock pin is spring-loaded to the downlock position. It is cammed open by the side brace when the gear is extended. The downlock pin tip is cut at an angle to permit the side brace to cam the lockpin. The pin is flat on one side, to prevent the cam angle from turning. The pin is lubricated through an external fitting.

MAIN GEAR DOWNLOCK RELIEF CHECK VALVES

The main gear downlock pressure relief check valves block unwanted low-pressure return line surges from reaching the downlock actuator. Under certain conditions, in the absence of the pressure relief check valves, such return line surges might prevent complete downlock extension or result in momentary downlock retraction and subsequent collapse.

MAIN GEAR UPLOCK ACTUATING CYLINDER

The main gear uplock actuating cylinder operates the gear uplock mechanism to the unlocked and locked positions. The cylinder is a hydraulically operated, balanced-type unit rated at 3000 psi. The uplock roller on the main gear strut trips the overcenter toggle of the uplock, allowing the lock cylinder to move to the locked position. The uplock cylinder is on the forward side of the main gear strut well, near the inboard end. The clevis end of the piston rod is connected to the uplock mechanism, while the body of the cylinder is bolted to the uplock mechanism support fitting.

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DESCRIPTION AND OPERATION (Continued)

MAIN GEAR UPLOCK MECHANISM

The main gear uplock mechanism locks and unlocks the main gear and the strut fairing door, and operates a sequence and warning system switch. The mechanism is in the inboard end of the strut well on the wing structure. The assembled unit consists of bellcranks, hooks, and links. The mechanism moves to an overcenter condition when locked or unlocked. Approximately one inch of actuating cylinder movement is required to shift the hooks to either position. The parts of the mechanism are assembled to a single casting.

MAIN GEAR RESTRICTORS

Four nonadjustable orifice-type flow restrictors are in the main gear hydraulic lines. The restrictors control the rate of landing gear retraction and extension. An internal, metal-mesh filter is used in each restrictor to protect the orifice from clogging. These restrictors are in the landing gear extension and retraction lines. The restrictors are one-way restrictors, allowing free flow in the opposite direction. The restrictors are mounted on or near the main gear actuating cylinders.

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MAIN GEAR AND DOORS

MAINTENANCE PRACTICES

MAIN GEAR SHOCK STRUT REPLACEMENT

Equipment/Tools

- Set of Jacks
- Jacking and Mooring Kit
- If Available, a Hydraulic Test Stand Capable of Delivering 4 Gpm at 3,000 Psi
- 28 VDC External Electrical Power

Main Gear Shock Strut Removal

WARNING

Deplete normal and auxiliary accumulator hydraulic pressures when through using hydraulic pressure. This will ensure that the auxiliary accumulator power assist retraction system will not operate when the gear handle is placed in the up position. This will prevent serious injury to personnel and/or damage to airplane.

NOTE

The procedure that follows is typical for both main gear assemblies.

1. Place airplane on jacks and raise airplane enough to permit gear retraction.
2. Disconnect strut door link near inbd end of side brace (see Figure 32-10-401).
3. Remove wheels and brake assemblies.

CAUTION

To prevent damage to retract cylinders, do not move landing gear by any means other than hydraulic power.

NOTE

If a hydraulic test stand is used, slowly increase hydraulic pressure to partially retract landing gear. If the airplane hydraulic pump is used, dump the acumulator pressure and cycle hydraulic pump control switch as necessary to half retract landing gear.

4. Partially retract landing gear by moving landing gear control handle up.
5. Support landing gear with chocks or suitable stops.
6. Disconnect main gear actuating cylinder rod end at main gear strut.
7. Remove disconnected landing gear supports and let gear settle back against downlock pin.
8. Remove electrical and hydraulic power from airplane.
9. Remove main gear fairing doors.

NOTE

Do not remove main gear downlock switch.

10. Remove main gear load switch and electrical harness from main gear strut.
11. Stow (tie-up) harness and main gear load switch inbd of shock strut to prevent damaging switch and harness when removing and installing strut.
12. Disconnect and cap brake hydraulic lines at flex hose connections at upper end of strut.
13. Remove brackets securing brake lines to side of strut.
14. Remove brake lines.
15. Remove side brace retainer bolt and pull side brace retainer pin. (This allows side brace to hang in a vertical position.)
16. Using a 7/8-in. socket with extension, remove shock strut retainer nut from shock strut trunnion (this nut is full fwd and inside shock strut trunnion).
17. Remove strut by sliding shock strut aft and out of trunnion support fitting.

Installing and Adjusting Main Gear Shock Strut

Equipment/Tools

- Set of Jacks
- Jacking and Mooring Kit
- If Available, a Hydraulic Test Stand Capable of Delivering 4 Gpm at 3,000 Psi
- 28 VDC External Electrical Power

Consumables

- Grease, MIL-G-81827
- Laminated Shim, Brass 1/2 Hard, MIL-B-5697, 0.063 × 2-1/4 in.

WARNING

Before applying landing gear hydraulic pressure for any reason (checks, adjustments, etc.), deplete normal and auxiliary accumulator hydraulic pressures. This will prevent the auxiliary accumulator power assist retraction system from operating and prevent serious injury to personnel and/or damage to the airplane.

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MAINTENANCE PRACTICES (Continued)

Installing and Adjusting Main Gear Shock Strut (Cont'd)

NOTES

- ***The following procedure is typical for both main gear shock strut assemblies.***
- ***All rigging must be accomplished at 1800 psi by installing the nose gear ground safety pin and maintaining 1800 psi using the airplane hydraulic pump or a hydraulic test stand.***

1. Insert Rosan fitting inside shock strut trunnion, with small end of fitting protruding from end of trunnion.
2. With Rosan fitting held firmly against end of trunnion, measure amount fitting extends beyond end of trunnion. The fitting should extend no more than 0.040 in. and no less than 0.010 in.
3. Inspect and grease all contact surfaces with grease MIL-G-81827. Inspect wheel well area for damaged lines and fittings. Check wing trunnion fitting for scratches, galling, and evidence of wear on bearing surfaces.
4. Position trunnion bearing retainer behind retainer support.
5. Lift and slide main gear shock strut into wing trunnion.
6. Slide Rosan fitting over stud in wing trunnion, with small end of fitting toward fwd end of wing trunnion.
7. Install strut retainer nut on stud. Tighten nut to 800-1000 in.-lbs.
8. Pull aft on shock strut. Check that fwd and aft end play is 0.010 to 0.040 in. Check to see that gear swings freely.

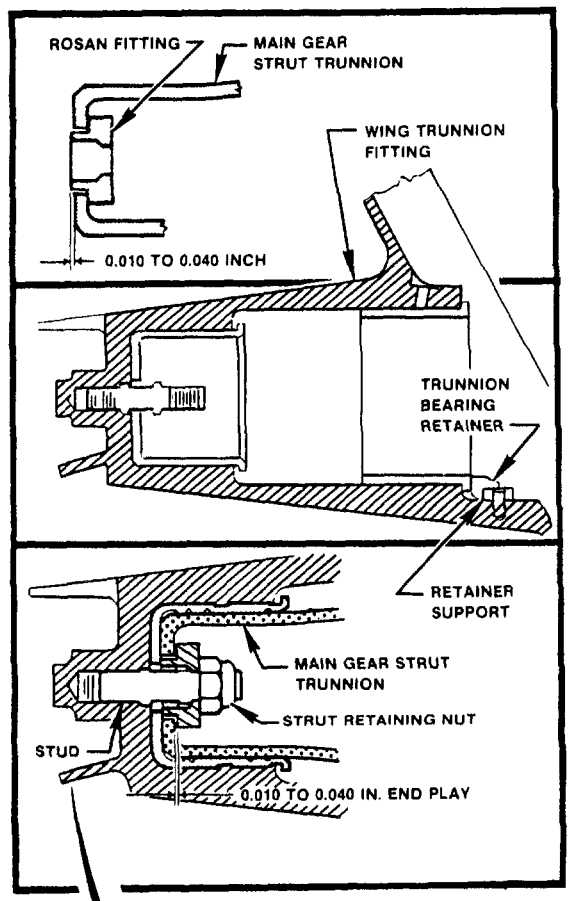
CAUTION

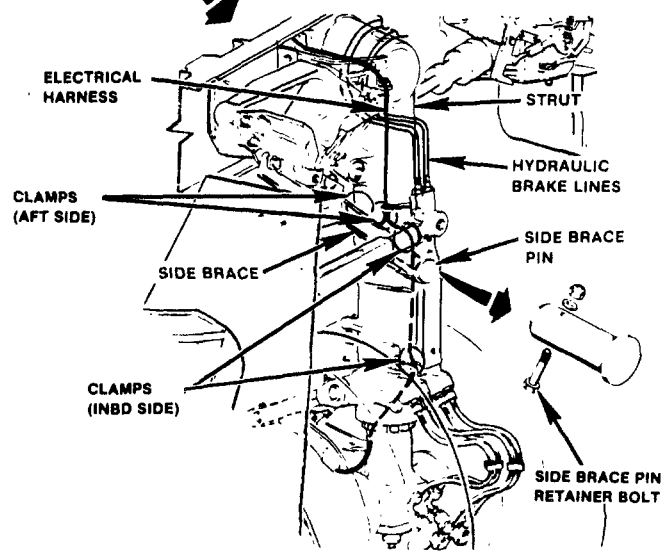
Side brace pin must be 1.125 to 1.1240 in. dia.

NOTE

Side brace bore is 1.1245 to 1.1260 in. If worn, may be reworked to 1.1265 in.

9. Align side brace with fitting on strut. Install side brace pin and pin retainer bolt. Side brace end play should be 0.002 to 0.005 in.
10. Install previously removed brake lines to side of strut.
11. Position electrical harness on side of strut and install support clamps.
12. Push main gear to down-and-locked position.





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MAINTENANCE PRACTICES (Continued)

Installing and Adjusting Main Gear Shock Strut (Cont'd)

13. Install wheel and brake assembly.
14. Uncap, connect, and tighten brake lines at flex line disconnect points.
15. Install and adjust main gear load switch.

CAUTION

To prevent possible damage to retract cylinders, do not move landing gear by any means other than hydraulic power.

16. If the actuator clevis pin has been removed from outer cylinder, refer to Replacement of Actuator Clevis Pin.

CAUTION

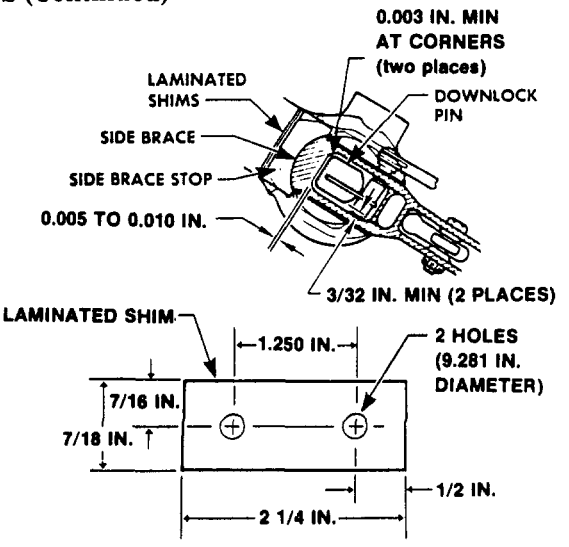
Failure to ensure a slip fit of the actuating cylinder rod end can result in fatigue cracking if the adjustment is too tight.

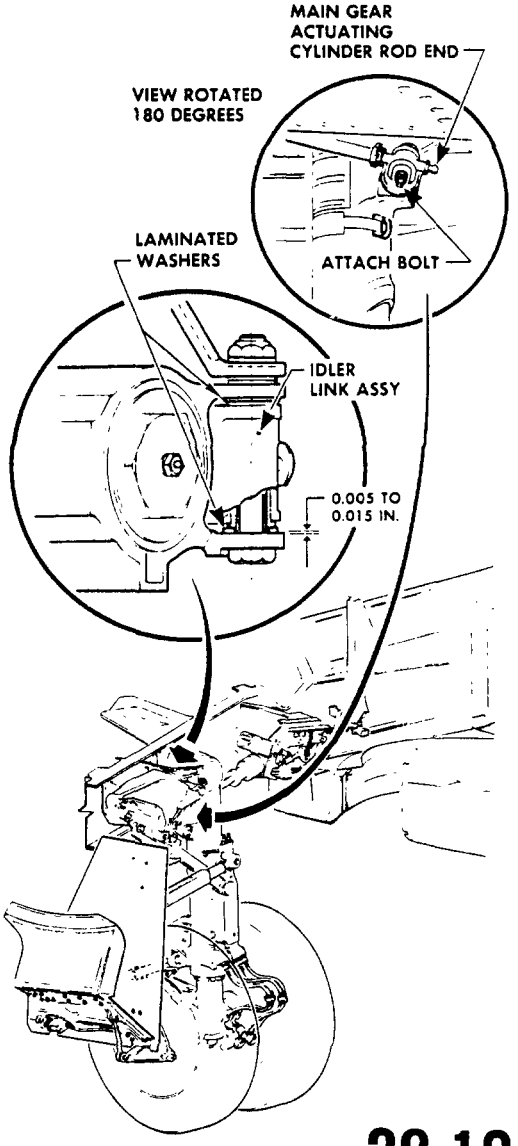
17. Fully extend main gear actuating cylinder. Adjust actuating cylinder rod end to obtain a slip fit of actuating cylinder attach bolt to strut fitting; then lengthen cylinder rod end 3/32 in. (two full turns of rod end). (This is the initial adjustment of the actuating cylinder, so the cylinder rod end need not be safetied at this time.)

NOTE

Landing gear has to be in the half retracted position against wheel chocks or other suitable stops, to remove or install bolt attaching actuator cylinder rod end to clevis pin on strut.

18. Install actuating cylinder attach bolt with head up.
19. Lubricate complete main gear shock strut.
20. Fill and inflate shock strut.
21. Run operational check-out of landing gear system.
22. With main gear down and locked, check that outbd end of side brace is centered in side brace stop and that there is a 3/32 in. min clearance on both sides of downlock pin, and that there is 0.003 in. min clearance at corners of downlock pin.





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MAINTENANCE PRACTICES (Continued)

Installing and Adjusting Main Gear Shock Strut (Cont'd)

23. If necessary, disconnect idler link assembly attach bolt at top of idler link assembly. Add, remove, or reposition 2W1LB-40-32-063 laminated washers to center side brace in side brace stop and maintain idler link total end play to 0.005-0.015 in.

NOTE

If additional shims are needed, they can be locally fabricated. Make shims from laminated shim stock.

24. With side brace firmly against side brace stop, check that there is 0.005 to 0.010 in. clearance between outbd side of downlock pin and side brace. If necessary, partially retract landing gear and support it in partially retracted position. Add or remove laminated shims on outbd side of side brace stop to obtain specified clearance when gear is down and locked.

25. With gear down and locked, check that downlock pin extends into side brace 25/32 in. (The tool illustrated on page 411 may be used as an aid in measuring downlock pin extension.) If necessary, partially retract and support landing gear. Remove idler link assembly lower attach bolt. Loosen idler link rod end jam nut and adjust length of link so that downlock pin is extended 25/32 in. into side brace when gear is down and locked.

26. Install and tighten idler link attach bolt.

27. Check adjustment of downlock switch.

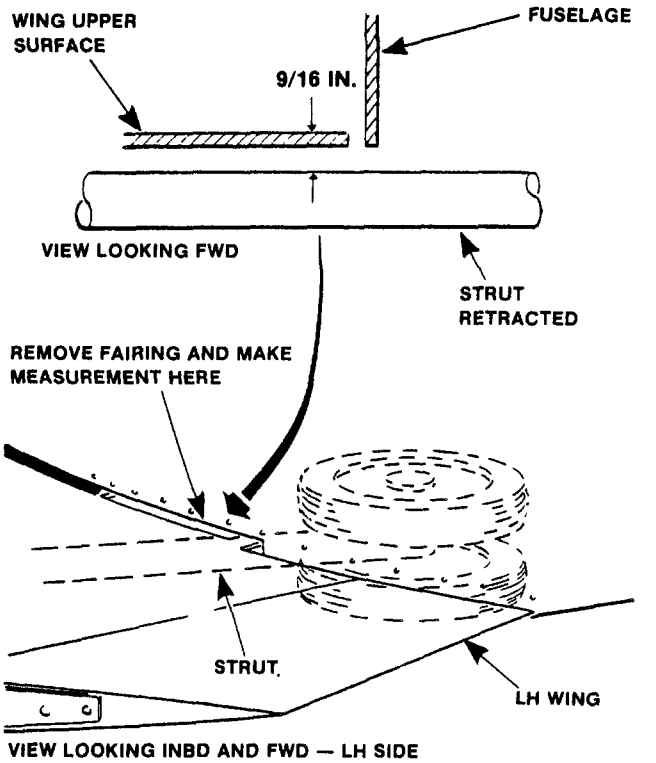
28. Loosen lower jam nuts on up-stop bumper by several turns of nut, then tighten upper jam nut on bumper to raise bumber away from strut when it is retracted.

29. Install nose gear maintenance safety lock pin. (See note.) Move landing gear control handle up. Slowly increase hydraulic pressure (not to exceed 1800 psi) and retract gear to up-and locked position.

NOTE

With nose gear maintenance safety lock pin installed (prevents nose gear retraction) hydraulic pressure will remain on the main gears actuating cylinders when up and locked. This will prevent uplock rollers from hanging on uplock hooks when checking clearance in steps 25 through 34.

30. Check clearance between gear uplock roller and throat of uplock hook. Clearance should be 1/16 in. To obtain specified clearance, add or remove shims between uplock roller support fitting and gear strut to obtain specified clearance when gear is up and locked (see Figure 32-10-401).



VIEW LOOKING INBD AND FWD — LH SIDE

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MAINTENANCE PRACTICES (Continued)

Installing and Adjusting Main Gear Shock Strut (Cont'd)

31. With gear up and locked and hydraulic pressure at 1800 psi, check clearance between top of main gear outer cylinder and wing panel skin. Clearance should be 9/16 in. (In order to make this measurement, it is necessary to remove a section of fairing from the top of the wing where it joins the fuselage.) Shorten (or lengthen) main gear actuating cylinder rod end to obtain noted dimension when gear is retracted and actuating cylinder has 1800 psi pressure applied. The rod end adjustment may be made with the gear remaining up and locked. Dump hydraulic pressure allowing gear to settle on uplock hook; then rotate cylinder piston rod, using a wrench on wrench flats at outbd end of piston rod.

32. With main gear up and locked, and actuating cylinder pressurized at 1800 psi, check clearances between bottom of gear uplock roller and uplock hook (0.005 to 0.035 in.), and between inbd and outbd of gear uplock roller and uplock hook (1/16 in.). To obtain the specified clearances, adjust gear uplock roller support fitting up or down on serrations, or inbd and outbd adjustment by laminated shim. Tighten roller support fitting attach bolts, and safetywire bolts.

33. With gear up and locked, check that up-stop is centered on landing gear strut outer cylinder. If up-stop is not centered, loosen up-stop jam nuts and reposition stop in slot in stop support bracket.

34. Loosen upper jam nut on up-stop and allow bumper to rest on top of strut. Tighten lower jam nut on up-stop support bracket. Tighten nut one more full turn (compressing rubber bumper approximately 3/64 in.), then tighten upper jam nut to lock adjustment. Safety jam nuts to each other with safetywire.

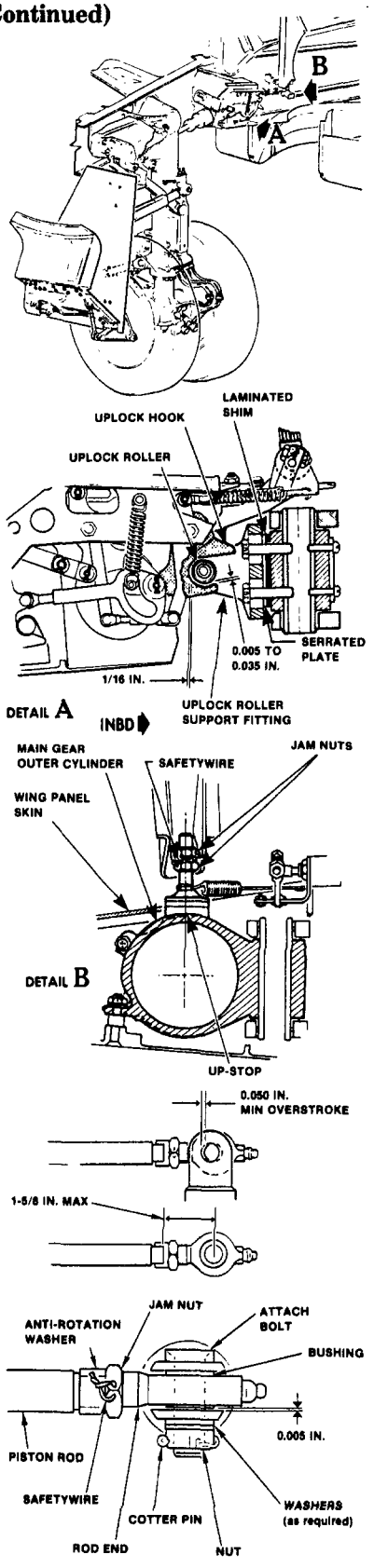
35. With landing gear down and locked and hydraulic pressure shut off, disconnect landing gear actuating cylinder from strut. Do not change actuating cylinder rod-end adjustment. Apply hydraulic pressure to fully extend actuating cylinder, and check that there is at least 0.050 in. of actuator overstroke available. Check also that 1-5/8 in. dimension is not exceeded.

NOTE

Washers under nut may be removed if necessary to obtain 0.005 in. clearance.

36. Turn off hydraulic pressure and reconnect actuating cylinder to strut. Install attach bolt, head up, and tighten nut as necessary to maintain a clearance of 0.005 in. min between actuator rod-end bushing and clevis. Max torque is 100 in.-lbs. Safety nut with cotter pin.

37. Tighten actuating cylinder rod-end jam nut, and safety to anti-rotation washer with safetywire. Remove nose gear maintenance safety lockpin.



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MAINTENANCE PRACTICES (Continued)

Installing and Adjusting Main Gear Shock Strut (Cont'd)

38. Check adjustment of uplock switch.
39. Extend landing gear. Connect strut door link to side brace clevis. Retract gear and check adjustment of strut door.
40. Run complete check of landing gear. Check indicating and warning system, sequencing, and emergency operation.
41. Extend landing gear. Check landing gear installation for installation of all safeties.
42. Lower airplane and remove jacks. Install and adjust static ground so that static ground is dragging about one inch on ground. Tighten static ground jam nut.
43. Bleed wheel brake system.
44. Check anti-skid system for proper operation.
45. Remove electrical and hydraulic power. Install all access doors.

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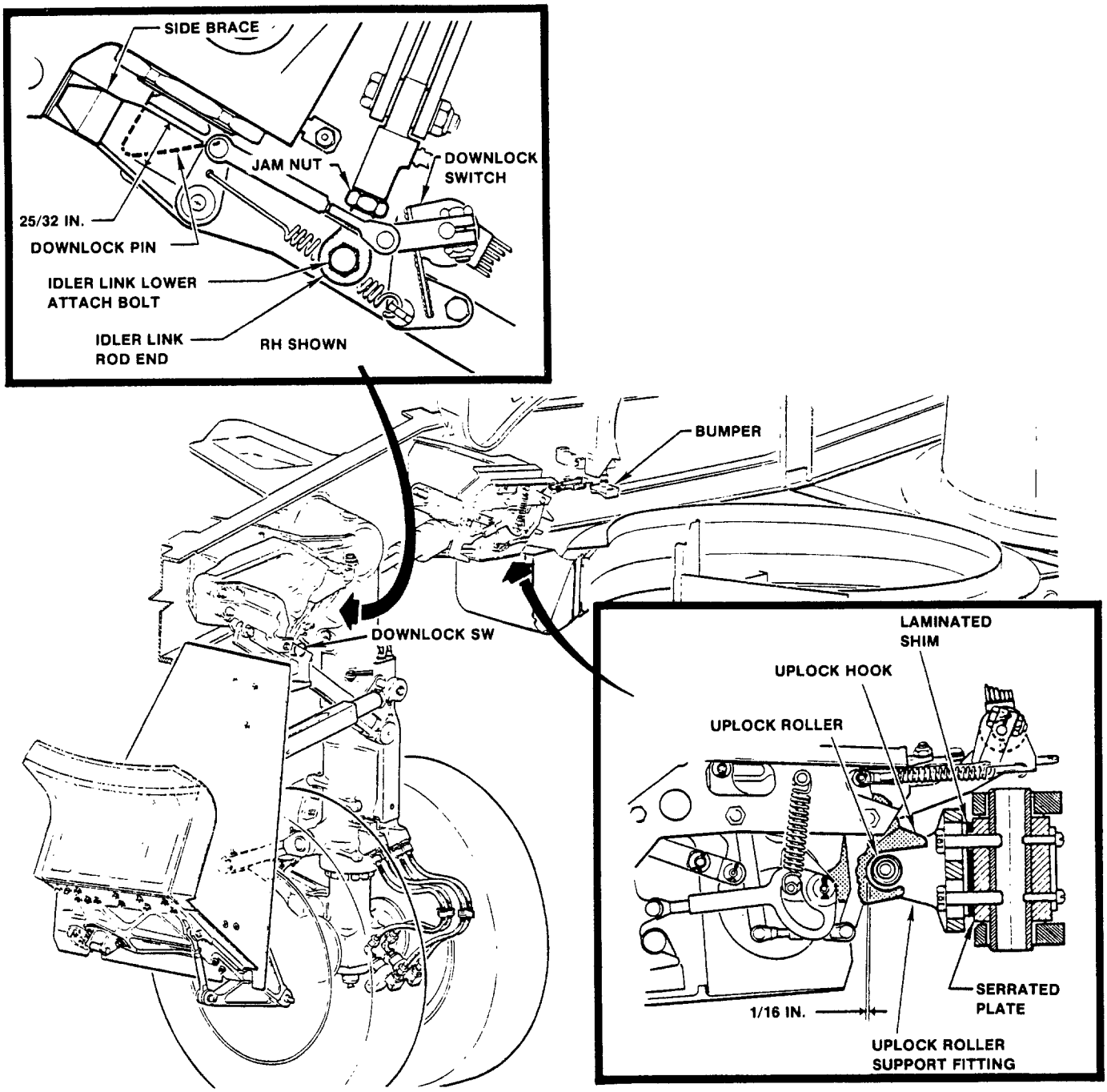

Figure 32-10-401 — Installing and Adjusting Main Gear Shock Strut

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MAINTENANCE PRACTICES (Continued)

MAIN GEAR ACTUATING CYLINDERS REPLACEMENT

Removal:

WARNING

Deplete normal and auxiliary accumulator hydraulic pressures when through using hydraulic pressure. This will ensure that the auxiliary accumulator power assist retraction system will not operate, when the gear handle is placed in the up position. This will prevent serious injury to personnel and/or damage to the airplane.

CAUTION

To prevent possible damage to main gear actuating cylinders, do not move landing gear by any means other than hydraulic power.

NOTE

Landing gear has to be in the half retracted position against wheel chocks or other suitable stops to remove or install bolt attaching actuator cylinder rod end to clevis pin on strut.

1. Place jacks under airplane and raise airplane enough to permit gear retraction.
2. Disconnect actuating cylinder rod end at strut attachment point.
3. Disconnect and cap actuating cylinder hydraulic lines.
4. Disconnect inbd end of actuating cylinder and remove actuating cylinder.

Installation:

WARNING

Deplete normal and auxiliary accumulator hydraulic pressure when through using hydraulic pressure. This will ensure that the auxiliary accumulator power assist retraction system will not operate, when the gear handle is placed in the up position. This will prevent serious injury to personnel and/or damage to the airplane.

CAUTION

To prevent damage to actuating cylinder attachment structure, do not tighten bolt and nut against structure.

NOTE

The following procedure is typical for both LH and RH main gear actuating cylinder.

1. Position actuating cylinder into inbd attachment structure and install bolt from the fwd end (aligning bolt head locking flat in casting recess).
2. Install washer and nut.
3. Rotate nut until clearance of 0.015 to 0.030 inch is obtained between washer and aft face of attachment casting when washer is held against nut.
4. Safety nut with cotter pin (see Figure 32-10-402).
5. Manually place landing gear to the downlock position (actuating cylinder not attached).
6. Uncap, connect, and tighten actuating cylinder hydraulic lines.

CAUTION

When installing a new actuator, and prior to connecting actuator to strut (connected in strut well only), position actuator so that rod end will clear the adjacent structure. Hydraulically extend and retract (slowly) actuator several cycles to eliminate any trapped air. To prevent damage to the system, use a low hydraulic pressure during cycling.

7. Slowly extend and retract actuator several cycles hydraulically to eliminate any trapped air. Use minimum hydraulic pressure during cycling to prevent damage to components.
8. Remove blocks from under landing gear and allow gear to fully extend and lock.
9. With 1800 psi hydraulic pressure, fully extend actuating cylinder hydraulically very carefully. Adjust actuating cylinder rod end to obtain a slip-fit of attach bolt when actuating cylinder is connected to strut fitting. Then, lengthen cylinder rod end to 3/32 inch (two full turns of cylinder rod end). **This is a preliminary adjustment only. See step 14 for final adjustment.**

NOTES

- ***This is the initial adjustment of the actuating cylinder. The cylinder rod end is not to be safetied at this time.***
- ***Upstop bumper must be raised to prevent damage to upstop when step 14 is accomplished.***

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MAINTENANCE/CHECK (Continued)

INSTALLATION (Cont'd)

10. Loosen lower jam nuts on upstop bumper by several turns of the nut. Then, tighten upper jam nut on bumper to raise bumper away from strut when it is retracted.

WARNING

Before applying landing gear hydraulic pressure for checks, adjustments, etc., deplete the normal and auxiliary accumulator pressure. This will prevent the auxiliary accumulator power assist retraction system from operating and prevent serious injury to personnel and/or damage to the airplane.

11. Install nose gear ground safety pin. This prevents nose gear from retracting while allowing 1800 psi to be maintained on the airplane for subsequent procedures.
12. Remove the uplock roller.

NOTES

- ***While making rod end adjustment, verify that shaft is not bent by ensuring that clearance changes consistently with no gross increases or decreases. Half-turn movement should result in small, incremental variances in clearance. Large incremental variances indicate a bent retract cylinder piston or shaft.***
- ***If rod end adjustment cannot be made (rod end bottoms in the cylinder piston rod), remove cylinder attach bolt, fully extend cylinder and check for 21.851 inches from center of inboard attach bolt to center of cylinder piston rod end. See figure 32-10-411.***

13. Move the LDG GEAR handle UP. Slowly increase hydraulic pressure up to (and maintain) 1800 psi and retract gear.
14. With gear up and locked and 1800 psi hydraulic pressure, check clearance between the top of main gear outer cylinder and wing panel skin. Clearance should be 9/16 inch unless 1127L001-5 or -6 upper cylinder is installed, then clearance is 1/2 inch. (In order to make this measurement, it is necessary to remove a section of the fairing from the top of the wing where it joins the fuselage.) Shorten (or lengthen) main gear actuating cylinder rod end to obtain noted dimension when gear is retracted and actuating cylinder has 1800 psi pressure applied. (Rod end adjustment may be made with gear down and locked. Dump hydraulic pressure and rotate cylinder piston rod using a wrench on wrench flats at outboard end of piston rod.)
15. With hydraulic pressure off, disconnect landing gear actuating cylinder from strut. Position landing gear down and locked. Do not change actuating cylinder rod end adjustment. Apply hydraulic pressure to fully extend actuating cylinder, and check that there is a least 0.050 inch of actuator overstroke available. Check also that 1-5/8-inch dimension is not exceeded. See 32-10-01, page 405.

NOTE

Landing gear has to be in the half-retracted position against the wheel chocks or other suitable stops to remove or install bolt attaching actuator cylinder rod end to clevis pin on strut.

16. Turn off hydraulic pressure and reconnect actuating cylinder to strut. Tighten attach bolt nut as necessary to maintain a clearance of 0.005 inch minimum between actuator rod end bushing and clevis. Maximum torque is 100 inch-pounds. Safety nut with cotter pin.

NOTES

- ***If actuator clevis attach bolt was inserted upside down for rigging, reinstall head-up.***
- ***Washers under nut may be removed if necessary to obtain 0.005-inch clearance.***

17. Tighten actuating cylinder rod end jam nut and safety to the anti-rotation washer with lockwire.
18. Check clearance between gear and uplock roller and throat of uplock hook. Clearance should be 1/16 inch. To obtain specified clearance, add or remove shims between uplock roller support fitting and gear strut to obtain specified clearance when gear is up and locked.
19. With main gear up and locked and actuating cylinder pressurized to 1800 psi, check clearance between the bottom of gear uplock roller and uplock hook. Clearance must be 0.005 to 0.035 inch. Adjust gear uplock roller support bracket up or down on serrations to obtain specified clearance when gear is up and locked and hydraulic pressure on actuating cylinder is 1800 psi. Tighten roller support bracket attach bolts, and safety bolts with lockwire.

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MAINTENANCE PRACTICES (Continued)

Installation (Cont'd)

20. Loosen upper jam nut on upstop and allow bumper to rest, centered on top of strut. Tighten lower jam nut on upstop until nut surface contacts the lower surface of upstop support bracket. Tighten nut one more full turn (compressing the rubber bumper approximately 3/64 inch) while maintaining 1800 psi. Then, tighten upper jam nut to lock the adjustment.
21. Check the adjustment of uplock switch. Refer to Figure 32-60-403.
22. Extend landing gear and safety jam nuts to each other and to upstop bumper with lockwire.
23. Remove nose gear ground safety pin.
24. Retract landing gear and shut off hydraulic pressure. Check that hydraulic lines on actuating cylinder clear all the adjacent structures and all hydraulic lines.
25. Extend gear and check actuating cylinder for evidence of hydraulic leaks.
26. Run complete check of the landing gear. Check indicating and warning system, sequencing, and emergency operation.
27. With landing gear extended, check landing gear for installation of all safeties.
28. Lower airplane and remove jacks.

MAIN GEAR DOWNLOCK ASSEMBLY REPLACEMENT

Removal:

NOTE

The following procedure is typical for both LH and RH main gear downlock assemblies.

1. Remove wing access panel above downlock assembly.
2. Place airplane on jacks; apply hydraulic pressure and partially retract and support landing gear.
3. Discharge hydraulic pressure and disconnect electrical power.
4. Place landing gear control handle down.
5. Disconnect and cap downlock assembly hydraulic lines.
6. Mark position and remove downlock assembly hydraulic fittings.
7. Cut safetywire and remove downlock assembly retainer nut. Lift downlock assembly from casting.

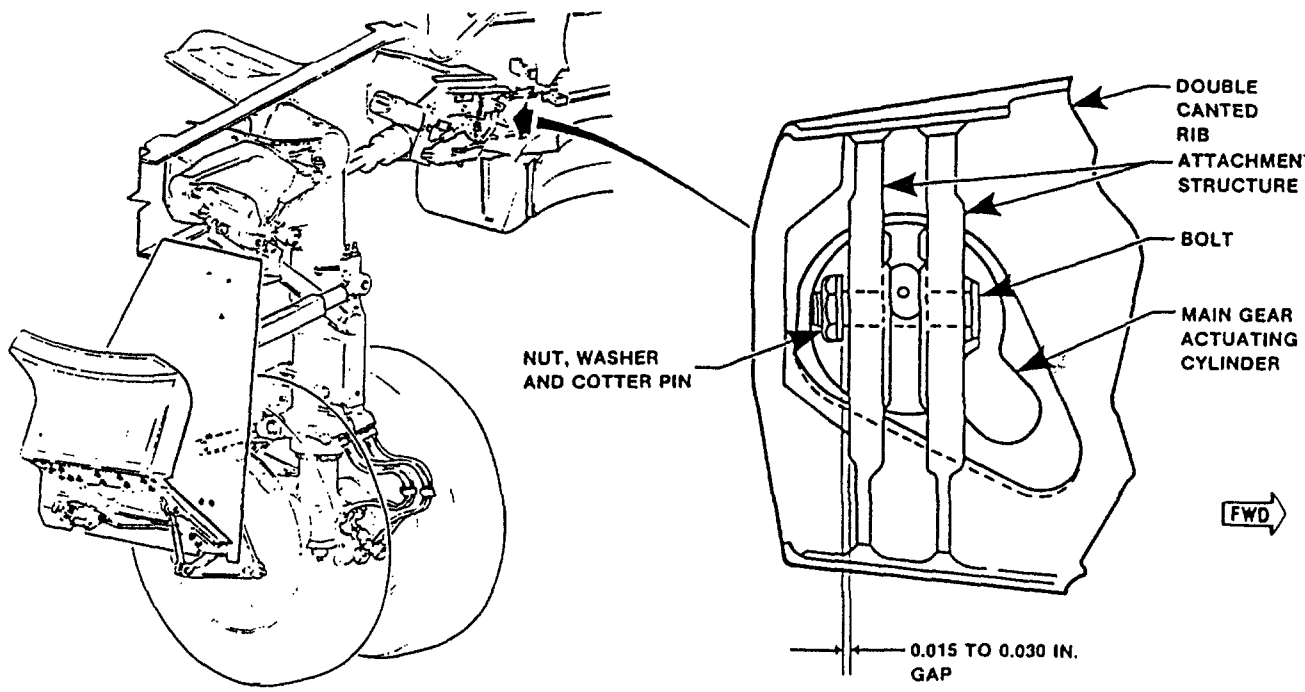

Figure 32-10-402 — Main Gear Actuating Cylinder Inboard Attachment

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MAINTENANCE PRACTICES (Continued)

Installation:

Equipment/Tools

- Set of Jacks
- Jacking and Mooring Kits

WARNING

Before applying landing gear hydraulic pressure for any reason (checks, adjustments, etc.), deplete normal and auxiliary accumulator pressures. This will prevent the auxiliary accumulator power assist retraction system from operating and prevent serious injury to personnel and/or damage to the airplane.

NOTE

The following procedure is typical for both LH and RH main gear downlock assemblies.

1. Position downlock assembly in downlock assembly casting with flat edge of downlock pin toward outbd end of drag brace.
2. Install new O-rings and packings on hydraulic fittings. Install and tighten fittings in downlock casting.
3. Uncap, connect, and tighten downlock assembly hydraulic lines (torque "B" nuts to 135-185 in.-lbs.).
4. Install and torque downlock assembly retainer nut to 200-300 in.-lbs. Safetywire nut.
5. Install wing skin panel.
6. Check adjustments and necessary clearances around downlock pin (see Installing and Adjusting Main Gear Shock Strut illustration for clearances).
7. Check downlock assembly hydraulic lines for evidence of hydraulic leaks.
8. Check operation of landing gear position indicating and warning system. Also check normal and emergency operation of landing gear.

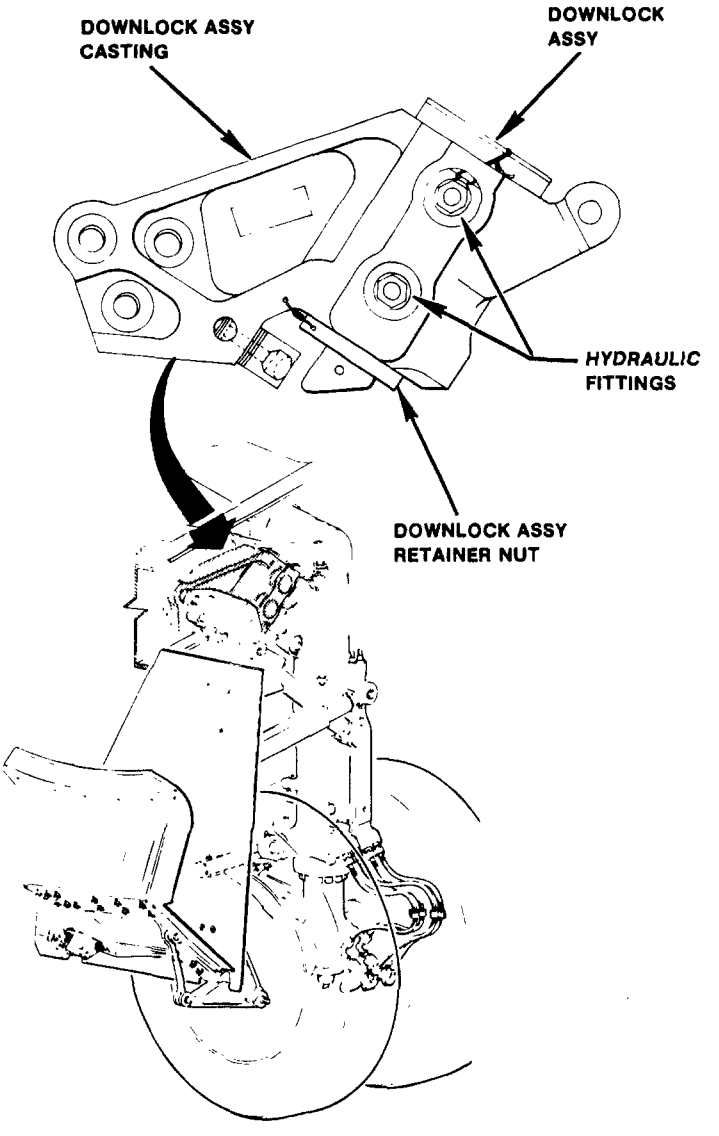
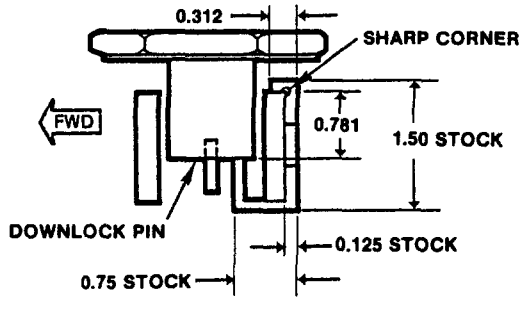


Figure 32-10-403 — Installing and Adjusting Main Gear Downlock Assembly

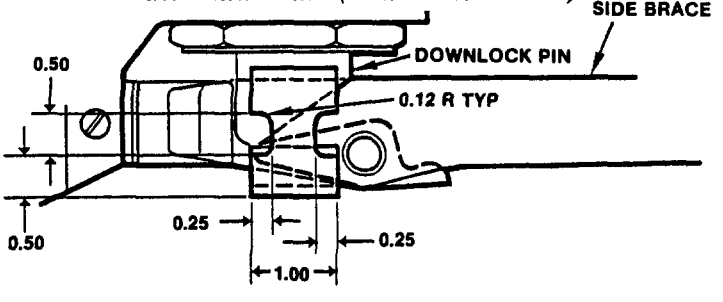
This tool may be manufactured locally and used as an aid to check main gear downlock pin extension. The tool should be installed on aft of side brace as shown. If downlock pin is not extended far enough into side brace, a gap will exist between tool and downlock pin. If downlock pin is extended too far into side brace, downlock pin will interfere with installation of tool. Adjustment of side brace idler link may be necessary.



(LOOKING INBD)

NOTES

- ***All dimensions are in inches.***
- ***Manufacture from 6061-T5 rectangular aluminum tube (0.125 × 0.75 × 1.50).***



(LEFT-SIDE BRACE — LOOKING FWD)

MAIN GEAR DOWNLOCK PIN EXTENSION CHECK TOOL

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MAINTENANCE PRACTICES (Continued)

Adjusting Main Gear Uplock Overcenter Stop

WARNING

Before applying landing gear hydraulic pressure for any reason (checks, adjustments, etc.), deplete normal and auxiliary accumulator hydraulic pressures. This will prevent the auxiliary accumulator power assist retraction system from operating and prevent serious injury to personnel and/or damage to the airplane.

NOTES

- ***The overcenter stop provides a means of positioning the main gear uplock hook and the strut door hook in an overcenter position with the hooks in either the locked or unlocked position.***
- ***The following procedure is typical for both LH and RH main gear uplock assemblies.***

1. Manually position main gear uplock hook to locked position. Check that toggle assembly is against overcenter stop.
2. Lay a straightedge across pivot points shown. Check that center of toggle assembly pivot is 3/32 (+1/32, -0) inch above straightedge. (An alternate method of checking this dimension is shown on sheet 2 of 2).
3. If gear uplock hook is not overcenter as specified, rotate uplock hook to unlock position.
4. Add or remove washers under head of overcenter stop; then repeat steps 1 and 2.
5. Retract landing gear and check normal and emergency extension.

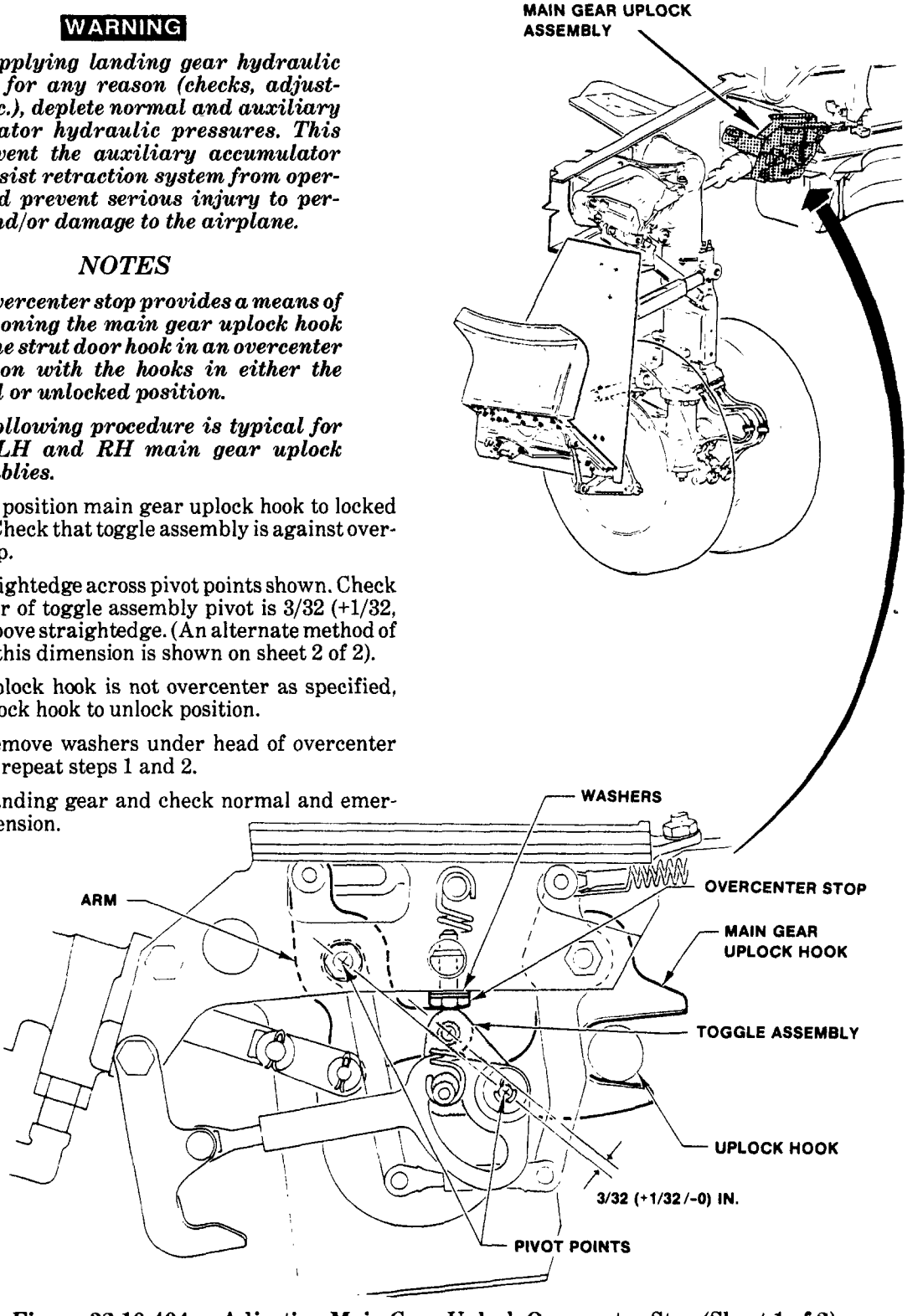

Figure 32-10-404 — Adjusting Main Gear Uplock Overcenter Stop (Sheet 1 of 2)

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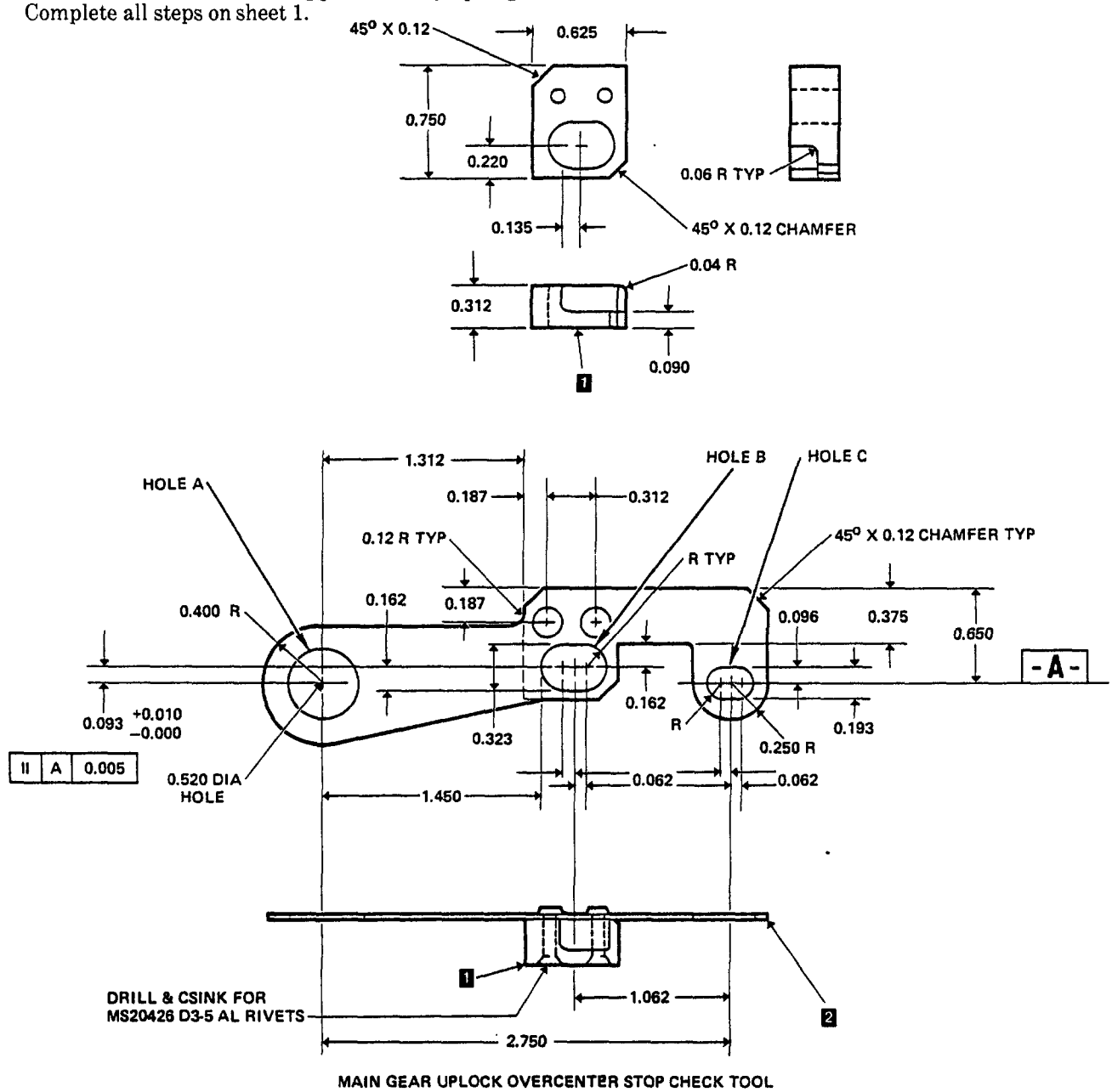
MAINTENANCE PRACTICES (Continued)

Adjusting Main Gear Uplock Overcenter Stop (Alternate Method)

1. Remove uplock toggle assembly spring.
2. Hold toggle assembly against overcenter stop (hooks in locked position) and place tool on uplock assembly with holes A and C over pivot points shown on sheet 1. (Tool hole B should fit over toggle assembly pivot point shown on sheet 1 without being forced.)
3. Add or remove washers under head of overcenter stop to move toggle assembly pivot up or down to align with tool hole B.
4. Remove tool and connect toggle assembly spring. Complete all steps on sheet 1.

NOTES

- *All dimensions are in inches.*
- *Break all sharp edges.*
- *Manufacture from:*
  - 1 *2024-T-351 aluminum bar or plate (0.312 × 0.625 × 0.750).*
  - 2 *17-7PH CRES sheet AMS 55929 (0.020 × 1.50 × 4.00). Heat treat in accordance with AMS 5529.*



MAIN GEAR UPLOCK OVERCENTER STOP CHECK TOOL

Figure 32-10-404 — Adjusting Main Gear Uplock Overcenter Stop (Sheet 2 of 2)

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MAINTENANCE PRACTICES (Continued)

Torquing Main Gear Downlock Pin Retaining Bolt

1. Remove access panel from wing, above downlock assembly.
2. Torque retaining bolt to 190 to 210 in.-lbs and install safetywire.
3. Install access panel.

MAIN GEAR STRUT DOORS REPLACEMENT

Equipment/Tools

- Set of Jacks
- Jacking and Mooring Kit
- 28 VDC External Electrical Power
- Push-pull Scale 0 to 150 Lbs

Removal:

1. Disconnect strut door actuating link from strut door.
2. Remove attaching parts that secure strut door to its hinge at three locations and remove strut door.
3. If necessary, remove strut door hinge support and/or strut door hinge per steps A thru H.
   A. Remove screws that retain inbd trailing edge wing structure.
   B. Remove retaining pins that attach inbd trailing edge wing structure to fuselage.
   C. Remove inbd trailing edge wing structure from wing.
   D. On left wing, remove flap position transmitter.
   E. Remove two bolts, washers, and nuts that attach strut door hinge to hinge support.
   F. Remove hinge.
   G. Remove the three bolts that secure hinge support to main landing gear downlock support assembly.
   H. Remove hinge support.

Installation:

NOTES

- ***When using the airplane hydraulic pump to cycle the landing gear, ensure that the hydraulic reservoir is properly serviced (hydraulic fluid and air pressure), and that sufficient cooling time for the hydraulic pump motor is allowed between landing gear retraction cycles.***
- ***The following procedure is typical for both LH and RH main landing gear strut doors.***
- ***The landing gear may be cycled slowly (after all accumulator pressures are dumped) when checking for door and hook clearances by utilizing the hydraulic pump motor control switch. This will allow slight movements of the landing gear during the retraction cycle.***

1. Place jack under airplane and raise airplane to permit landing gear retraction.
2. To gain access to upper strut door hinge support, remove both inbd wing trailing edge panels.
3. Install upper strut door hinge support to main landing gear downlock support assembly using three bolts. Torque bolts to 800-1000 in.-lbs.
4. Install upper strut door hinge fitting on upper strut door.
5. Install one washer between hinge and door at the fwd mounting bolt.
6. Tighten nuts sufficiently to hold strut door and still allow a minor fore and aft movement at the bottom of the strut door.
7. Install upper strut door in upper strut door hinge support.
8. Install bolts, washers, and nuts in hinge fitting. Tighten nuts until an 0.005 to 0.010 in. end play of each attach bolt is obtained.
9. Install wing trailing edge panels removed in step 2.
10. Install lower strut door to upper strut door hinge.

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MAINTENANCE PRACTICES (Continued)

Installation (Cont'd)

11. Install bolts, washer, and nuts. Tighten nuts sufficiently to remove end play from bolts.
12. Check that lower strut door does not bind when moved.
13. Remove door roller from bracket and install lower strut door lock roller bracket.
14. Center bracket in mounting holes and install bolts, washer, and nuts. (See Figure 32-10-406.)
15. Tighten nuts sufficiently to hold roller brackets and still allow movement of roller bracket.
16. Retract landing gear chassis to up-and-locked position.
17. Remove hydraulic and electrical power.
18. Dump accumulator hydraulic pressure.
19. Manually push upper and lower strut fairing door to closed position.

NOTES

- ***If strut door cannot be moved sufficiently to center hook in roller bracket, the roller bracket may be positioned as necessary.***
- ***A minor fore and aft movement is provided by oversized holes in the upper strut door serrated washer bushings.***

20. Position strut doors fwd or aft as necessary to center the door uplock hook in the door uplock roller bracket.
21. While holding strut doors as in step 19, mark upper and lower strut doors where trim is needed to maintain 1/32- to 5/32-in. clearance between edge of doors and edge of wing skins and between the upper and lower doors.
22. Carefully lower doors and torque the three upper strut door hinge attach bolts. Torque nuts to 275 in.-lbs.
23. Manually close upper and lower strut doors. Check that door uplock hook is centered in lower strut door uplock roller bracket. If hook is not centered, repeat steps 19 thru 22.
24. Lower doors and trim as necessary.

NOTE

If adjustment is necessary in step 25 (shim or washer added), the upper strut door hinge can be marked in relation to the upper strut door before the attaching bolts are removed. This will ensure that proper alignment of the door lock roller bracket to door hook is maintained.

25. Manually close upper and lower strut doors, and check that upper strut fairing door is faired to the wing skin mold line. If door is not faired, add or remove shims (370-333399-3) as required between the upper strut door hinge fitting and upper strut door. Additional adjustment to a max of 0.036-in washer thickness can be obtained by adding washers (LD153-0011-0015) between hinge and shims on the two aft hinge mounting bolts and by adding washers (LH153-0011-0016 and LD153-0011-0015) between the hinge and door on the fwd hinge to door mounting bolt. After adjustment, torque the three upper strut door hinge attach nuts to 275 in.-lbs.
26. If shim was installed in step 25, rivet shim to door. Pilot holes are provided in the shim for locating the rivets.
27. Manually close upper and lower strut doors. Adjust the lower strut door, inbd or outbd as required, by utilizing the slotted holes in the plate joining the two parts of the door.
28. Deleted
29. With the upper strut door actuating rod connected to door, manually close upper srut door. Position actuating rod to attachment point on landing gear.
30. Adjust rod end as necessary to provide for bolt installation. Final adjustments to the rod end will be performed during a later step.
31. Extend landing gear and connect upper strut door actuating rod.
32. Install bolt, washers, bushing, and nut (see Figure 32-10-406, Detail E). Torque nut to 290 to 410 in.-lbs.
33. Dump all accumulator hydraulic pressures.
34. Remove aft bolt from lower strut door roller bracket.
35. Temporarily install an eyebolt (AN41-4A or equivalent) with the eye towards the outer skin side (see Detail D, Figure 32-10-406).

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MAINTENANCE PRACTICES (Continued)

Installation (Cont'd)

36. Install washers and nut. Tighten nut.

NOTE

To obtain a slow movement of the main landing gear, ensure that the hydraulic system accumulator (normal and auxiliary) is dumped before the main landing gear is extended or retracted.

37. Slowly retract landing gear to up-and-locked position.

38. Check that upper strut door is held firmly against the wing and fuselage strut door lands. If upper strut door is not held firmly against strut door lands, extend landing gear and adjust upper strut door actuating rod as required. Recheck door position after adjustment.

39. Dump accumulator hydraulic pressure.

40. With the landing gear extended, connect the lower strut door actuating rods.

41. Slowly retract landing gear until the main landing gear uplock roller just touches the main landing gear uplock hook. At this point the upper and lower strut doors should be parallel with each other. This can be checked by positioning a straight edge across the two doors.

42. If doors are not positioned as stated in step 41, extend landing gear and adjust the two lower strut door actuating rods as necessary to obtain the parallel condition of strut doors. Repeat step 41 after each adjustment.

43. With landing gear extended, temporarily install the lower strut door roller in roller bracket.

44. Retract main landing gear and check that lower strut door uplock hook picks up lower strut door roller and hold doors firmly against strut door lands.

45. Adjust uplock roller bracket as necessary, to obtain a firm contact between the strut doors and strut door lands. This adjustment can be made by interchanging washers between the roller bracket and lower strut door. Always install 2W13-4 serrated washers with serrations against base of roller bracket.

46. Extend main landing gear, and place a small amount of clay (model or molding) on the lower strut door uplock hook (see Figure 32-10-406, Detail B). Apply a small amount of grease to clay. This will prevent the clay from sticking to the lower strut door roller.

NOTES

- ***The lower strut door hook can be centered in the roller bracket by positioning the roller bracket fore and aft as necessary to obtain the proper dimensions.***
- ***The hook to roller clearance can be obtained by moving the roller bracket inbd or outbd as necessary. This clearance is checked by measuring the clay that was not displaced during hook to roller engagement. This will provide an accurate method of determining the proper clearance. Ensure that roller bracket is tightened after each adjustment.***

47. Retract landing gear to the full up-and-locked position. Check lower strut door hook to roller clearances. (See Figure 32-10-406, Detail A and B.)

48. If adjustments are required in step 45, extend landing gear. Adjust roller bracket as necessary and repeat step 45.

49. Remove the lower strut door roller and retract landing gear to up and locked position. Attach push-pull scale to eyebolt installed in step 34.

50. Pull on scale until 50, then 100 lbs, are indicated on scale.

NOTE

Strut door deflection shall not exceed 5/16 in. at 50 lbs pull or 5/8 in. at 100 lbs pull.

51. Measure deflection at fwd side of strut doors (see view C, Figure 32-10-406).

52. Extend landing gear and adjust lower door actuating rods as necessary.

53. Install the lower strut door roller in roller bracket.

54. Retract main landing gear to the full up-and-locked position. Check that lower strut door hook fully engages roller and that strut doors are held firmly against strut door lands.

55. Ensure that the red warning light in the landing gear control handle is out when the landing gear is up and locked. Adjust strut fairing door closed switches if necessary (see Figure 32-10-405).

56. Remove eyebolt installed in step 34. Reinstall roller bracket attach bolt, washer, and nut. Tighten nut.

57. Check installation for security and safeties.

58. Check operation of landing gear, normal and emergency extension. Ensure that adequate clearance exists between all components.

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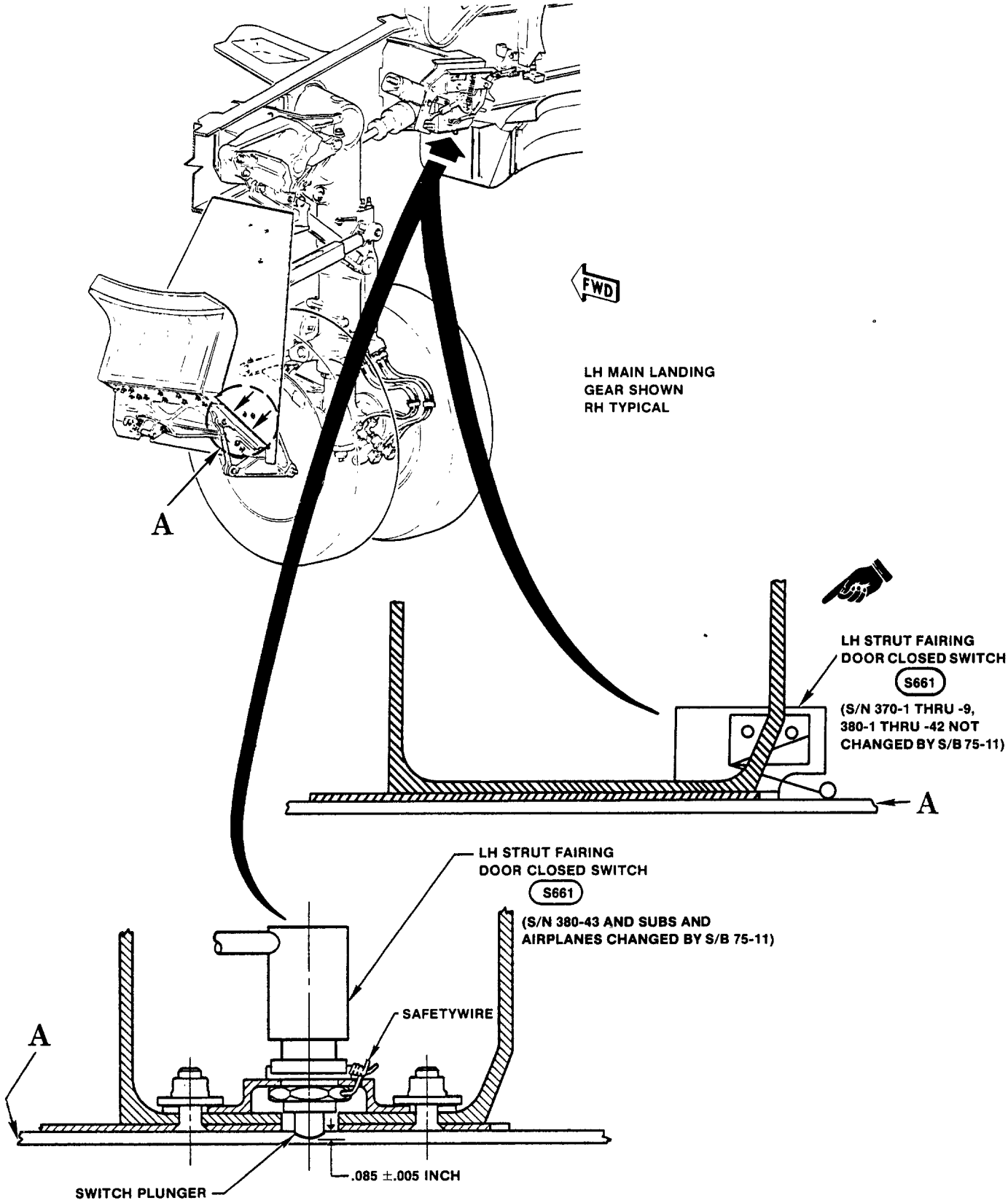

Figure 32-10-405 — Adjusting Strut Fairing Door Closed Switches

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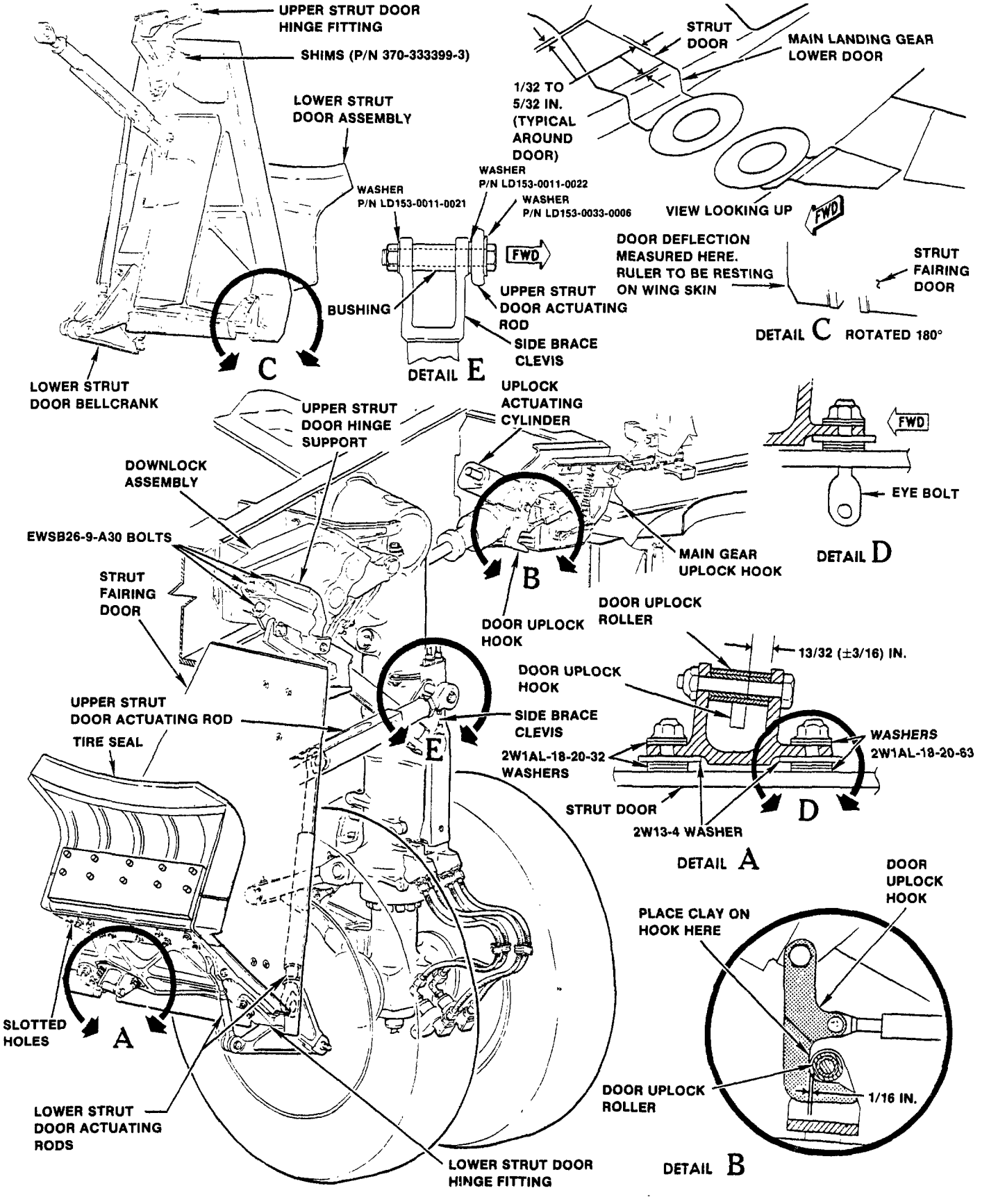

Figure 32-10-406 — Main Landing Gear Strut Doors

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MAINTENANCE PRACTICES (Continued)

ACTUATOR CLEVIS PIN REPLACEMENT

Removal:
(See Figure 32-10-407)

1. Place airplane on jacks and raise airplane enough to permit gear retraction.

WARNING

Before applying landing gear hydraulic pressure for any reason (checks, adjustments, etc.), deplete normal and auxiliary accumulator pressures. This will prevent the auxiliary accumulator power assist retraction system from operating and prevent serious injury to personnel and/or damage to the airplane.

2. Dump hydraulic pressure and normal and auxiliary accumulator pressure.
3. Place hydraulic pump control switch to OFF position.

CAUTION

To prevent possible damage to retract cylinders, do not move landing gear by any means other than hydraulic power.

4. Activate airplane electrical power. Move landing gear control handle to the UP position.
5. If hydraulic test stand is not available, cycle hydraulic pump control switch as necessary to partially retract main landing gear.
6. Place wheel chocks or other suitable stops under both main gear tires.
7. Place hydraulic pump control switch to the OFF position. Dump all hydraulic accumulator pressure.
8. Place landing gear control handle to the DOWN position and let both main gear tires settle back by their own weight in an approximately one-half retracted position against wheel chocks or other suitable stops.
9. Shut off airplane electrical and hydraulic power.
10. Remove cotter pin, nut, washers, and cylinder rod end pin from actuating cylinder.

NOTE

Do not change rod end adjustment

11. Remove actuating cylinder rod end from clevis.
12. Cut safetywire and remove nut and washer from clevis pin.
13. Remove clevis pin and thrust bearing from shock strut.

Installation:
(See Figure 32-10-407)

NOTE

- ***P/N 1127L003-1 clevis pin may be installed as a replacement for P/N 1127L003 clevis pin using the existing hardware; P/N 1127L026 thrust bearing, P/N 1127L004 sleeve bushing, and P/N 1127L005 flanged bushing. Ensure that cupped side of thrust bearing is facing toward the radius runout at head of clevis pin.***
- ***For main landing gear struts with P/N 1127L004-1 and 1127L005 flanged bushings installed into the upper cylinder, P/N 1127L026 thrust bearing is not required.***
- ***Airplanes changed by S/L 82-1 have a flanged bushing installed in place of the thrust bearing.***

1. Install thrust bearing (if required) on clevis pin, (airplanes not changed by S/L 82-1).
2. Install clevis pin in shock strut.
3. Install safetywire washer on end of clevis pin verifying that the flats of the washer engage the flats of clevis pin.
4. Install nut on end of clevis pin and torque to 100 in.-lbs. Verify that the clevis pin will rotate freely in the flanged and sleeve bushings with minimal axial travel. If the clevis pin is not free to rotate, reduce torque on nut until free rotation is obtained. Max axial movement of clevis pin should not exceed 0.005 in. Safetywire nut to washer.
5. Install actuating cylinder rod end in clevis pin, then install rod end pin, washer, and nut. Torque rod pin nut to 100 in.-lbs max and check for 0.005 in. clearance min between actuating cylinder rod end and clevis.
6. Install cotter pin through nut and actuating cylinder rod end pin. Add or remove washers until cotter pin can be inserted.
7. Lubricate clevis pin and actuating cylinder rod end with MIL-G-81827 grease or equivalent.
8. Move landing gear control handle to the UP position. Activate airplane electrical power.

CAUTION

To prevent possible damage to retract cylinder, do not move landing gear by any means other than hydraulic power.

9. Cycle hydraulic pump control switch to partially retract main landing gear sufficiently to remove chocks or stops.
10. Extend landing gear. Ensure that landing gear is down and locked.
11. Perform landing gear extension and retraction checks.
12. With landing gear down and locked, shut off airplane electrical and hydraulic power.
13. Lower and remove jacks from the airplane.

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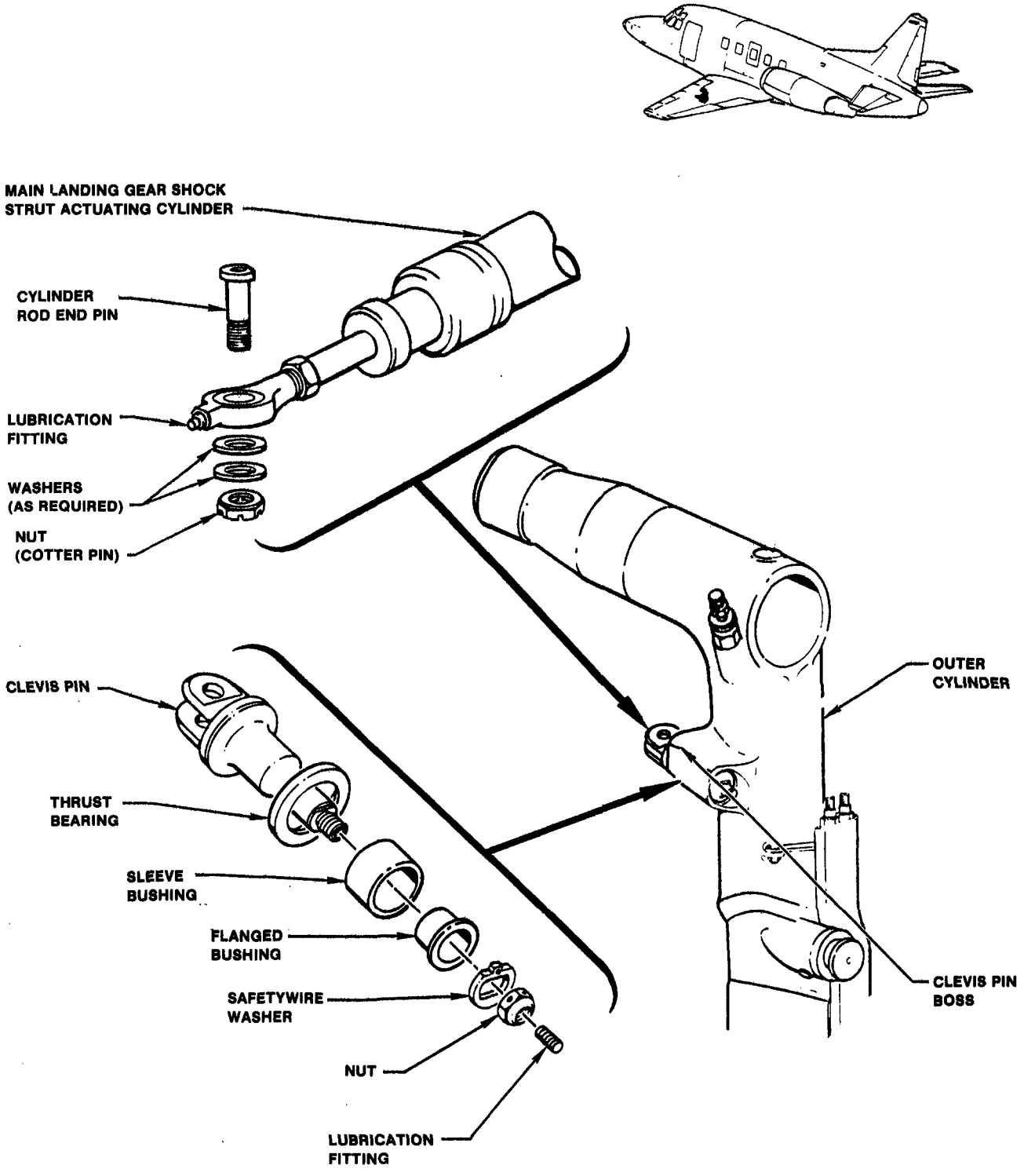

Figure 32-10-407 — Main Landing Gear Actuator Clevis Pin

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MAINTENANCE PRACTICES (Continued)

REPLACEMENT OF MAIN GEAR UPLOCK CYLINDER

Equipment/Tools

- Set of Jacks
- Jacking and Mooring Kit
- Hydraulic Test Stand Capable of Delivering 4 Gpm at 3000 psi if Available
- 28 VDC External Electrical Power

Main Gear Uplock Cylinder Removal

The main gear uplock cylinder is directly above the main landing gear actuating cylinder. (See Figure 32-10-406.)

NOTE

The following procedure is typical for both LH and RH main gear uplock cylinders.

1. Disconnect external electrical and hydraulic power, if connected.

WARNING

Before applying landing gear hydraulic pressure for any reason (checks, adjustments, etc.), deplete normal and auxiliary accumulator hydraulic pressures. This will prevent the auxiliary accumulator power assist retraction system from operating and prevent serious injury to personnel and/or damage to the airplane.

2. Bleed off reservoir air pressure and dump normal and auxiliary accumulator pressure.
3. Place jacks under airplane and raise airplane enough to permit gear retraction.
4. Disconnect link from uplock cylinder at cylinder inbd end. Retain hardware for reuse.
5. Disconnect and cap actuating cylinder hydraulic lines.
6. Remove uplock cylinder attach bolts, nuts, washers, and spacers. Note location of spacers and strut door uplock hook.
7. Remove uplock cylinder and secure strut door uplock hook to prevent movement.
8. Note position of hydraulic fittings and remove fittings from cylinders. Retain fittings for reuse.

Main Gear Uplock Cylinder Installation

NOTE

The following procedures are typical for both LH and RH main gear uplock cylinders.

1. Install new packings and backup rings on hydraulic fittings.
2. Install and tighten fittings in uplock cylinder.
3. Place cylinder into position in uplock fitting and install strut door uplock hook, spacers, bolts, washers, and nuts.
4. Uncap, connect, and tighten uplock cylinder hydraulic lines. Tighten fitting jam nuts.
5. Install pin, washer, and cotter pin attaching cylinder piston rod to link.
6. Check normal and emergency operation of landing gear. Also check operation of landing gear position indicating and warning system. Ensure that adequate clearance exists between all components.

MAIN GEAR SHOCK STRUT PACKING (O-RING) REPLACEMENT

Equipment/Tools and Consumables

- Set of Jacks
- Jacking and Mooring Kit
- Hydraulic Test Stand Capable of Delivering 4 Gpm at 3000 psi if Available
- 28 VDC External Power
- Metering Pin Installation Tool (P/N AT-19, Menasco)
- Adjustable Spanner Wrench (2 to 4-3/4 in.)
- Fabricated Metering Pin Knock-out Tool (Refer to Note Following Step 23.)
- Metering Pin Orifice Tube Installation Tool (P/N AT-8, Menasco)
- Hydraulic Fluid (MIL-H-5606)

Removal:

NOTES

- ***All parts referred to in the following procedures are shown in Figure 32-10-408.***
- ***The following procedures are typical for both LH and RH main landing gear shock struts.***

1. Place jacks under airplane and raise airplane enough to permit landing gear retraction.

WARNING

Before applying landing gear hydraulic pressure for any reason (checks, adjustments, etc.), deplete normal and auxiliary accumulator hydraulic pressures. This will prevent the auxiliary accumulator power assist retraction system from operating and prevent serious injury to personnel and/or damage to the airplane.

2. Remove electrical power, hydraulic pressure and dump normal and auxiliary accumulator pressure on airplane.

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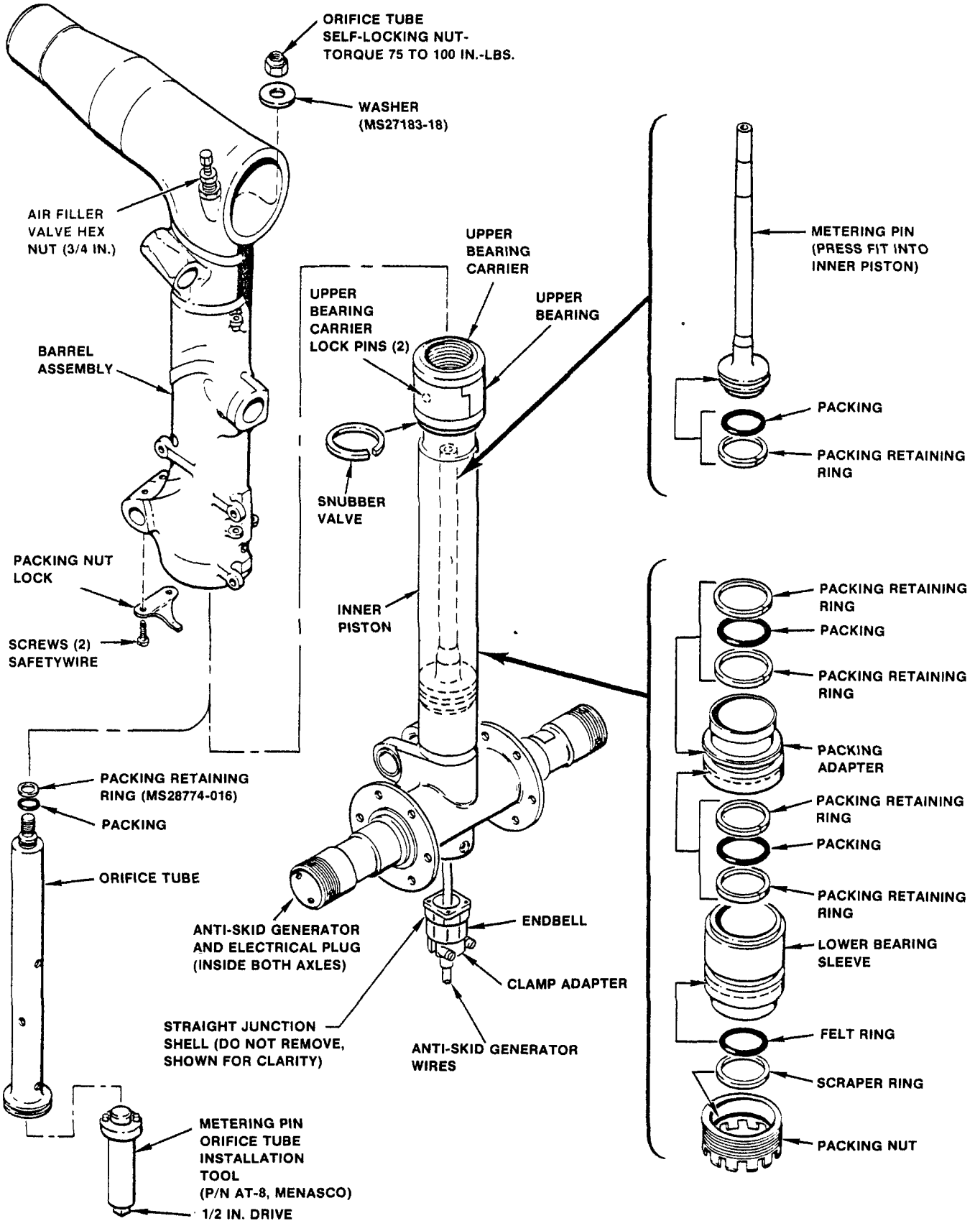

Figure 32-10-408 — Installing Main Gear Shock Strut Packings (O-Rings)

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MAINTENANCE PRACTICES (Continued)

Removal: (Cont'd)

WARNING

To prevent possible serious injury to personnel and/or damage to airplane, make certain that all air is bled from shock strut before disassembly and that all personnel and equipment are clear of landing gear during partial retraction.

NOTE

For easier strut handling, main landing gear wheels and brakes may be removed.

3. Remove strut air filler valve dust cap. Slowly loosen filler valve 3/4 in. hexnut enough to allow air to escape from shock strut. Partially compress strut to release any trapped air.

CAUTION

To prevent possible damage to retract cylinders, do not move landing gear by any means other than hydraulic power.

4. Remove landing gear ground safety locks.
5. Position landing gear handle to UP position.
6. Apply electrical power and slowly apply hydraulic pressure to airplane until main landing gear strut is approximately 30 degree angle from full down position.
7. Using chocks or other suitable stops, support both main landing gear struts approximately 30 degree angle from full down position.

WARNING

Before applying landing gear hydraulic pressure for any reason (checks, adjustments, etc.), deplete normal and auxiliary accumulator hydraulic pressures. This will prevent the auxiliary accumulator power assist retraction system from operating and prevent serious injury to personnel and/or damage to the airplane.

8. Remove electrical power and hydraulic pressure from airplane. Deplete normal and auxiliary accumulator pressure.
9. Disconnect wheel brakes hydraulic lines.
10. Remove lock-ring securing hub caps to wheels, and remove caps from wheels.
11. Remove two bolts retaining each generator to axle, and pull out each generator and disconnect plug.
12. Disassemble anti-skid generator electrical plugs, so as to remove electrical plugs from the axle center hole and inner piston assembly. Refer to Disassembly and Assembly of Anti-Skid Electrical Generator Plugs.
13. Underneath shock strut, loosen cable clamp adapter screws on endbell and remove endbell from straight junction shell.
14. Tag wires (INBD and OUTBD) on a small piece of masking tape and carefully remove anti-skid generator disassembled plugs and wires from bottom of gear strut.
15. Disconnect main gear scissors and secure upper link to upper part of strut.

NOTE

Before the inner piston is removed, a suitable container should be placed under the shock strut to catch the hydraulic fluid contained in the shock strut.

16. Remove packing nut lock, packing nut (with spanner wrench) and inner piston from barrel assembly.
17. Remove upper bearing from upper bearing carrier.
18. Remove the two upper bearing carrier lockpins; then remove bearing carrier from piston.
19. Remove snubber valve from bearing carrier internal gland groove.
20. Remove packing adapter from piston.
21. Remove internal packings (O-rings) and packing retaining rings from packing adapter.
22. Remove lower bearing sleeve and packing nut from piston.
23. Remove felt wiper from lower bearing sleeve and remove scraper ring from gland nut.

NOTES

- ***Perform steps 24 and 25 if hydraulic fluid is leaking from bottom center of inner piston. These procedures remove the metering pin packing for replacement.***
- ***Fabricate a metering pin knock-out tool from a 5/16 in. diameter steel rod (length, 24 in.) with a phenolic tip (radius edge) to prevent damaging aluminum metering pin.***

24. Knock out metering pin from bottom of inner piston with fabricated knock-out tool.

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MAINTENANCE PRACTICES (Continued)

Removal: (Cont'd)

25. Remove packing and packing back-up ring from base of metering pin.

NOTE
Perform the steps 26 through 29 if hydraulic fluid is leaking inside the strut trunnion.

26. Hold orifice tube with orifice tube installation tool (P/N AT-8, Menasco) and remove self-locking nut and washer from inside of trunnion.
27. Remove orifice tube from inside barrel assembly.
28. Remove packing retaining ring and packing from orifice tube.

CAUTION
Do not interchange parts between struts.

29. Clean all nonkitted parts with cleaning solvent. Dry all parts thoroughly with a clean, lint-free cloth and/or compressed air.
30. Check all parts for damage and wear. Replace damaged or worn parts with serviceable items.

NOTE
DELETED.

Installation:

NOTES
- ***During assembly and installation of main gear shock strut, ensure that all parts are serviceable, thoroughly cleaned and lubricated with hydraulic fluid (MIL-H-5606). Refer to Figure 32-10-408 for parts identification and part numbers.***
- ***If metering pin and orifice tube was removed, perform steps 1 through 4.***

1. Install new packing (O-ring) and packing retaining ring in groove on bottom of metering pin.
2. Install metering pin from top of inner piston until metering pin base is on inner piston seat, using metering pin installation tool.
3. Install packing and packing retaining ring in groove on top of orifice tube.
4. Position orifice tube into the strut barrel assembly with orifice tube installation tool (AT-8, Menasco) and install washer and new self-locking nut. Hold orifice tube with installation tool and torque nut to 75 to 100 in.-lbs.
5. Slide packing nut and scraper ring on inner piston, with flat surface of ring facing top of piston.
6. Install felt ring in internal groove in lower bearing sleeve. Slide lower sleeve bearing on piston.
7. Install packing adapter seals as follows.
   A. Install packing with packing retaining ring on each side, external seal groove.
   B. Install packing with packing retaining ring on each side, internal seal groove.
8. Slide packing adapter on inner piston.
9. Slide the snubber valve on inner piston, with greatest diameter side of valve facing top of piston.
10. Install upper bearing carrier tight against the top of the piston and check to see if two of the holes through the wall of the bearing carrier, 180° apart, are in line with the existing holes in the piston. If the holes are in line, install the two lock pins.

    If lockpin holes in bearing carrier and piston do not align perfectly, and the holes can be aligned by backing the nut off within 1/4 turn, install a SK14893-003 laminated shim between the bearing carrier and the piston as shown per Figure 32-10-412. Peel the shim as required to maintain positive contact between the carrier, shim and piston and also achieve a tight fit on the end of the piston.

    If lockpin holes in bearing carrier and piston cannot be aligned perfectly using the SK14893-003 shim, rework as follows:

    A. If the holes in the bearing carrier and the holes in the piston maintain a minimum of two hole diameters minimum edge distance, machine two (2) .2495/.2520 in. diameter holes through piston wall on centerline of existing holes in bearing carrier, using drill bushing as a pilot. Brush cadmium plate new holes in piston.
    B. If holes in bearing carrier and holes in piston have less than two hole diameters minimum edge distance, machine two (2) .2495/.2520 in. diameter holes through piston wall on centerline of the alternate existing holes in carrier 180° apart. Brush cadmium plate new holes in piston.
    C. If neither set of holes in the bearing carrier align with holes in piston, mark two (2) new hole locations on the bearing carrier (180° apart) at a location between the existing holes, maintaining the maximum edge distance possible, and a minimum edge distance of two diameters. Remove bearing carrier and machine two (2) .3740/.3755 in. diameter holes in bearing carrier at the locations marked. Reinstall bearing carrier tight

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MAINTENANCE PRACTICES (Continued)

Installation: (Cont'd)

against the end of piston. Machine two (2) .2495/.2520 in. diameter holes through piston wall on centerline of existing holes in bearing carrier, using a drill bushing as a pilot. Brush cadmium plate new holes in piston and alodine new holes in bearing carrier.

NOTE

If either bearing carrier or piston has three sets of holes (6 total) and the holes in the bearing carrier and the holes in the piston do not align, contact Sabreliner Corporation for a disposition.

11. Install upper bearing on upper bearing carrier.
12. Install inner piston into the bore of the barrel assembly and start the packing nut on barrel.
13. Tighten packing nut with a spanner wrench until nut bottoms out. If packing nut lock does not align with packing nut notch, back off packing nut to the first notch alignment. Install packing nut lock with two screws and lockwire.

NOTE

When connecting main gear scissors, tighten nut to obtain a clearance of 0.001 to 0.002 in. between torque arms at apex point.

14. Connect main gear scissors and wheel brake hydraulic lines. Safety nut with cotter pin.
15. Install anti-skid generator disassembled plugs and wiring through straight junction shell.

NOTE

Be sure the outbd and inbd anti-skid generator plug wires go to their respective generator.

16. Assemble both anti-skid generator electrical plugs. Refer to Disassembly and Assembly of Anti-Skid Electrical Generator Plugs.
17. Connect anti-skid electrical plug to each generator and safetywire.
18. Install anti-skid generators into axle.
19. Position each generator in axle and install two retaining bolts through wheel nut, axle and generator.
20. Install endbell on straight junction shell and tighten cable clamp adapter screws.

CAUTION

Make certain anti-rotation pin between hubcap and wheel is correctly aligned.

21. Perform operational check of anti-skid system and install hubcaps and lockrings on wheels.

CAUTION

To prevent possible damage to retract cylinders, do not move landing gear by any other means than hydraulic power.

22. Retract landing gear and remove chocks or supports, if landing gear is supported at a 30 degree angle. Extend landing gear and install ground safety pins.
23. Remove electrical power and hydraulic pressure from airplane.
24. Service main landing gear shock strut (refer to chapter 12). Check strut for leaks.
25. Perform operational check of landing gear to ensure proper retraction, and shock strut extension.
26. Bleed wheel brake system.

ANTI-SKID ELECTRICAL GENERATOR PLUG DISASSEMBLY AND ASSEMBLY

Anti-Skid Generator Electrical Plug Disassembly

NOTE

These procedures are performed when removing and installing main landing gear shock strut inner piston. Refer to Figure 32-10-409.

WARNING

Before applying landing gear hydraulic pressure for any reason (checks, adjustments, etc.), deplete normal and auxiliary accumulator hydraulic pressures. This will prevent the auxiliary accumulator power assist retraction system from operating and prevent serious injury to personnel and/or damage to the airplane.

1. Remove all electrical power and deplete all hydraulic pressure from airplane.
2. Remove safetywire from electrical plug and loosen clamp adapter screws on the endbell. Insert electrical plug into an anti-skid generator receptacle to keep barrel from rotating while loosening (do not remove) endbell and coupling.
3. Remove electrical plug from generator; then remove endbell from coupling and slide endbell and rubber sleeve back on electrical wires.
4. Remove coupling from barrel and slide coupling back against endbell.
5. Remove barrel and socket insert from insulator, grommet and grommet retainer.

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MAINTENANCE PRACTICES (Continued)

Anti-Skid Generator Electrical Plug Disassembly (Cont'd)

6. Note the letters on end of grommet and identify electrical wires with their corresponding pin (A or B) to ensure that the electrical plug will be assembled properly.
7. Remove grommet retainer from grommet.
8. Holding sockets with one hand, carefully slide insulator and grommet toward endbell end until ceramic retaining rings and silicone bushings are exposed.

NOTE

To prevent damage to ceramic retaining ring, use extreme care when removing ring from socket.

9. Carefully slide silicone bushing(s) toward endbell end until out of diameter B (see illustration).
10. Slide ceramic retaining ring(s) toward endbell end until positioned over diameter B; then carefully remove retaining ring(s).
7. Place landing gear control handle to the DOWN position and let both main gear tires settle back by their own weight in an approximate one half retracted position against wheel chocks or other suitable stops.
8. Shut off airplane electrical and hydraulic power.
9. Remove cotter pin, nut, washers and cylinder rod end pin from actuating cylinder.
10. Remove actuating cylinder rod end from clevis. Do not change rod end adjustment.

CAUTION

Retain all parts removed from anti-skid generator electrical plugs for reinstallation.

11. Remove silicone bushing(s) from socket(s).
12. Holding electrical wires with one hand, remove insulator and grommet from sockets.
13. Remove grommet retainer, coupling, rubber sleeve and endbell from electrical wires.

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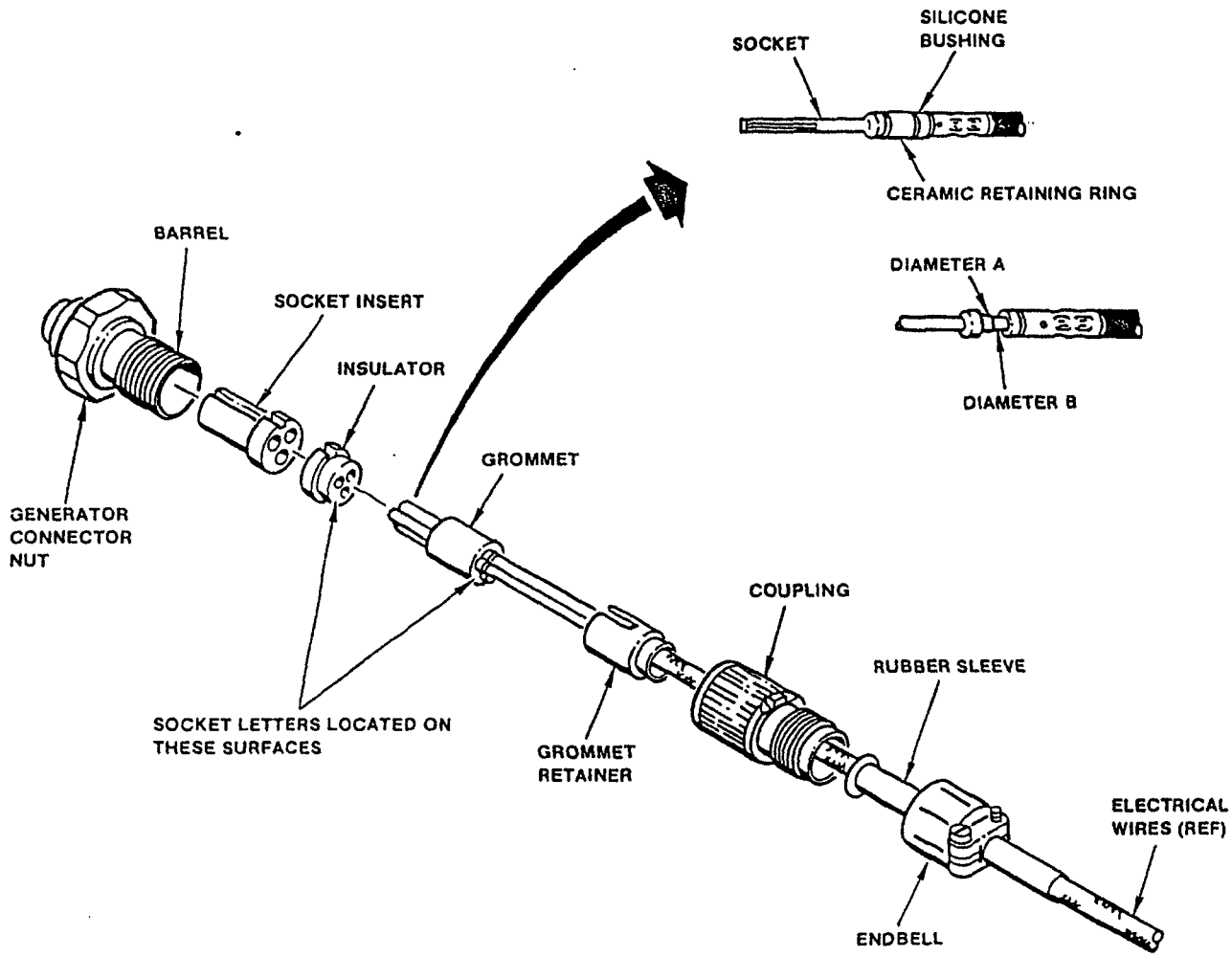

Figure 32-10-409 — Disassembly of Anti-Skid Electrical Generator Plug

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MAINTENANCE PRACTICES (Continued)

Anti-Skid Generator Electrical Plug Assembly

NOTE

Refer to Figure 32-10-409.

1. Remove all electrical power from airplane.
2. Install endbell rubber sleeve (flange end toward barrel), coupling and grommet retainer over electrical wires.

CAUTION

To prevent possible damage to anti-skid system control box, ensure that sockets are inserted in proper grommet holes.

3. Install correct socket (A or B) in its corresponding hole in the grommet. Letters on grommet must be toward endbell end. (Refer to illustration.)
4. Slide grommet and insulator over socket until diameter A and B are exposed through the insulator. Check that corresponding sockets A and B are mated with corresponding holes in grommet and insulator.
5. Slide a silicone bushing over each socket and position on sockets past diameter B.
6. Position ceramic retaining ring over diameter B; then carefully slide ceramic retaining ring up on diameter A of socket(s).
7. Slide silicone bushing on diameter B of socket(s).
8. Slide grommet retainer over grommet. (Ensure that keyway in grommet retainer is aligned with keyway in insulator.)
9. Hold electrical wires and pull grommet, grommet retainer, and insulator (toward barrel end) until ceramic retaining rings are positioned in insulator. Insert spare socket in remaining hole in insulator.
10. Place socket insert over sockets so that keyways are aligned on insulator and socket insert.
11. Locate index key in barrel; then install barrel over grommet.
12. Align grommet retainer keyway with barrel index key and slide grommet retainer in barrel. Push grommet retainer in barrel until keyway in grommet retainer just disappears in barrel.
13. Remove electrical wire identification tag(s).
14. Place flange of rubber sleeve against end of grommet retainer. Install coupling on barrel and endbell on coupling.
15. To keep barrel from rotating while tightening endbell and coupling, index electrical plug to anti-skid generator receptacle; then hand tighten coupling and endbell.
16. Tighten wire clamp screws on endbell.
17. Remove electrical plug from anti-skid generator and lockwire endbell to coupling.
18. Check continuity (Simpson Model 260, or equivalent) between both anti-skid generator electrical plugs to anti-skid control box (CN1000). Refer to Figures 32-40-1 and 32-40-1A.
    A. On S/N 370-2 thru -6, -8, and -9 disconnect plug P1062 at control box and check continuity as follows:
        (1) LH outbd anti-skid generator plug (P1087)

            Pin B (P1087) to airplane ground
            Pin A (P1087) to pin E (P1062)

        (2) LH inbd anti-skid generator plug (P1063)

            Pin A (P1063) to airplane ground
            Pin B (P1063) to pin D (P1062)

        (3) RH outbd anti-skid generator plug (P1086)

            Pin B (P1086) to pin B (P1062)
            Pin A (P1086) to airplane ground

        (4) RH inbd anti-skid generator plug (P1066)

            Pin B to airplane ground
            Pin A to pin C (P1062)

        (5) Connect plug P1062 to control box and lockwire.

    B. On S/N 370-1 and -7 and S/N 380-1 and Subs, disconnect plugs P964 and P965 at control box and check continuity as follows:
        (1) LH outbd andi-skid generator plug (P1087)

            Pin A (P1087) to pin A (P964)
            Pin B (P1087) to pin B (P964)

        (2) LH inbd anti-skid generator plug (P1063)

            Pin A (P1063) to pin X (P964)
            Pin B (P1063) to pin G (P964)

        (3) RH outbd anti-skid generator plug (P1086)

            Pin A (P1086) to pin B (P965)
            Pin B (P1086) to pin A (P965)

        (4) RH inbd anti-skid generator plug (P1066)

            Pin A (P1066) to pin G (P965)
            Pin B (P1066) to pin X (P965)

        (5) Connect plugs P964 and P965 to control box and safetywire.

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MAINTENANCE PRACTICES (Continued)

MAIN LANDING GEAR TORQUE ARM BUSHING REPLACEMENT

NOTES

- ***Bushings must be replaced in pairs.***
- ***Refer to Figure 32-10-410.***

Removal:

1. Remove torque arms from main landing gear.
2. Press defective bushings out of torque arms using an arbor press and appropriate adapter.
3. Clean bore and inspect for scratches, corrosion, out-of-round, or other defects.

Installation:

1. Install bushings per steps A thru E.
   A. Coat outside of bushing with zinc chromate primer.
   B. Place torque arm on bed of arbor press.
   C. Place bushing and bushing installation tool in position on torque arm.
   D. Align hole in root bushing with tapped hole in torque arm to allow free passage of grease.
   E. Carefully press bushing into bore while primer is still wet.
2. Install torque arm assembly in mill holding fixture. Mill the flange of bushings to dimensions shown in Figure 32-10-410.
3. Install torque arm assembly in the ream holding fixture. Line ream the bushings to dimensions shown in Figure 32-10-410.
4. Install torque arms on main landing gear. Grease fittings (refer to chapter 12).

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NOTES

- *Allowable side play at upper and lower strut hinge points are .0005 in. min, .0065 in. max.*
- *Bushings must be replaced in pairs.*
- *Outside of bushing must be wet with zinc chromate primer when pressing bushing into bore.*

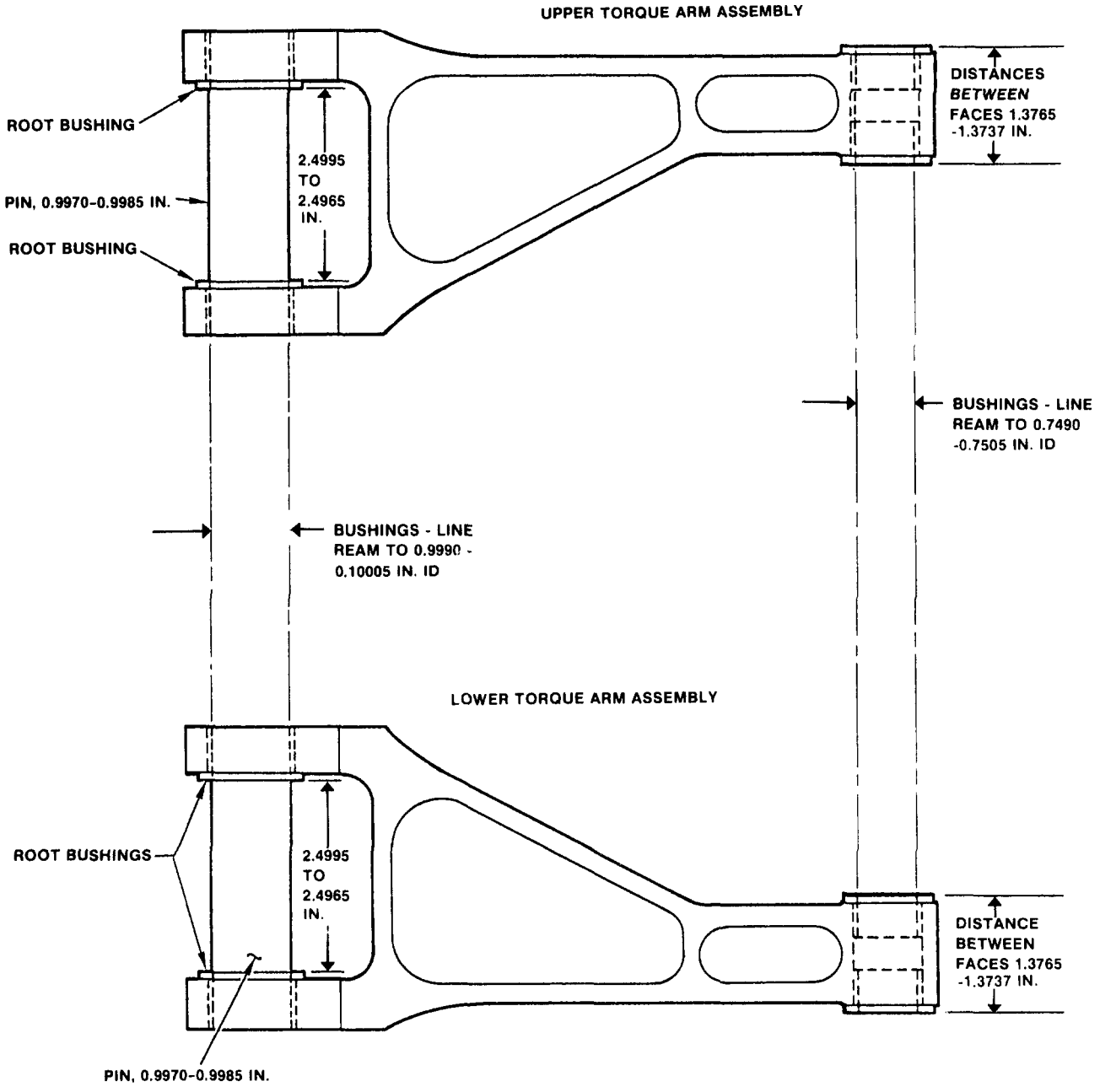
UPPER TORQUE ARM ASSEMBLY
ROOT BUSHING
PIN, 0.9970-0.9985 IN.
ROOT BUSHING
2.4995 TO 2.4965 IN.
DISTANCES BETWEEN FACES 1.3765 -1.3737 IN.
BUSHINGS - LINE REAM TO 0.7490 -0.7505 IN. ID
BUSHINGS - LINE REAM TO 0.9990 - 0.10005 IN. ID
LOWER TORQUE ARM ASSEMBLY
ROOT BUSHINGS
2.4995 TO 2.4965 IN.
DISTANCE BETWEEN FACES 1.3765 -1.3737 IN.
PIN, 0.9970-0.9985 IN.

Figure 32-10-410 — Main Landing Gear Scissors Bushings Replacement

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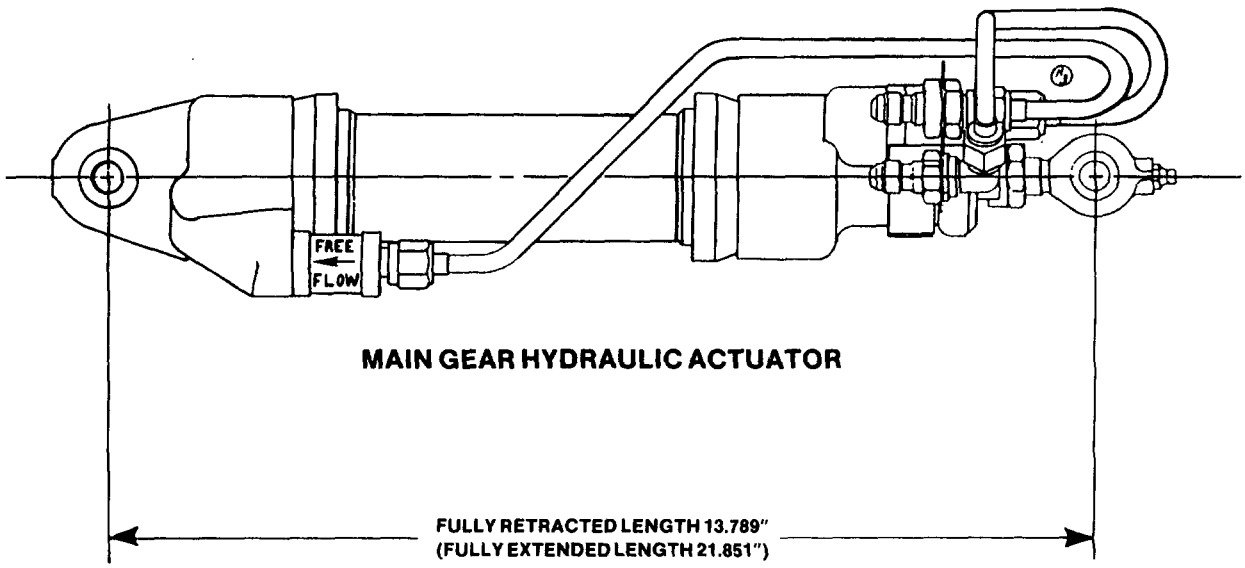

Figure 32-10-411 — Main Landing Gear Actuator Rod End Dimensions

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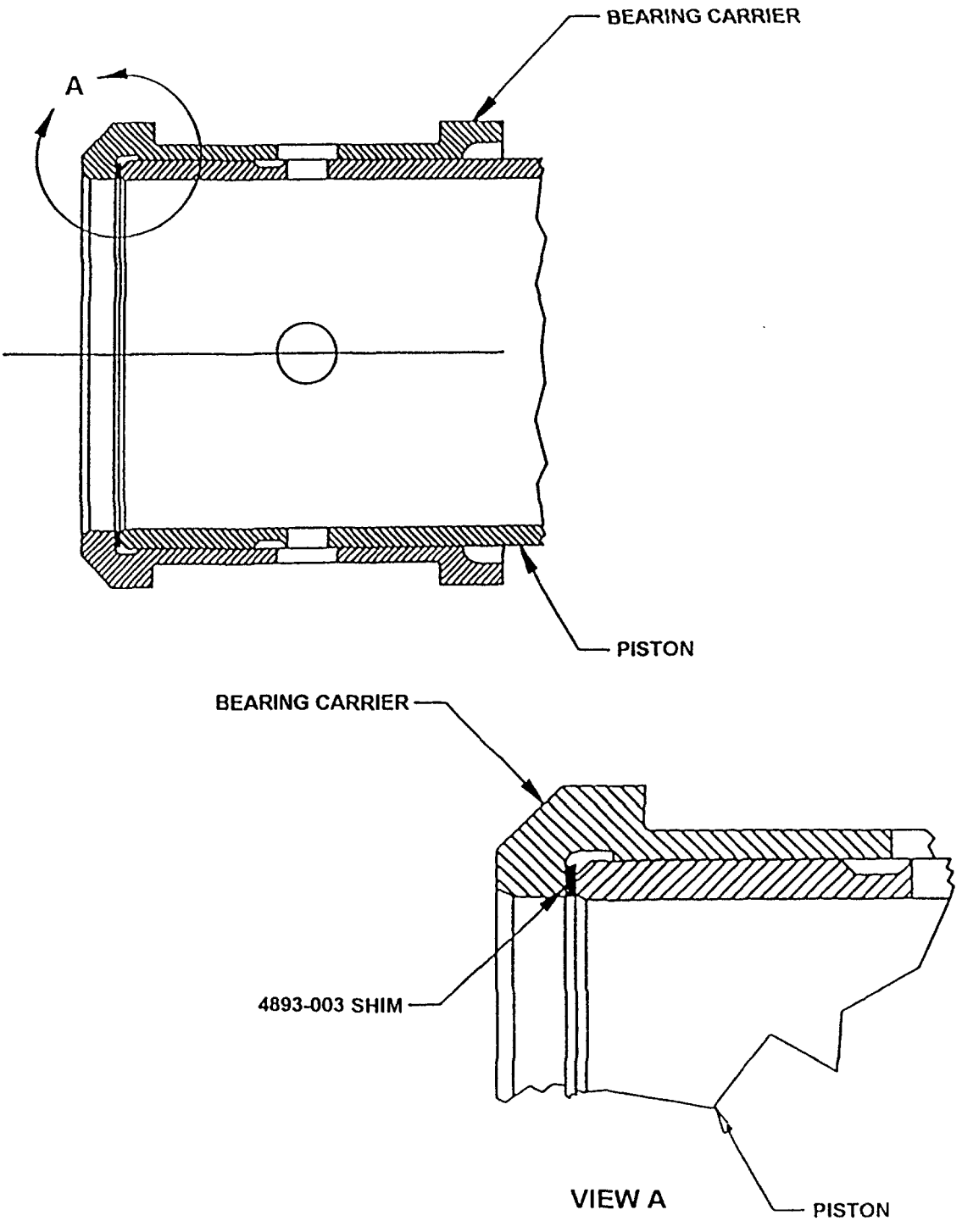

Figure 32-10-412 — Installation of Main Gear Shock Strut Packings

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NOSE GEAR AND DOORS

DESCRIPTION AND OPERATION

NOSE GEAR SHOCK STRUT

The nose gear shock strut is of the air-and-oil, shock-absorber type, containing a specific amount of compressed air and hydraulic fluid. The strut consists of an outer cylinder assembly at the upper end and an inner cylinder assembly at the lower end. The outer cylinder assembly and the inner cylinder assembly are connected by torque links and steering links.

The outside of the outer cylinder is machined to provide trunnion support fittings and attach fittings for towing the airplane and for the nose wheel steering mechanism, actuating cylinder, drag brace, and upper torque links. A filler valve for servicing purposes is at the top of the strut. Internally, the outer cylinder consists of an orifice support tube, extending down from the top of the cylinder to support an orifice in the lower portion of the cylinder. A centering cam and inner cylinder support assembly are at the bottom of the cylinder. The piston, or inner cylinder assembly, consists of the lower torque links and axle. The inner cylinder consists of a piston and a metering rod extending through the orifice in the cylinder assembly, a small orifice in a chamber between the inner cylinder and outer cylinder assembly, and a centering cam at the lower end of the cylinder.

NOTE

During overhaul procedures of the nose gear shock strut, it is possible for the upper and lower bearings incorporating the centering cams to be installed 180 degrees out-of-phase when the strut is reassembled. This would prevent full extension, center locking, and proper retraction of the strut.

The upper portion of the inner cylinder is filled with hydraulic fluid. The outer cylinder contains both air and oil. The proportion of air to oil is determined by strut extension. The strut is designed to absorb impact loads during takeoff, landing, taxiing, and braking operation.

For nose wheel steering operation, the torque links connected between the inner cylinder assembly and outer cylinder assembly strike a stop attached to the outer cylinder, preventing rotation of the nose wheel beyond 35-degrees left or right (Model NA 265-70 and NA 265-80 airplanes not changed by SOK 24) or 45 ±1/2 degrees left to right (Model NA 265-70 and NA 265-80 airplanes changed by SOK 24). When the airplane is to be towed, a latch at the forward side of the strut is disengaged, allowing the nose wheel to turn 360 degrees.

NOSE GEAR ACTUATING CYLINDER

NOTE

When adjusting the actuating cylinder, it is important to understand that the nose gear actuating cylinder serves a dual purpose: it extends and retracts the gear, and it serves as an upstop for the nose gear. When the cylinder is fully extended, the piston will bottom out on the end cap, thus becoming a mechanical stop.

The nose gear actuating cylinder extends and retracts the nose gear. It is a hydraulically operated, unbalanced-type cylinder rated at 3,000 psi. The cylinder is in the aft end of the nose wheel well. It is attached to the right side of the drag brace at one end and to the shock strut at the other end. The cylinder is retracted to lower the nose gear and is extended to raise the gear. The nose gear actuating cylinder also folds the drag brace during gear retraction, and unfolds the drag brace to the down-and-locked position when the nose gear is extended.

CAUTION

Under no circumstances should the nose gear actuating cylinder bottom out in the fully retracted position. The airplane downlock support stop is adjusted to be the nose gear downstop.

An adjustable rod end permits varying the piston rod length. Connecting attachment points on each end of the unit have grease fittings for lubrication.

NOSE GEAR UPLOCK ACTUATING CYLINDER

The nose gear uplock actuating cylinder is attached to the nose gear uplock mechanism at the top of the nose wheel well. The actuating cylinder is a balanced-type unit rated at 3,000 psi. There are no hooks or adjustable rod ends attached to the actuating cylinder. The actuating cylinder piston extends from the actuating cylinder body, acting as a plunger. In operation, the piston rod is retracted out of the path of the gear uplock hook during retraction of the nose gear. During the landing gear extension cycle, the nose gear is unlocked by extension of the piston rod, pushing the uplock hook and forcing it to the unlocked position.

NOSE GEAR DOWNLOCK ACTUATING CYLINDER

The nose gear downlock actuating cylinder is attached to a cross member of the nose gear drag brace. Hydraulic

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DESCRIPTION AND OPERATION (Continued)

NOSE GEAR DOWNLOCK ACTUATING CYLINDER (Cont'd)

lines are connected to opposite sides of a balance-type piston. When the landing gear system is actuated, hydraulic pressure moves the piston to either the extended or retracted position. An arm is installed between twin downlock pins located on each side of the nose gear drag brace. Hydraulic pressure when applied to one side of the actuating cylinder piston pushes the piston rod against this arm, pulling both downlock pins simultaneously from the extended (locked) position. When hydraulic pressure is applied to the opposite side of the actuating cylinder piston, the piston rod moves away from the lockpin arm and the downlock pins are closed by their springs.

NOSE GEAR DRAG BRACE ASSEMBLY

The nose gear drag brace assembly is a folding H-shaped member attached to the wheel well structure and to both sides of the nose gear shock strut. Twin spring-loaded downlock pins on each side of the drag brace hold the nose gear to the down-and-locked position. The downlock pins are unlocked simultaneously by a downlock actuating cylinder mounted on the cross member of the drag brace. Springs push the downlock pins to the extended (locked) position. During landing gear retraction, the downlock pins are pulled by the downlock actuating cylinder. The drag brace is pivoted by hydraulic pressure applied to the nose gear actuating cylinder attached to the forward end of the drag brace assembly. This causes the drag brace to fold and the nose gear to retract. During landing gear extension, the drag brace is unfolded as the nose gear is extended. The downlock pins are extended and ride the circular cams on each leg of the drag brace until the drag brace is unfolded. The downlock pins are then seated in the lower position of the drag brace assembly, holding the nose gear down and locked.

NOSE GEAR FAIRING DOORS

The nose gear fairing doors are hinged on the right and left sides of the nose wheel well. The doors are mechanically actuated to the closed and open positions by retraction and extension of the nose gear shock strut. They are locked in the closed position by mechanical linkage attached to the nose wheel well structure and by door lock rollers engaged with an uplock hook at the forward end of the nose wheel well. The uplock hook is held in the locked position by a torsion spring and is actuated by the nose gear uplock mechanism. When the nose gear is extended, the fairing door uplock hooks are pulled to the unlocked position when the nose gear uplock hook is moved to the unlocked position. Extension of the nose gear shock strut pushes the fairing doors open ahead of the strut to the full open and overcenter position. Retraction of the nose gear pulls the fairing door closed behind the nose gear shock strut. When the fairing doors near the closed position, the door uplock roller cams the door uplock hook back and allows the hook to be seated under the door uplock roller, locking the doors in the closed position.

NOSE GEAR RESTRICTORS

Two nonadjustable orifice-type flow restrictors are in the nose gear hydraulic lines. The restrictors control the rate of landing gear extension. An internal metal-mesh filter element is used in each restrictor to protect the orifice from clogging. The restrictors are connected to the hydraulic lines attached to the nose gear actuating cylinder.

NOSE GEAR BALANCE WEIGHT

A 10-pound dynamic balance weight significantly improves the nose strut dynamic stability. It is attached on a bracket 5 inches forward of the wheel axle.

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NOSE GEAR AND DOORS

MAINTENANCE PRACTICES

REPLACEMENT OF NOSE GEAR SHOCK STRUT ASSEMBLY

Equipment/Tools

- Set of jacks
- Jacking and mooring kit
- Hydraulic test stand capable of delivering 4 gpm at 3,000 psi
- 28 VDC electrical power

Removal:

WARNING

Before applying landing gear hydraulic and auxiliary accumulator pressure for check, adjustments, etc., deplete normal and auxiliary accumulator pressure. This will prevent the auxiliary accumulator power assist retraction system from operating and will prevent serious injury to personnel and/or damage to the airplane.

WARNING

When working in or around wheel well areas, use ground maintenance locks on MLG actuators and pin the nose gear down to prevent inadvertant retraction. Failure to do this could result in serious injury or death.

1. Place airplane on jacks and raise enough to permit gear retraction. Refer to chapter 7.

WARNING

Verify that both normal and auxiliary hydraulic pressure have depleted to 0 psi.

2. Shut off hydraulic pressure and remove electrical power from airplane.
3. Deplete normal and auxiliary hydraulic systems.
4. Remove nose gear wheel assemblies.
5. Remove nose gear load switch and electrical wiring.
6. Remove nose wheel steering actuating cylinder and feed-back potentiometer. Do not remove feed-back potentiometer from actuating cylinder.

NOTE

If nose gear shock strut assembly is to be replaced, remove steering potentiometer assembly electrical disconnect bracket.

7. Remove clamps from strut and remove electrical wiring from the side of strut. Lay electrical wiring over nose gear drag brace.
8. Disconnect nose gear actuating cylinder at arm on nose gear strut.
9. Disconnect nose gear drag brace at both sides of nose gear strut. If necessary, disconnect overcenter bungee attached to the left side of nose gear drag brace. This will allow drag brace to hang down away from nose gear shock strut.
10. Remove access panels to gain access to nose gear strut trunnion pins.
11. Working at left and right sides of nose gear trunnion, remove left and right trunnion pin retainer bolts.
12. Working at outboard side of left and right trunnion fittings, cut lockwire and remove cap from trunnion pins. On some airplanes, remove screws and nuts securing caps.
13. Remove cotter pin, nut, and washers from left and right trunnion pins.
14. With an assistant, or assistants, supporting the nose gear strut, pull left and right trunnion pins and remove strut.
15. If nose gear shock strut is to be replaced, remove balance weight and bracket assembly. Ensure that washers installed between bracket and nose gear uplock fitting are kept with the applicable side of balance weight bracket. This will ensure that balance weight bracket is properly centered during installation of bracket.

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Installation and Adjustment of Nose Gear Shock Strut Assembly

CAUTION

Torque links (scissors) must be free to rotate. Do not overtighten.

1. Raise strut into position and temporarily insert both trunnion pins. Visually center strut in nose wheel well. Support strut in this position and pull right trunnion pin out of strut trunnion. Add 2W1AL-64-42-32 washers as necessary between strut trunnion and trunnion bearing in fuselage to obtain a 0.005 to 0.010-inch clearance between inserted washers and trunnion bearing. Push trunnion pin into strut trunnion. Insert trunnion pin retainer bolt through strut trunnion and trunnion pin with bolthead facing aft. Install and tighten retainer bolt nut.
2. Push strut against right trunnion bearing. Working in access, position washer on trunnion pin. Install trunnion pin nut. Tighten trunnion pin nut until a 0.005 to 0.010-inch clearance is obtained between washer and trunnion bearing when cotter pin holes in trunnion nut and trunnion pin are aligned. (Torque of trunnion nut should not exceed 800 inch-pounds.) Install cotter pin.
3. Leave strut against right trunnion bearing. Support strut and back left trunnion. Add 2W1AL-64-42-32 washers as necessary between strut trunnion and trunnion bearing in fuselage to obtain 0.005 to 0.010-inch clearance between inserted washers and trunnion bearing. Push trunnion pin into strut trunnion. Install trunnion pin retainer bolt with bolthead facing aft. Install and tighten the retainer bolt nut.
4. Push strut against left trunnion bearing or as far as it can be pushed. Working in access, position washer on trunnion pin. Install trunnion pin nut. Tighten trunnion pin nut until a 0.005 to 0.010-inch clearance is obtained between washer and trunnion bearing when cotter pin holes in trunnion nut and trunnion pin are aligned. (Torque of trunnion nut should not exceed 800 inch-pounds.) Install cotter pin. With gear installed, the total clearance at one side should be 0.010 to 0.020 inches with gear held tight against the opposite trunnion.
5. Position cap over trunnion pins. Secure cap in place (torque 100 to 140 inch-pounds) and lockwire.
6. Connect drag brace overcenter bungee if bungee was disconnected during nose gear strut removal procedure.
7. Connect nose gear drag brace to left and right sides of nose gear strut. Install and tighten drag brace retainer pins. Torque retainer pin nuts not to exceed 100 inch-pounds so that drag brace is free to rotate. Install cotter pins through drag brace retainer pin nuts.

NOTE

With nose gear down and locked, overcenter bungee should be compressed 9/32 (+1/32, -0) inch as measured at lower face of bungee piston assembly.

8. Check clearance between drag brace downlock pin and ramp. Adjust if necessary.
9. Install electrical harness on side of nose gear strut. Provide enough slack between Adel clamp and disconnect plug to allow full nose wheel steering rotation without interference. On back side of strut, be sure electrical harness has 1-3/4 (±3/4) inch of slack between strut and wire bundle.
10. Install and adjust nose gear load switch.
11. If a new nose gear shock strut assembly is being installed, remove the Loud electrical disconnect bracket and replace it with an NR265-542209 bracket.
12. Connect and safety nose wheel steering feed-back potentiometer electrical plug with lockwire.
13. Install nose gear wheels. Fill nose gear shock strut with hydraulic fluid and inflate strut with air. (Refer to chapter 12.)
14. If not previously removed, disconnect nose wheel fairing door linkage. This will permit adjustment of the nose gear uplock hook.
15. Connect trunnion end of actuating cylinder to the drag brace. Install attach bolt with bolthead outboard. Install and tighten attach bolt nut and washer.

CAUTION

When installing a new actuator and prior to connecting the actuator to the strut (connected in strut well only), position the actuator so that the rod end will clear the adjacent structure. Hydraulically extend and retract (slowly) the actuator several cycles to eliminate any trapped air. To prevent damage to the system, use a low hydraulic pressure during the cycling.

16. With nose gear down and locked, fully bottom actuating cylinder piston in actuating cylinder. Adjust actuating cylinder rod end to obtain a slip-fit of rod end attach bolt to attach point on nose gear shock strut. Then, shorten rod end one full turn and one-half. Lock rod end jam nut.

NOTE

This is the initial adjustment of the rod end. The rod end will be re-adjusted in subsequent steps.

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MAINTENANCE PRACTICES (Continued)

Installation and Adjustment of Nose Gear Shock Strut Assembly (Cont'd)

17. Remove upstop bumper.

WARNING

Before applying landing gear hydraulic pressure for checks, adjustments, etc., deplete normal and auxiliary accumulator pressure. This will prevent the auxiliary accumulator power assist retraction system from operating, and prevent serious injury to personnel and/or damage to the airplane.

NOTE

This rigging must be accomplished at 1800 psi by installing the main gear ground safety locks and maintaining 1800 psi using the airplane hydraulic pump or a hydraulic test stand.

18. Install main gear ground safety locks.
19. Slowly increase hydraulic pressure to 1800 psi while retracting the nose gear only to the locked position.
20. If necessary, loosen rod end until roller makes contact with hook.
21. Inspect area for contact of strut components, tires, or wheels with other items in the wheel well. If contact is made with roller contacting hook, the item making contact with strut component should be moved or cleared of nose gear.

NOTE

The maximum dimension from the rod end to the cylinder is 1-17/32 in. (See Figure 32-20-412.) If this dimension and step 16 cannot be met, check length of retract cylinder assembly (see Figure 32-10-411).

22. Check for a 1/8 (+1/32, -0) inch clearance between top of nose gear uplock hook and end of nose gear uplock hook actuating cylinder piston (hydraulic pressure applied). If clearance is not within tolerance, extend nose gear and adjust nose gear uplock roller on serations to obtain required clearance.
23. Reinstall upstop bumper.
24. On S/N 370-1 thru -9 and 380-1 thru -27 not changed by S/B 74-20, adjust nose gear bumper stop to just contact nose gear strut. Extend nose gear. Then, extend nose gear bumper stop one half turn. Tighten and lockwire stop jam nut.
25. On S/N 380-28 and subs, and airplanes changed by S/B 74-20, adjust nose gear ice breaker bolt to just contact rubber pad bonded to nose gear strut with hydraulic pressure off. Extend nose gear. Then, extend ice breaker bolt three turns. Tighten and lockwire stop jam nut.
26. Retract nose gear to position nose gear strut between tips of nose gear fairing door operating cams. Support nose gear in this position.
27. Adjust door-open stop bolts so that nose gear strut is centered between tips of door operating cams. Then, adjust stop bolt on one side to provide a 1/32 (+1/32, -0) inch clearance between head of stop bolt and nose wheel well structure. Tighten stop jam nuts.
28. Disconnect one side of nose wheel fairing door. Retract nose gear. Check that cable attached to wheel fairing door lock hook is adjusted to slip-fit at attach point to door lock hook. This should be carefully checked because if the cable is too tight, the hook will be partially pulled from under the door lock roller. If the cable is too loose, movement of the nose gear uplock hook to unlocked position will not pull the hook clear of the door lock rollers, causing the gear and doors to hang up on the extension cycle.
29. Extend landing gear and connect wheel fairing door link. Check gear installation for all safeties.

NOTE

When installing the balance weight and torquing attach screws (2), the axle must be torqued prior to torquing the attach screws.

30. Install nose gear balance weight to bracket. Torque screws (2) to 20-25 inch-pounds. Then, install balance weight assembly on nose gear. Ensure that balance weight bracket is centered on nose gear uplock fitting.
31. Check normal and emergency operation of landing gear system.
32. Check operation of landing gear position indicating and warning system.
33. Lower airplane and remove jacks.

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MAINTENANCE PRACTICES (Continued)

REPLACEMENT OF NOSE GEAR DRAG BRACE ASSEMBLY (See Figure 32-20-402)

Removal:

WARNING

Before applying landing gear hydraulic and auxiliary accumulator pressure for check, adjustments, etc., deplete normal and auxiliary accumulator pressure. This will prevent the auxiliary accumulator power assist retraction system from operating and will prevent serious injury to personnel and/or damage to the airplane.

WARNING

When working in or around wheel well areas, use ground maintenance locks on MLG actuators and pin the nose gear down to prevent inadvertant retraction. Failure to do this could result in serious injury or death.

1. Place airplane on jacks and raise enough to permit gear retractions. Refer to chapter 7.

WARNING

Verify that both normal and auxiliary hydraulic pressure have depleted to 0 psi.

2. Shut off hydraulic pressure and remove electrical power from airplane.
3. Deplete normal and auxiliary hydraulic systems.
4. Disconnect nose gear downlock switch link at drag brace. Remove two bolts securing downlock switch to drag brace. Remove downlock switch and electrical wiring from drag brace.
5. Disconnect and cap nose gear downlock cylinder hydraulic lines at flexible lines disconnects.
6. Disconnect drag brace at each side of nose gear shock strut.
7. Disconnect overcenter bungee from left leg of drag brace.
8. Remove accesses to nose gear drag brace trunnions.
9. Cut lockwire and remove cap over the end of drag brace trunnion.
10. Remove drag brace trunnion pin retainer bolts.
11. Have an assistant support drag brace while removing the left and right trunnion pin nuts and washers.
12. Remove trunnion pins from drag brace from nose wheel well side of trunnion pin. Remove drag brace.

Installation and Adjustment of Nose Gear Drag Brace Assembly

1. Position nose gear drag brace in nose wheel well area. Temporarily insert drag brace trunnion pins in left and right sides of drag brace.
2. Pull left trunnion pin back and insert 265-348810-1 washer and 2W1C-48-36-32 washers between drag brace and side of drag brace trunnion bearing. The 265-348810-1 washer must be positioned with the countersunk side next to the drag brace.
3. Push left trunnion through trunnion bearing.
4. Install trunnion pin retainer screw through drag brace trunnion, trunnion pin, and drag brace overcenter bungee support bracket. Install screw with screwhead down. Install nut and tighten screw.
5. Working in access, install washer (2W1AL-37-32-125) and nut on drag brace trunnion pin. Tighten nut until washer contacts bearing with zero play. Tighten nut to next drive pin hole, and install drive pin (NAS561P4-16). Check that trunnion pin is free to rotate.
6. Install drag brace right trunnion pin. Repeat steps 2 through 5 except in step 4, install bolt through trunnion and trunnion pin instead of screw.
7. Install caps over ends of left and right drag brace trunnion pin. Torque cap to 100-140 inch-pounds. Safety caps with lockwire.
8. Connect overcenter bungee to drag brace. Install and tighten attach bolt. Connect and adjust nose gear actuating cylinder.
9. Connect drag brace to left and right sides of nose gear shock strut. Install washer between drag brace bearing and nose gear shock strut and under head of drag brace trunnion pin retainer nut. Install, tighten, and safety retainer nut. Torque retainer nut not to exceed 100 inch-pounds so that drag brace is free to rotate.

NOTE

With the nose gear down and locked, the overcenter bungee should be compressed 9/32 (+1/32, -0) inch as measured at the lower face of the bungee piston assembly.

10. Uncap, connect, and tighten hydraulic lines to downlock actuating cylinder.
11. Install and adjust nose gear downlock switch.

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MAINTENANCE PRACTICES (Continued)

Installation and Adjustment of Nose Gear Drag Brace Assembly (Cont'd)

12. With nose gear down and locked, check clearance between downlock pin and downlock pin ramp at left and right sides of nose gear drag brace. There should be 0.005- to 0.015-inch clearance (with hydraulic pressure applied) between the downlock pin and ramp. If clearance is not within tolerance, partially retract nose gear and support it in the partially retracted position. Remove screw securing ramp to drag brace. Add or remove shims as necessary to obtain required clearance.

NOTE

The screw securing the ramp to the drag brace contains a special nylon locking insert. The screw should be replaced with a new lockscrew (NAS1162-2LW) each time it is removed.

13. With nose gear down and locked (no hydraulic pressure applied), adjust stop bolt in downlock crossbar assembly to provide a gap of 0.000 to 0.010 inch (maximum) between stop bolt head and the end of piston rod in downlock cylinder assembly.
14. Lubricate nose gear drag brace.
15. Check normal and emergency operation of landing gear system. Check that nose gear position indicator shows that nose gear is down and locked. Check hydraulic lines for evidence of hydraulic leaks.

REMOVAL OF NOSE GEAR STRUT INNER PISTON

NOTE

Removal of the inner piston of the nose landing gear shock strut is required to adjust the clearance between the nose gear steering collar and gland nut, or whenever packings (O-rings) replacement is required. The parts referred to in the following procedures are identified in Figure 32-20-403.

Equipment/Tools

- Set of jacks
- Jacking and mooring kit
- Hydraulic test stand capable of delivering 4 gpm at 3000 psi, if available
- 28 VDC external electric power, if available
- Metering pin installation tool (P/N AT-19, Menasco)
- Hydraulic fluid (MIL-H-5606)
- Adjustable spanner wrench (2 to 4-3/4 inch)

WARNING

Before applying landing gear hydraulic and auxiliary accumulator pressure for check, adjustments, etc., deplete normal and auxiliary accumulator pressure. This will prevent the auxiliary accumulator power assist retraction system from operating and will prevent serious injury to personnel and/or damage to the airplane.

WARNING

When working in or around wheel well areas, use ground maintenance locks on MLG actuators and pin the nose gear down to prevent inadvertant retraction. Failure to do this could result in serious injury or death.

1. Place airplane on jacks and raise enough to permit gear retraction. Refer to chapter 7.

WARNING

Verify that both normal and auxiliary hydraulic pressure have depelted to 0 psi.

2. Shut off hydraulic pressure and remove electrical power from airplane.
3. Deplete normal and auxiliary hydraulic systems.
4. Remove both nosewheel tire and wheel assemblies.

CAUTION

If wheels are not removed, damage to the piston ring will occur.

5. Remove strut air filler valve dust cap. Slowly loosen filler valve 3/4-inch hexnut enough to allow air to escape from shock strut. Partially compress strut to release any trapped air.

CAUTION

To prevent possible damage to the retract cylinders, do not move the landing gear by any means other than with hydraulic power.

6. Remove landing gear ground safety locks. Then, position the LDG GEAR handle to UP.
7. Apply electrical power and slowly apply hydraulic pressure to the airplane until nose landing gear strut is approximately 30 degrees from the full down position.

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MAINTENANCE PRACTICES (Continued)

REMOVAL OF NOSE GEAR STRUT INNER PISTON (Cont'd)

8. Using chocks or other suitable stops, support nose landing gear strut approximately 30 degrees from full down. Then, remove electrical power and hydraulic pressure from airplane. Deplete normal and auxiliary accumulator pressure.
9. Disconnect nose gear scissors and secure upper torque arm to upper part of strut. Then, remove gland nut locking screws.
10. Remove balance weight assembly.
11. Place a suitable container under shock strut to catch hydraulic fluid contained in strut.
12. Remove gland nut with spanner wrench and then carefully slide inner piston from strut barrel.

NOTES

- ***If all packings (O-rings) and felt washer are to be replaced, perform the remaining steps in this procedure.***
- ***Note the position of upper and lower bearing assemblies. The bearing assemblies incorporate the centering cams and can be installed 180 degrees out of phase during the assembly procedures.***

13. Remove upper bearing from inner piston.
14. Remove upper bearing key and ring from groove of inner piston.
15. Remove lower bearing and gland nut from inner piston.
16. Remove felt and scraper ring retaining ring from gland groove of lower bearing assembly. Then, remove felt and scraper ring.
17. Remove retaining ring (2010792-1) and washer (2010788-1) and shim washer (2010790-1).
18. Remove internal and external packings and packing back-up rings from lower bearings.

NOTE

The orifice tube and metering pin packings (O-rings) need not be replaced unless leakage is evident or suspected in these areas. If these packings are to be replaced, continue with step 19. Otherwise, continue with step 23.

19. Remove bolt securing jack pad to bottom of piston and then remove pad.
20. Remove nut (7/8-inch socket, extension and ratchet) and washers (2) retaining metering pin to inner piston.
21. Push out metering pin from bottom of inner piston with fabricated push-out tool (9/16-inch wood dowel).
22. Remove packing and packing back-up rings from base of metering pin.
23. Remove strut filler valve from top of strut.
24. Remove lock key and nut retaining orifice tube to strut barrel assembly. Then, remove orifice tube.
25. Remove piston ring from bottom of orifice tube and packing and packing backup rings from top of tube.
26. Clean all parts with cleaning solvent. Dry all parts thoroughly with a clean, lint-free cloth and/or compressed air.
27. Check all parts for damage and wear. Replace damaged or worn parts with serviceable items.

INSTALLATION OF NOSE GEAR STRUT INNER PISTON

NOTES

- ***During assembly and installation of nose gear strut inner piston, ensure that all O-rings and/or packings are serviceable, and that all parts are thoroughly cleaned and lubricated with hydraulic fluid MIL-H-5606.***
- ***If metering pin and orifice tube were removed, perform steps 1 through 8.***

1. Install metering pin using metering pin installation tool from the top of inner piston and align and install washer (with pin) into metering pin base, through bottom of piston.
2. Install washer and new lock-nut on metering pin through bottom of piston. Torque nut to 15 inch-pounds (maximum).
3. Install packing, with packing backup ring on each side, in seal groove at top of orifice tube.
4. Install piston ring on bottom of orifice tube.
5. Install orifice tube in strut barrel assembly so that threaded end of tube passes through hole in top of barrel. Rotate orifice tube so that keyway is aligned with slot in top of barrel.
6. Place key (with safetying hole up) inside nut and tighten nut snugly on exposed threaded end of orifice tube.

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MAINTENANCE PRACTICES (Continued)

INSTALLATION OF NOSE GEAR STRUT INNER PISTON (Cont'd)

7. Install strut filler valve in nut and lockwire.
8. Install steering collar, thrust washer P/N 2010778-1, torque link collar, second thrust washer P/N 2010778-1 and No. 2 retainer ring on sleeve assembly. Do not install shim washer P/N 2010790-1 at this time.
9. Remove upper bearing retaining ring P/N 2010791-1, upper bearing retaining nut P/N 3010762-1, upper bearing and upper bearing key P/N 2010767-1 from inner piston. Remove gasket P/N AN6230B10 from exterior of lower bearing. See Figure 32-20-403.

CAUTION

The alignment of the gland nut screw holes with the slots in the lower sleeve assembly as described in steps 3 and 4 is critical to proper rigging of the nose landing gear.

10. Insert lower bearing in base of barrel assembly and install gland nut. Torque gland nut to 15-25 foot-pounds. If slots in lower sleeve assembly are not aligned with screw holes in gland nut, back off gland nut until first set of two slots in lower sleeve assembly are aligned with threaded screw holes in gland nut. Verify that there is no vertical movement of lower bearing.
11. Using tape, ink marker or other suitable means, mark a spot on the upper portion of the sleeve assembly located directly above the gland nut locking screws. The object of this is to identify the two slots in the lower sleeve assembly that are now aligned with the gland nut screw holes. The alignment of these two particular slots with the gland nut screw holes after final assembly is critical to proper rigging of the nose landing gear.
12. Measure and record clearance (measured at point of minimum clearance) between thrust washer P/N 2010778-1 and gland nut.
13. Using clearance obtained in steps 5, add or remove laminations from shim washer P/N 2010790-1 as necessary to obtain a 0.001 to 0.005 inch clearance between shim washer and gland nut after shim washer is installed on sleeve assembly. See Figure 32-20-405.
14. Remove gland nut and lower bearing. Remove retainer ring No. 2, add shim washer, then reinstall retainer ring. Reinstall lower bearing and gland nut. Torque gland nut to 15-25 foot-pounds and ensure that screw holes align with same slots in lower sleeve assembly as identified in step 10.
15. Measure clearance between shim washer P/N 2010790-1 and gland nut to verify a 0.001 to 0.005 inch clearance. (See Figure 32-20-405.) Also verify that there is no vertical movement of the lower bearing. If both of these requirements are met, then proceed to step 16, otherwise, remove shim washer and repeat steps 10 thru 15 as necessary to achieve them.

NOTE

After the proper clearance requirements are achieved, a thrust washer P/N 2010778-1 machined to the same thickness as the prepared shim washer may be voluntarily substituted in place of the shim washer.

16. Remove gland nut and install on inner piston.
17. Replace the following consumable items on the lower bearing:
    A. Two external teflon packing retaining rings P/N MS28774-232.
    B. Two internal teflon packing retaining rings P/N MS28774-332.
    C. One external gasket P/N AN6230B10.
    D. One internal packing P/N AN6227B35.
    E. One felt washer P/N MS28932-1 (cut to length).
    F. One scraper ring P/N MS28776-24.

CAUTION

It is possible to install the upper and lower bearings (which incorporate centering cams) 180° out-of-phase. This condition would prevent full extension, center locking, and proper retraction of the strut.

18. Align keyway in upper bearing with key on inner piston and then slide bearing on piston.
19. Install and tighten upper bearing retaining nut. Install lock ring.
20. Install washer (2010788-1) on top face gland nut.
21. Slide inner piston assembly into bore of barrel assembly. Align the two keyways in lower bearing flange with two keys in sleeve assembly. Slide inner piston up to point at which lower bearing is seated firmly. Start gland nut on sleeve assembly. Using spanner wrench, torque gland nut to 15 to 25 foot-pounds.

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MAINTENANCE PRACTICES (Continued)

INSTALLATION OF NOSE GEAR STRUT INNER PISTON (Cont'd)

22. If slots in lower sleeve assembly are not aligned with screw holes in gland nut, back off gland nut until first set of two slots in lower sleeve assembly are aligned with threaded screw holes in gland nut.
23. Install lower bearing packings and packing retaining rings as follows:
    A. Install packing with packing retaining ring on each side in external seal groove. (See Figure 32-20-403.)
    B. Coat external lower bearing with petrolatum (Vaseline) (VV-P-236) or equivalent.
    C. Soak external packing and retainer rings in hydraulic fluid (MIL-H-5606A).

CAUTION

Do not cut packing retaining rings as leaks will occur.

    D. Install packing with packing retaining ring on each side in external seal groove.
    E. Install a three inch hose clamp around packing retaining rings as some deformation may have occurred during installation. Tighten hose clamp finger tight.
    F. Remove hose clamp after packing retaining rings have returned to shape.
    G. Remove all petrolatum from lower bearing surface.
24. Install ring (2010791-1) near top of inner piston.
25. Install key (2010767-1) in keyway at top of inner piston.

CAUTION

It is possible to install the upper and lower bearings (which incorporate centering cams) 180° out-of-phase. This condition would prevent full extension, center locking, and proper retraction of the strut.

26. Align keyway in upper bearing with key on inner piston and then slide bearing on piston.
27. Install and tighten upper bearing retaining nut. Install lock ring.
28. Install washer (2010788-1) and adjust shim washer 2010790-1 or 2010788-1 modified per Figure 32-20-403.
29. Install gland nut lockscrews and washers. Safety with lockwire.

NOTE

Lockscrews should be installed in the same locking slots as previously attained in step 22.

30. Install felt washer (cut to length), scraper ring, and retaining ring groove of lower bearing assembly.

CAUTION

Torque arms (scissors) must be free to rotate at apex pivot. Do not over tighten nut.

31. Connect and safety the nose gear scissors as follows:
    A. Place AN960C616 washer between torque arms at apex.
    B. Install bolt (P/N 2010764-1).
    C. Install washer (P/N 2010765-1) and then washer (P/N AN960-416L).
    D. Install nut (P/N AN310-4).
    E. Tighten nut to obtain a clearance of 0.001 to 0.002 inches between torque arms at apex pivot.
    F. Safety nut with cotter pin.
32. Install nose gear balance weight to bracket. Torque screws (2) to 20 to 25 inch-pounds. Then, install balance weight assembly on nose gear. Ensure that balance weight bracket is centered on nose gear uplock fitting.

NOTE

When installing balance weight and torquing attach screws (2), axle must be torqued prior to torquing the attach screws.

CAUTION

To prevent possible damage to retract cylinders, do not move landing gear by any means other than with hydraulic power.

33. Extend landing gear and install ground safety pins.
34. Service nose landing gear shock strut (refer to chapter 12). Then, check strut for leaks.
35. Remove ground safety pins and perform operational check of landing gear to ensure proper nose gear centering, retraction, and shock strut extension.

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MAINTENANCE PRACTICES (Continued)

ADJUSTING CLEARANCE BETWEEN NOSE GEAR SHOCK STRUT STEERING COLLAR AND GLAND NUT

1. Remove nose landing gear strut inner piston. (Refer to "Removal of Nose Gear Shock Strut Inner Piston" this section.)
2. Inspect the following items for corrosion and/or excessive wear:
   A. Steering Collar I.D. 3.920 to 3.922 inches
   B. Steering Sleeve O.D. 3.9138 to 3.9162 inches
   C. Gland nut
   D. Torque link collar
   E. Laminated shim washer
   F. Thrust collar
   G. Mating bearing surfaces
3. Coat the portion of barrel assembly underlying steering collar sleeve with corrosion preventive compound, MIL-C-16173, Grade 1.
4. Coat the lower two inches of steering collar sleeve with MIL-C-16173, Grade 1 and re-install. Reseal top of sleeve with Products Research Sealant 1436G-B-2, No. MMI-30-57. Remove excess sealant.
5. Coat the O.D. of sleeve with MIL-G-23827 lubricant and I.D. of the steering collar, torque link collar, and thrust washers as each item is installed.

NOTE

A single laminated shim washer of proper thickness should be used to assure the correct adjustment for axial clearance (0.001 to 0.005 inch) between steering collar and gland nut. Shim laminations must be intact (not torn, damaged, or partially delaminated). The use of multiple shim sections to "Fine Tune" the adjustment is not recommended. The use of a solid washer (2010788-1) eliminates the possibility of partially delaminated or torn shims.

6. Install washer (2010788-1), lower bearing, and gland nut on lower end of barrel assembly and torque gland nut to 15 to 25 foot-pounds.
7. Back off gland nut until first set of two slots in lower end of barrel assembly are aligned with threaded screw holes in gland nut.
8. Check for 0.001 to 0.005 inch clearance between steering collar and gland nut (measured at point of minimum clearance). If clearance is not as specified, see Figure 32-20-405.
9. Remove gland nut, lower bearing, and washer.
10. Using clearance obtained in step 8, remove laminations from shim washer as necessary to provide a 0.001 to 0.005 inch clearance between washer (2010788-1) and steering collar when final installation of gland nut, shim washer and washer (2010788-1) is accomplished.

NOTES

- ***An alternate procedure is to replace the shim washer (2010790-1) with a second washer (2010788-1) machined to clearance obtained in step 8. Verify 0.001 to 0.005 inch clearance.***
- ***If the noted clearance is excessive, a new shim washer should be used in order to obtain the proper clearance. If a new shim washer was installed to obtain the proper clearance, the nose gear uplock clearance should be checked. The addition of a new shim may vary the uplock roller slightly which in turn will change the uplock clearance.***

11. Install the nose landing gear inner piston. (Refer to "Installation of Nose Gear Strut Inner Piston" this section.)
12. Lubricate steering collar and torque link collar fittings until grease extrudes from all bearings and washer surfaces.
13. Rotate nose wheel full left and right to distribute the lubricant uniformly.

REPLACEMENT OF NOSE GEAR ACTUATING CYLINDER

Removal:

1. Place airplane on jacks and raise enough to permit gear retraction. Refer to chapter 7.

WARNING

Verify that both normal and auxiliary hydraulic pressure have been depleted to 0 psi.

2. Shut off hydraulic pressure and remove electrical power from airplane.
3. Deplete normal and auxiliary hydraulic systems.
4. Disconnect and cap actuating cylinder hydraulic lines.
5. Disconnect actuating cylinder at nose gear shock strut arm.
6. Disconnect actuating cylinder at nose gear drag brace attach point and remove actuating cylinder.
7. Remove hydraulic lines and restrictor from actuating cylinder.

Installation and Adjustment of Nose Gear Actuating Cylinder

1. Place airplane on jacks and raise enough to permit gear retraction. Refer to chapter 7.

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MAINTENANCE PRACTICES (Continued)

Installation and Adjustment of Nose Gear Actuating Cylinder (Cont'd)

2. Connect trunnion end of actuating cylinder to drag brace. Install attach bolt with bolthead outboard. Install and tighten attach bolt nut and washer.

CAUTION

When installing a new actuator, and prior to connecting the actuator to the strut (connected in strut well only) position the actuator so that rod end will clear adjacent structure. Hydraulically extend and retract (slowly) the actuator several cycles to eliminate any trapped air. To prevent damage to the system, use a low hydraulic pressure during the cycling.

3. With nose gear down and locked. Fully bottom actuating cylinder piston in actuating cylinder. Adjust actuating cylinder rod end to obtain a slip-fit of rod end attach bolt to attach point on nose gear shock strut. Then shorten rod end one and one-half turn. Lock rod end jam nut.

NOTE

This is the initial adjustment of the rod end. The rod end will be re-adjusted in subsequent step.

4. Remove upstop bumper.

WARNING

Before applying landing gear hydraulic pressure for checks, adjustments, etc., deplete normal and auxiliary accumulator pressure. This will prevent the auxiliary accumulator power assist retraction system from operating, and prevent serious injury to personnel and/or damage to the airplane.

NOTE

This rigging must be accomplished at 1800 psi by installing the main gear ground safety locks and maintaining 1800 psi using the airplane hydraulic pump or a hydraulic test stand.

5. Install main gear ground safety locks.
6. Slowly increase hydraulic pressure to 1800 psi while retracting the nose gear only to the locked position.
7. If necessary, loosen rod end until roller makes contact with hook.
8. Inspect area for contact of strut components, tires, or wheels with other items in the wheel well. If contact is made with roller contacting hook, the item making contact with strut component should be moved or cleared of nose gear.

NOTE

The maximum dimension from the rod end to the cylinder is 1-17/32 in. (See Figure 32-20-412.) If this dimension cannot be met, check length of retract cylinder assembly (see Figure 32-10-411).

9. Check for a 1/8 (+1/32, -0) inch clearance between top of nose gear uplock hook and end of nose gear uplock hook actuating cylinder piston (hydraulic pressure applied). If clearance is not within tolerance, extend nose gear and adjust nose gear uplock roller on serations to obtain required clearance.
10. Reinstall upstop bumber.
11. On S/N 370-1 thru -9 and 380-1 thru -27 not changed by S/B 74-20, adjust nose gear bumper stop to just contact nose gear strut. Extend nose gear. Then, extend nose gear bumper stop one half turn. Tighten and lockwire stop jam nut.
12. On S/N 380-28 and subs, and airplanes changed by S/B 74-20, adjust nose gear ice breaker bolt to just contact rubber pad bonded to nose gear strut with hydraulic pressure on. Extend nose gear. Then, extend ice breaker bolt three turns. Tighten and lockwire stop jam nut.
13. Retract nose gear to position nose gear strut between tips of nose gear fairing door operating cams. Support nose gear in this position.
14. Adjust door-open stop bolts so that nose gear strut is centered between tips of door operating cams. Then, adjust stop bolt on one side to provide a 1/32 (+1/32, -0) inch clearance between head of stop bolt and nose wheel well structure. Tighten stop jam nuts.
15. Disconnect one side of nose wheel fairing door. Retract nose gear. Check that cable attached to wheel fairing door lock hook is adjusted to slip-fit at attach point to door lock hook. This should be carefully checked because if the cable is too tight, the hook will be partially pulled from under the door lock roller. If the cable is too loose, movement of the nose gear uplock hook to unlocked position will not pull the hook clear of the door lock rollers, causing the gear and doors to hang up on the extension cycle.
16. Extend landing gear and connect wheel fairing door link. Check gear installation for all safeties.

NOTE

When installing the balance weight and torquing attach screws (2), the axle must be torqued prior to torquing the attach screws.

17. Install nose gear balance weight to bracket. Torque screws (2) to 20-25 inch-pounds. Then, install balance weight assembly on nose gear. Ensure that balance weight bracket is centered on nose gear uplock fitting.

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MAINTENANCE PRACTICES (Continued)

Installation and Adjustment of Nose Gear Actuating Cylinder (Cont'd)

18. Check normal and emergency operation of landing gear system. Check operation of landing gear system. Check operation of landing gear position indicating and warning system.
19. Lower airplane and remove jacks.

REPLACEMENT OF NOSE GEAR FAIRING DOORS

Removal:

WARNING

Before applying landing gear hydraulic and auxiliary accumulator pressure for check, adjustments, etc., deplete normal and auxiliary accumulator pressure. This will prevent the auxiliary accumulator power assist retraction system from operating and will prevent serious injury to personnel and/or damage to the airplane.

WARNING

When working in or around wheel well areas, use ground maintenance locks on MLG actuators and pin the nose gear down to prevent inadvertant retraction. Failure to do this could result in serious injury or death.

1. Install nose gear ground safety pin.
2. Place airplane on jacks and raise enough to permit gear retraction. Refer to chapter 7.

WARNING

Verify that both normal and auxiliary hydraulic pressure have depleted to 0 psi.

3. Shut off hydraulic pressure and remove electrical power from airplane.
4. Deplete normal and auxiliary hydraulic systems.
5. Remove access doors.
6. Disconnect nose gear fairing door actuating link at fairing door attach point.
7. Remove fairing door forward and aft hinge attach bolts, and remove nose gear fairing door.

Installation and Adjustment of the Nose Gear Fairing Door

1. Position nose gear fairing door in wheel well area. Install forward hinge attach bolts as follows:
   A. Insert spacer in forward door hinge.
   B. Position two 2W1AL-21-20-91 washers between hinge and hinge support fitting (one washer on each side of forward door hinge).
   C. Position 2W18-10M washer under head of attach bolt.
   D. Insert bolt through hinge and hinge support fitting with bolthead facing aft.
   E. Tighten hinge attach bolt.
2. Install aft hinge attach bolt as follows:
   A. Insert spacer in aft door hinge.
   B. Insert bolt through hinge and hinge support fitting with bolthead facing forward.
   C. Install 2W18-10M washer and nut on attach bolt. Tighten hinge attach bolt.

WARNING

Do not connect nose gear fairing actuating link to fairing doors.

3. Remove nose gear ground safety pin.
4. Slowly increase hydraulic pressure and retract landing gear. Release hydraulic pressure.
5. With nose gear hanging on uplock hook, remove pressure from hydraulic systems.
6. Remove door uplock roller. Push fairing door closed. Hold door closed and mark edges of door where trim is needed to maintain 1/32- to 3/32-inch clearance around edge of door. Open door and trim as required.
7. Push door fully closed. Check that outboard edge of door is flush with fuselage skin. If door does not fair with fuselage skin, open door and adjust forward and/or aft hinges on serrations (at hinge attach point to door). Tighten hinge attach bolts.
8. Check total end play of nose gear fairing door uplock hook assembly. End play should be 0.005 to 0.040 inch. If end play is not within tolerance, remove uplock hook and add or remove washers between door uplock hook and support to obtain required end play. The number of washers used on either side must be equal to, or one more than, the washers used on the opposite side. Install and safety door uplock hooks.
9. Install door uplock roller. Push door to closed and locked position. Check that there is a 1/16 inch clearance between spacer and 265-343331 door hook support fitting. To obtain 1/16 inch clearance, open fairing door. Move 265-343331 door hook support fitting forward or aft by interchanging washer between fuselage structure and door hook support fitting and washers under door hook support fitting attach bolt nuts.

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MAINTENANCE PRACTICES (Continued)

Installation and Adjustment of the Nose Gear Fairing Door (Cont'd)

10. Close and lock gear fairing door. Again check clearance as directed in step 7.
11. With door up and locked, check that forward side of door is flush with fuselage skin. If door is not flush with fuselage skin line, open door and move door uplock roller up or down on serrations. Tighten door uplock roller support attach bolts and again check fit of door.
12. With nose gear up and locked and door uplock hook fully engaged under door uplock roller, adjust door lock release cable at door hook to a slip fit of cable to attach point on door hook. Install and safety cable attaching pin. Tighten cable jam nut.
13. Close and lock right hand nose gear fairing door.
14. Adjust length of actuating link until aft inboard corner of door just contacts door land. Then, shorten five full turns of actuating link rod end. Tighten rod-end jam nut.
15. Open right hand nose gear fairing door.
16. Close and lock left hand nose gear fairing door.
17. Adjust length of actuating link until aft inboard corner of door just contacts door land. Tighten rod-end jam nut.
18. Open left hand nose gear fairing door.
19. Extend landing gear. With nose gear down and locked, connect left and right nose gear fairing door actuating cylinders and safety.
20. Retract nose gear and close fairing doors. Re-verify a 1/16- to 3/16-inch clearance between fairing doors. If clearance is not within tolerance, extend nose gear and trim inboard edge of door to meet required clearance. Retract landing gear and recheck clearance.
21. Check operation of landing gear normal and emergency extension.
22. Check door and lock linkage, and that all safeties are installed.
23. Lower airplane and remove from jacks.

REPLACEMENT OF NOSE LANDING GEAR APEX BUSHINGS IN THE TORQUE ARMS (SCISSORS)

NOTE

The parts referred to in the following procedure are identitied in Figure 32-20-409.

1. Remove torque arms from nose landing gear.
2. Press defective bushings out of the torque arms using an arbor press and an appropriate adapter.

NOTE

Bushings must be replaced in pairs.

3. Clean bore and inspect for scratches, corrosion, out-of-round, or other defects.
4. Install bushings as follows:
   A. Place torque arm on bed or arbor press.
   B. Coat outside of bushing with zinc chromate primer.
   C. Place bushing and bushing installation tool in position on torque arm.
   D. Carefully press bushing into bore while primer is still wet.
5. Install the torque arm assembly in the mill holding fixture. Mill the flange of bushings to dimensions shown in Figure 32-20-409.
6. Install the torque arm assembly in the ream holding fixture. Ream the bushings to the dimensions shown in Figure 32-20-409.

CAUTION

Torque arms (scissors) must be free to rotate at apex pivot. Do not over tighten nut.

7. Grease roller bearings (two in each torque arm mounting base boss). Refer to chapter 12.
8. Install upper and lower torque arm bases to strut with bolt washer and castellated nut. The upper bolt head facing right hand and lower bolt head facing left hand. Torque nuts to 12 to 15 inch-pounds and cotter pin both nuts.
9. Connect torque arms at apex pivot as follows:
   A. Place AN960C616 washer between torque arms at apex.
   B. Install bolt P/N 2010764-1.
   C. Install washer P/N 2010765-1 and then washer P/N AN960-416L.
   D. Install nut P/N AN310-4.
   E. Tighten nut to obtain a clearance of 0.001 to 0.002 inch between torque arms at apex pivot.
   F. Safety nut with Cotter pin.
10. Grease apex pivot fittings. Refer to chapter 12.

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MAINTENANCE PRACTICES (Continued)

REPLACEMENT OF NOSE WHEEL DOOR CAM ASSEMBLY

The parts referred to in the following procedures are identified in Figure 32-20-410.

Removal:

WARNING

Before applying landing gear hydraulic pressure for any reason (checks, adjustments, etc.), deplete normal and auxiliary accumulator pressures. This will prevent the auxiliary accumulator power assist retraction system from operating and prevent serious injury to personnel and/or damage to the airplane.

1. Remove electrical power from the airplane and deplete all hydraulic pressure. Disconnect left and right door rod assemblies at left and right wheel door actuator arms by removing one bolt, washer, and nut from each side. Allow rods to hang free.
2. Remove two bolts, bushings, and nuts from left and right wheel door actuating arm assemblies.
3. Remove complete cam assembly containing both cams as one unit.
4. Remove helical extension spring, one bolt, two washers, one spacer, one nut and cotter pin from center of cam assembly below removed spring. Cam assembly will now separate into two parts.
5. From each cam, remove the cotter pin, spacer, pin, and three washers. Also three lube fittings if required on replacement cam.

Installation:

(See Figure 32-20-410.)

NOTE

Replace all removed cotter pins and retain all other removed hardware for reinstallation.

1. On each cam, install three lube fittings (if required), pin, three washers, and a new cotter pin.
2. Align center bolt holes in cams and install one bolt, two washers, one spacer, and one nut. Tighten nut and install cotter pin. Install helical extension spring by expanding spring slightly and installing each loop end over exposed spacers between washers on the upper backside of each cam.
3. With both cams assembled, and as one unit, lift and position assembly into nose wheel well, and align bolt holes in left and right cam assembly with bolt holes in left and right wheel door actuator arms. Install two bolts, bushings, and nuts on each side and tighten nuts. Do not install left and right door rods at this time.
4. Grease and lube assembly. (See chapter 12.)

WARNING

Before applying landing gear hydraulic pressure for checks, adjustments, etc., deplete normal and auxiliary accumulator pressure. This will prevent the auxiliary accumulator power assist retraction system from operating and prevent serious injury to personnel and/or damage to the airplane.

5. Position jacks under the airplane, install jacking fixtures, and lift the airplane high enough to swing gears. Remove ground safety locks from main gear fairing doors, nose and main gears. Provide hydraulic pressure source and 28 VDC power. Put LDG GEAR handle in the UP position. Clear all personnel and equipment from wheel wells and slowly apply hydraulic pressure. When nose gear strut is centered between lower edges of cams, stop movement of gear and support nose gear in this position.
6. Accomplish rigging procedures outlined in "Installation and Adjustment of Nose Gear Shock Strut Assembly" in this section.
7. Remove support from under nose gear. Apply 28 VDC electrical power and hydraulic pressure, and slowly complete cycling of landing gear. Lower gears. Install one bolt, washer, and nut to connect left and right door rod, and tighten nut. Complete all phases of landing gear operations and emergency landing gear operations.
8. Turn off hydraulic source, deplete all hydraulic pressure, remove electrical power, and install gear down and fairing door safety locks.
9. Lower airplane and remove jacks and jacking fixtures.

CLEARANCE CHECK OF NOSE LANDING GEAR STRUT STEERING COLLAR

NOTE

DELETED.

1. Ensure that the steering collar is tight against the shock strut. Then, check the clearance between the nose gear strut gland nut and 2010788-1 washer.

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CLEARANCE CHECK OF NOSE LANDING GEAR STRUT STEERING COLLAR (Cont'd)

2. If clearance is not between 0.001 and 0.005 inches, refer to "Clearance Adjustment Between Nose Gear Shock Strut Steering Collar and Gland Nut" in this section.

NOTE

If a new shim was installed to obtain the proper clearance, the nose gear uplock clearance should be checked. The addition of a new shim may vary the uplock roller slightly which in turn will change the uplock clearance. Refer to "Replacement of Nose Gear Shock Strut Assembly" in this section.

REMOVAL AND INSTALLATION OF NOSE GEAR SLEEVE FROM BARREL ASSEMBLY FOR CORROSION INSPECTION

Removal:

(See Figure 32-20-406.)

WARNING

Before applying landing gear hydraulic pressure for checks, adjustments, etc., deplete normal and auxiliary accumulator pressure. This will prevent the auxiliary accumulator power assist retraction system from operating and prevent serious injury to personnel and/or damage to the airplane.

WARNING

When working in or around wheel well areas, use ground maintenance locks on MLG actuators and pin the nose gear down to prevent inadvertant retraction. Failure to do this could result in serious injury or death.

1. Place airplane on jacks and raise enough to permit gear retraction. Refer to chapter 7.

WARNING

Verify that both normal and auxiliary hydraulic pressure have depleted to 0 psi.

2. Shut off hydraulic pressure and remove electrical power from airplane.
3. Deplete normal and auxiliary hydraulic systems.
4. Install ground maintenance safety locks.
5. Remove strut air filler valve dust cap. Slowly loosen filler valve 3/4 inch hexnut enough to allow air to escape from shock strut. Partially compress strut to release any trapped air.

CAUTION

To prevent possible damage to retract cylinders, do not move landing gear by any means other than with hydraulic power.

6. Remove landing gear ground safety locks. Then, position the LDG GEAR handle to UP.
7. Apply electrical power and slowly apply hydraulic pressure to the airplane until the nose landing gear strut is at approximately a 30-degree angle from the full down position.
8. Using chocks or other suitable stops, support the nose landing gear strut at approximately 30-degrees from the full down position. Remove electrical power and hydraulic pressure from airplane. Deplete normal and auxiliary accumulator pressure.

NOTE

Remove nose wheels.

9. Disconnect nose gear scissors and secure upper torque arm to upper part of strut. Then, remove gland nut locking screws.
10. Place a suitable container under shock strut to catch hydraulic fluid contained in strut.
11. Remove gland nut with spanner wrench and then carefully slide inner piston from strut barrel.
12. Disconnect steering link from steering collar.
13. Remove washer and shim washer.
14. Remove retainer ring No. 1 from sleeve.
15. Remove collar assembly, washer, and steering collar from sleeve.
16. Using a suitable pointed tool, carefully remove sealing compound from cavity between barrel and sleeve assembly. Use caution to prevent damage to barrel and sleeve assembly.
17. Tap the bottom of sleeve with a rubber or wooden mallet upwards to about 1/4 inch.
18. Carefully remove retainer ring No. 2 from sleeve by compressing ring thru both elongated holes and move ring upwards over ramp.
19. Slide sleeve from barrel.
20. Check sleeve and barrel for corrosion.

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MAINTENANCE PRACTICES (Continued)

Installation:

(See Figure 32-20-406.)

NOTE

Thoroughly clean sleeve and barrel assembly with MEK (TT-M 261 or equivalent) and dry parts.

1. Install retainer ring No. 2 inside sleeve to the first ridge.

NOTES

- ***Be sure retainer ring ends are not located at either elongated holes in sleeve (retainer ring ends should be about 90 degrees from elongated holes). This will ensure easy removal of retainer ring at the next removal of sleeve.***
- ***Coat outer barrel between bearing surfaces with MIL-C-16173D (Grade 1) Corrosion Preventative Compound.***

2. Slide sleeve onto barrel assembly.
3. Position retainer ring No. 2 into its groove by compressing the ring through both elongated holes in sleeve and move ring downwards over ramp.

NOTE

To ensure proper installation of retainer ring, check the following:

A. Visually inspect thru both elongated holes in sleeve.

B. Both keys are bottomed out in keyway with approximately 1/4 inch extending from bottom of upper barrel.

4. Install steering collar over sleeve, and then washer (P/N 2010788-1).
5. Install collar assembly and retainer ring No. 1 on sleeve.
6. Install shim washer (2010790-1) and washer (2010788-1) on top face gland nut.
7. Slide inner piston assembly into bore of barrel assembly. Align the two keyways in lower bearing flange with two keys in sleeve assembly. Slide inner piston up to point at which lower bearing is seated firmly. Start gland nut on sleeve assembly. Using spanner wrench, torque gland nut to 15 to 25 foot-pounds.
8. If slots in lower sleeve assembly are not aligned with screw holes in gland nut, back off gland nut until first set of two slots in lower sleeve assembly are aligned with threaded screw holes in gland nut.
9. Check for 0.001 to 0.005 inch clearance between steering collar and gland nut (measured at point of minimum clearance). If clearance is not as specified, see "Adjustment of Clearance Between Nose Gear Shock Strut Steering Collar and Gland Nut" in this section.
10. Install gland nut lockscrews and washers. Safety with lockwire.

CAUTION

Torque arms (scissors) must be free to rotate at apex pivot. Do not over tighten nut.

11. Connect and safety nose gear scissors as follows:
    - A. Place AN960C616 washer between torque arms at apex.
    - B. Install bolt (P/N 2010764-1).
    - C. Install washer (P/N 2010765-1) and then washer (P/N AN960-416L).
    - D. Install nut (P/N AN310-4).
    - E. Tighten nut to obtain a clearance of 0.001 to 0.002 inch between torque arms at apex pivot.
    - F. Safety nut with cotter pin.
12. Apply sealant between sleeve and barrel (see Figure 32-20-406.)

CAUTION

To prevent possible damage to retract cylinders, do not move landing gear by any means other than with hydraulic power.

13. Extend landing gear and install ground safety pins.
14. Service the nose landing gear shock strut (refer to chapter 12). Then, check strut for leaks.
15. Lower airplane and remove jacks.
16. Remove ground safety pins and perform operational check of landing gear to ensure proper nose gear centering, retraction, and shock strut extension.

ADJUSTMENT OF NOSE GEAR STEERING BELLCRANK BUSHINGS

CAUTION

Do not engage steering system as airplane damage could result.

NOTES

- ***Airplanes modified by SOK No. 24 are equipped with 45° nose wheel steering and will have a clearance of 0.002 minimum to 0.013 maximum.***
- ***Retain all hardware removed for reinstallation.***

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MAINTENANCE PRACTICES (Continued)

ADJUSTMENT OF NOSE GEAR STEERING BELLCRANK BUSHINGS (Cont'd)

1. Remove steering linkage arm.
2. Remove steering cylinder bellcrank.
3. Remove bellcrank indicator bracket.
4. Measure the face distance of the installed bellcrank bushings to determine proper dimensions of 1.750 ±0.003 inch exists. (See Figure 32-20-411.)

NOTE

Use a suitable tool to remove the bushings being careful not to damage surrounding material.

5. Remove and replace bushings:
   A. If dimensions in step 4 are not met.
   B. If bushings are excessively worn.
   C. Or machine to proper dimensions if oversized.
6. Coat outside diameter of new bushings with zinc chromate primer, MIL-P-8585.
7. Install new bushings in sets.
8. After installation, line ream inside diameter to 1.1870 inch minimum to 1.1880 inch maximum (see Figure 32-20-411).

NOTE

Lube bellcrank splines (only) with Lubricant CP-28 (E/M Lubricants), or with MolyCote G (Dow Corning).

9. Install steering cylinder bellcrank. Torque bolt to 290 to 410 inch-pounds.
10. Install bellcrank indicator bracket. Torque bolt to 10 inch-pounds.
11. Install steering linkage arm. Torque bolt to 10 inch-pounds.
12. Perform a functional check of the steering system.

INSPECTION OF NOSE LANDING GEAR SHOCK STRUT BELLCRANK P/N 3010812 (See Figure 32-20-407)

1. Remove bolt attaching nose steering link to arm.
2. Remove bolt attaching nose steering bellcrank to nose gear steering cylinder rod end.
3. Remove bolt attaching nose steering arm to bellcrank.
4. Loosen nose steering limit switch bracket bolt.
5. Remove bellcrank from nose gear shock strut.
6. Perform a magnetic particle (Magna-Glo) inspection of the nose gear steering bellcrank for cracks.

NOTE

If there is evidence of excessive bellcrank shaft or bushings radial wear, perform step 7.

7. Measure bellcrank shaft outside diameter. It must be within 1.1850 to 1.1860 inches. Measure bushings inside diameter. It must be within 1.1870 to 1.1895 inches.
8. Install serviceable bellcrank into nose gear shock strut. Install arm on bellcrank. Ensure that index mark (dot) on end of bellcrank is aligned with index mark on arm.
9. Install arm to bellcrank attach bolt. Torque bolt 290 to 410 inch-pounds, and safety with lockwire.
10. Install link to arm attach bolt. Tighten nut to 10 inch-pounds and install cotter key.
11. Position limit switch actuating arm bracket as necessary.
12. Install bolt attaching nose gear steering cylinder rod end to bellcrank. On aircraft where a self locking nut is used, tighten lock nut snugly until all end play between bolt head and nut is removed. Do not apply load to bellcrank clevis. On aircraft using a castellated nut, tighten to 10 inch-pounds and install cotter pin (on aircraft changed by S/L 19 or SOK 24).
13. Tighten limit switch supporting racket attach bolt.
14. Check for proper installation and safeties.
15. Perform operational checkout of nose wheel steering system.

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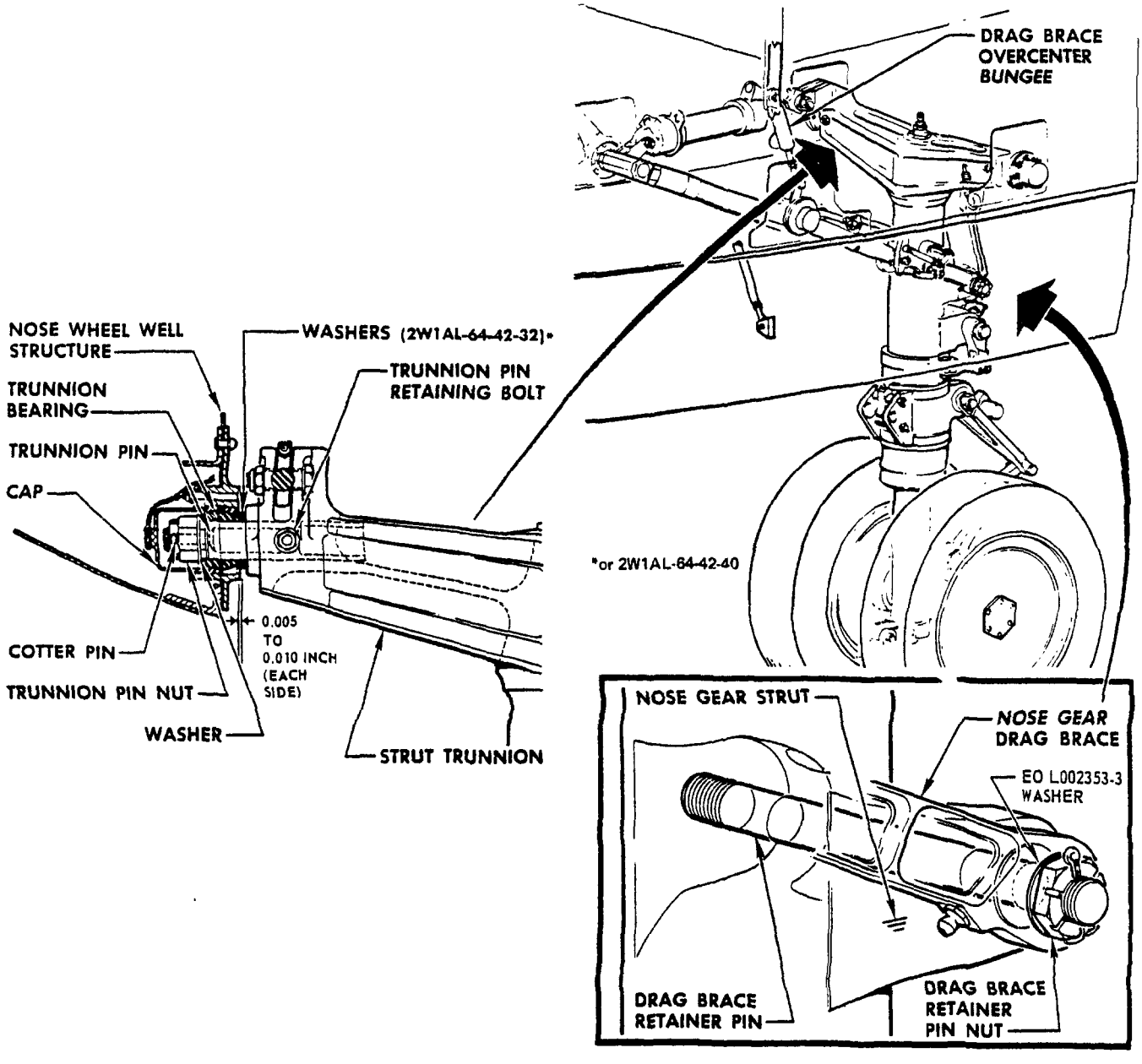

Figure 32-20-401 — Installing and Adjusting Nose Gear Shock Strut Assembly (Sheet 1 of 3)

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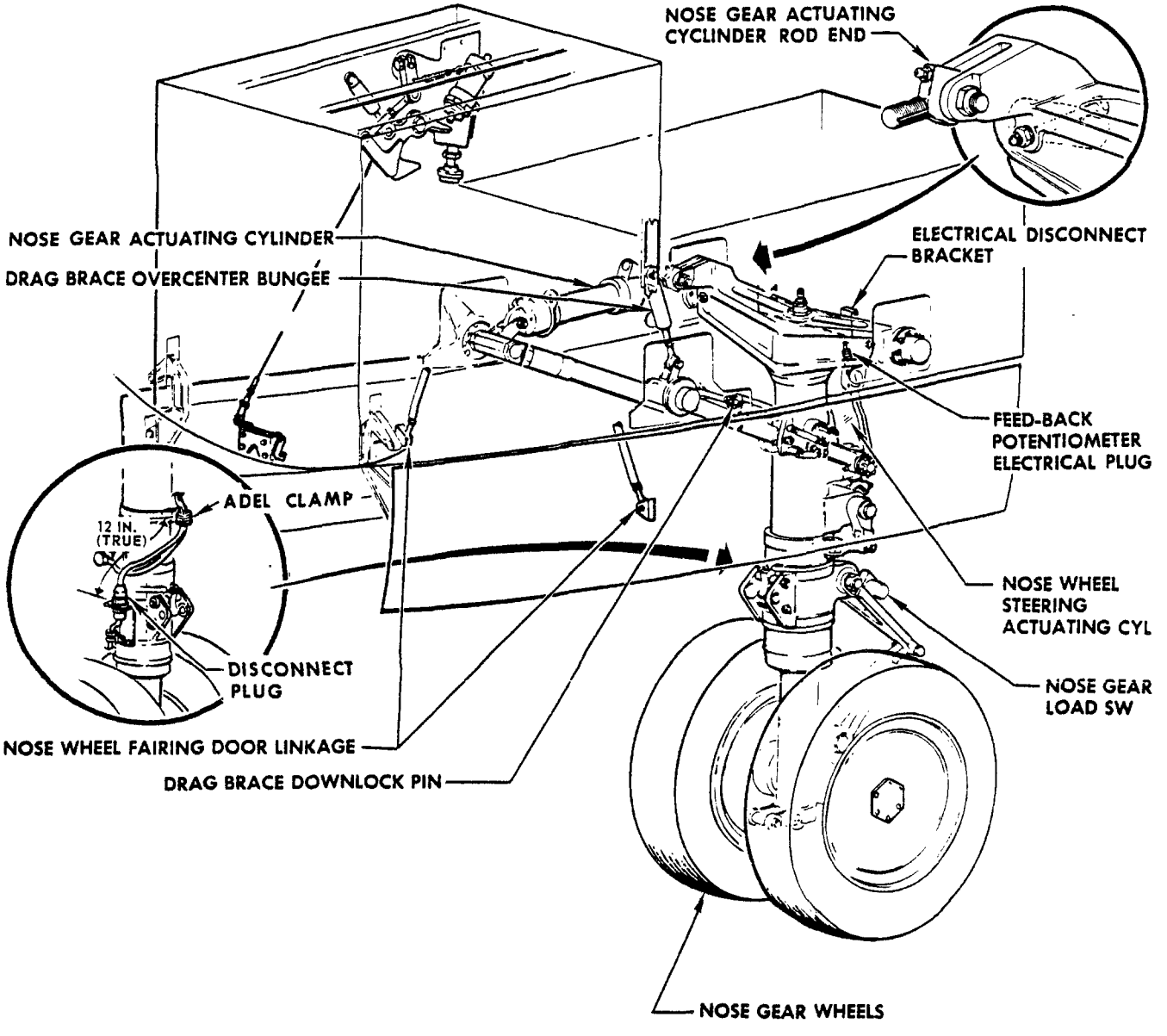

Figure 32-20-401 — Installing and Adjusting Nose Gear Shock Strut Assembly (Sheet 2 of 3)

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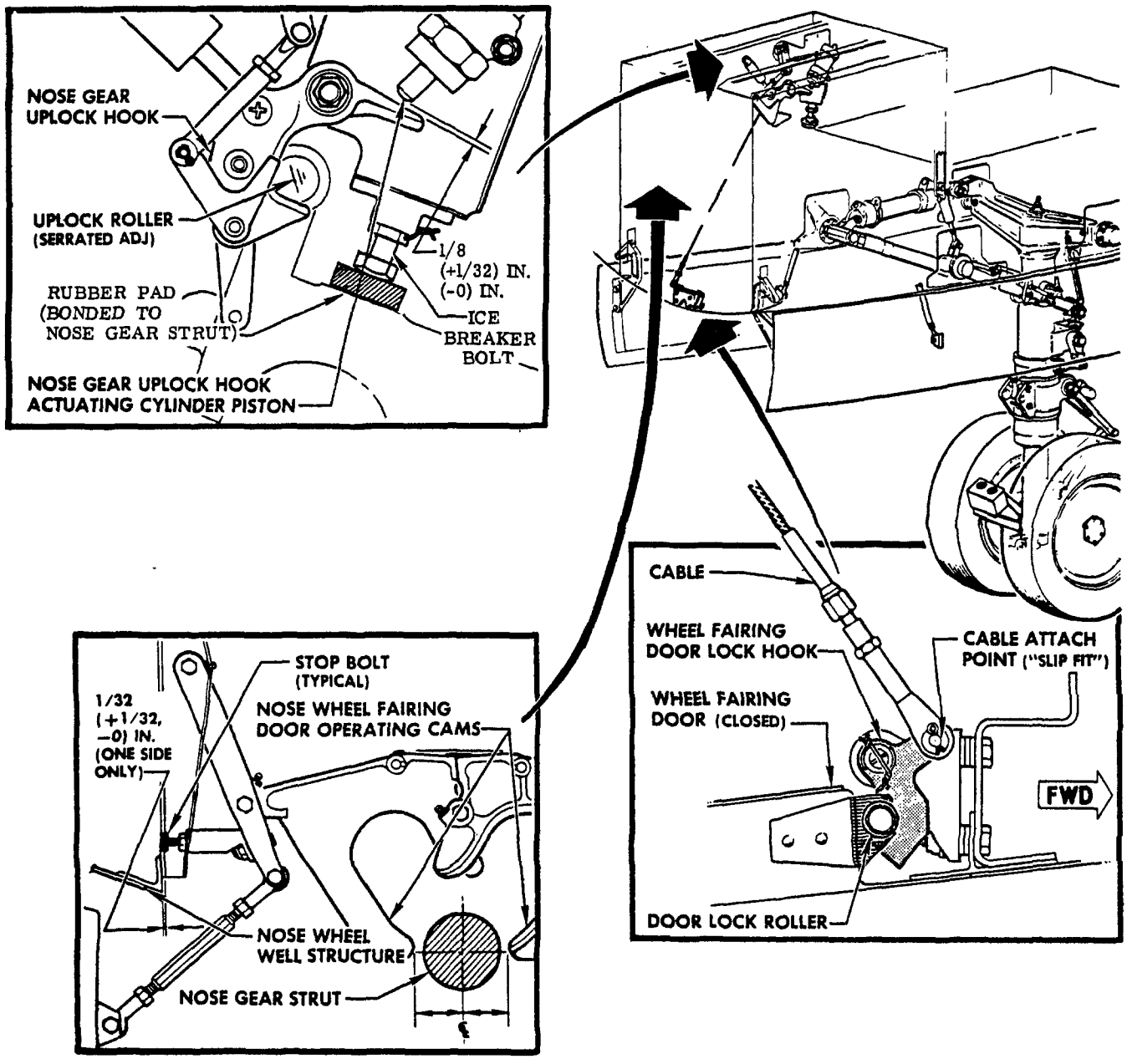
NOSE GEAR UPLOCK HOOK
UPLOCK ROLLER (SERRATED ADJ)
RUBBER PAD (BONDED TO NOSE GEAR STRUT)
NOSE GEAR UPLOCK HOOK ACTUATING CYLINDER PISTON
1/8 (+1/32) IN. (-0) IN.
ICE BREAKER BOLT
CABLE
WHEEL FAIRING DOOR LOCK HOOK
CABLE ATTACH POINT ("SLIP FIT")
WHEEL FAIRING DOOR (CLOSED)
FWD
DOOR LOCK ROLLER
STOP BOLT (TYPICAL)
NOSE WHEEL FAIRING DOOR OPERATING CAMS
1/32 (+1/32, -0) IN. (ONE SIDE ONLY)
NOSE WHEEL WELL STRUCTURE
NOSE GEAR STRUT

Figure 32-20-401 — Installing and Adjusting Nose Gear Shock Strut Assembly (Sheet 3 of 3)

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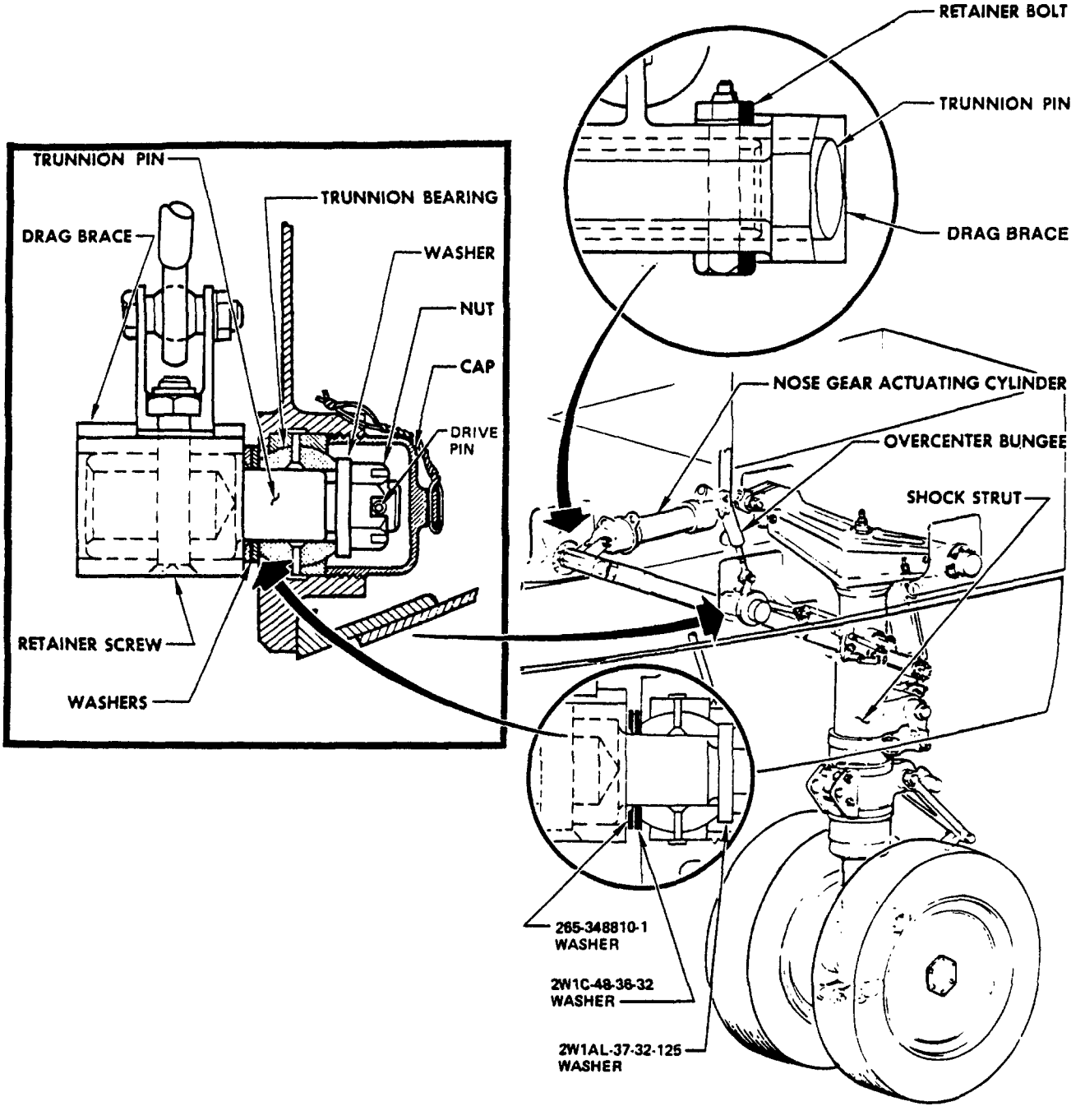

Figure 32-20-402 — Installing and Adjusting Nose Gear Drag Brace Assembly (Sheet 1 of 2)

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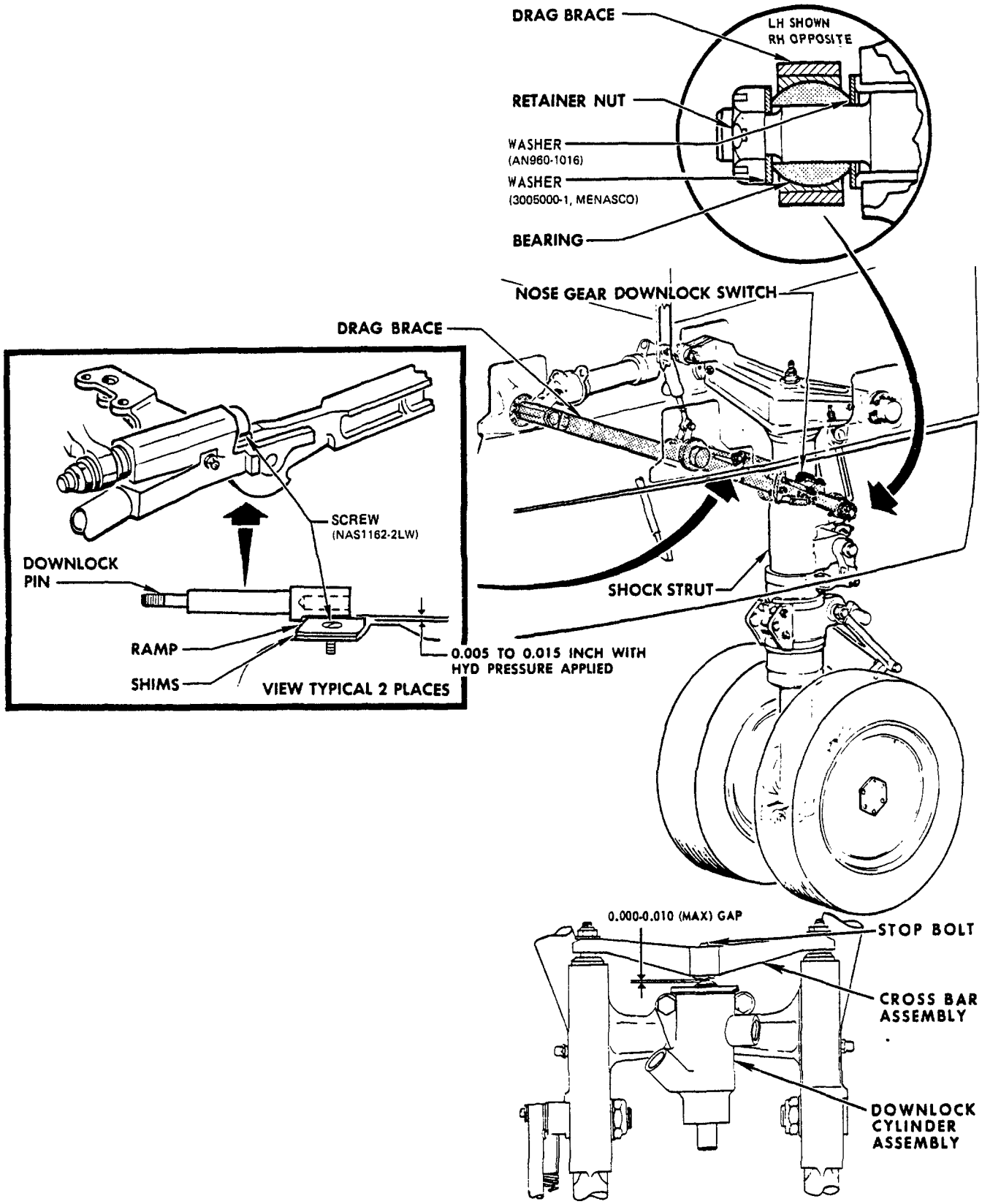

Figure 32-20-402 — Installing and Adjusting Nose Gear Drag Brace Assembly (Sheet 2 of 2)

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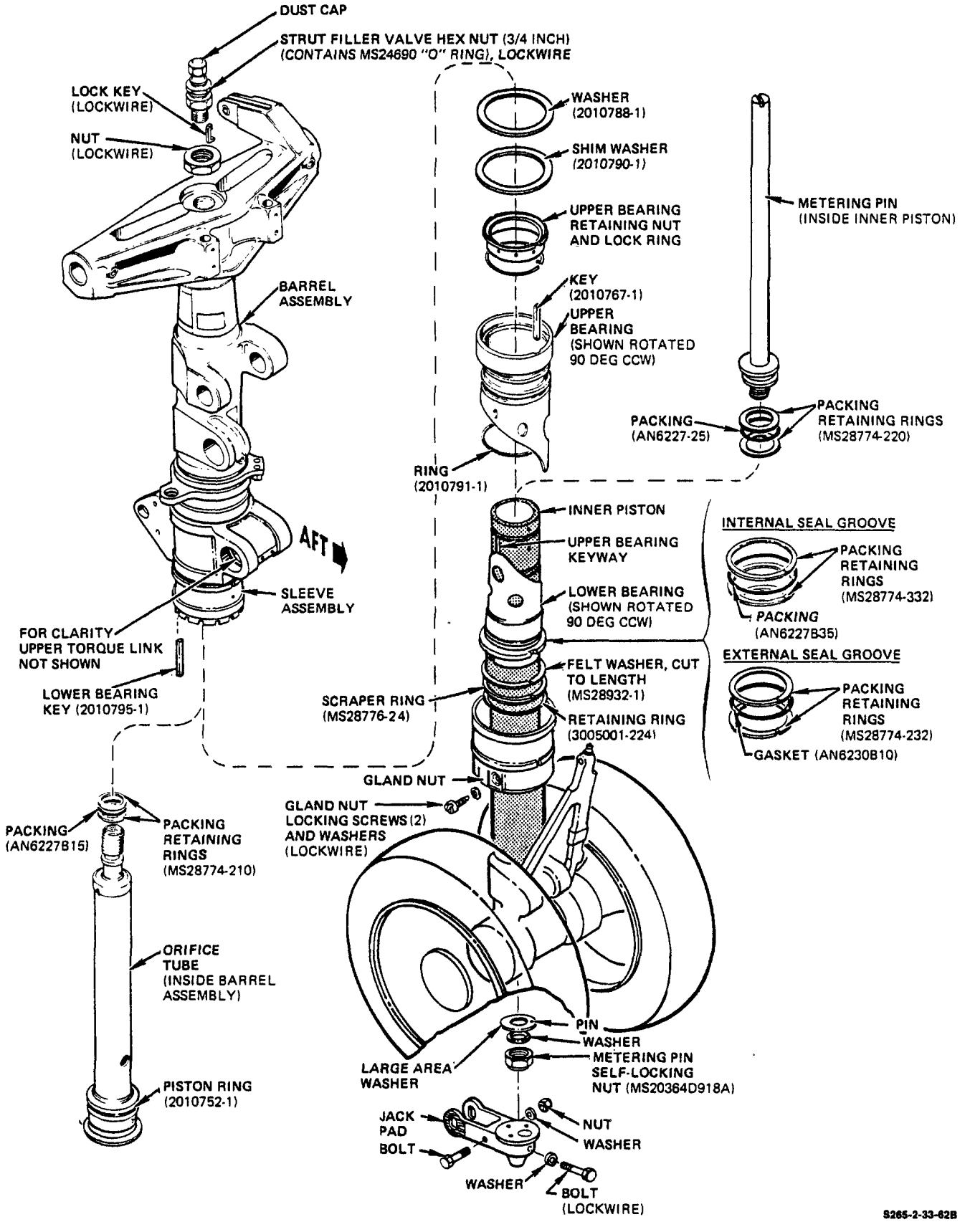

Figure 32-20-403 — Nose Gear Shock Strut

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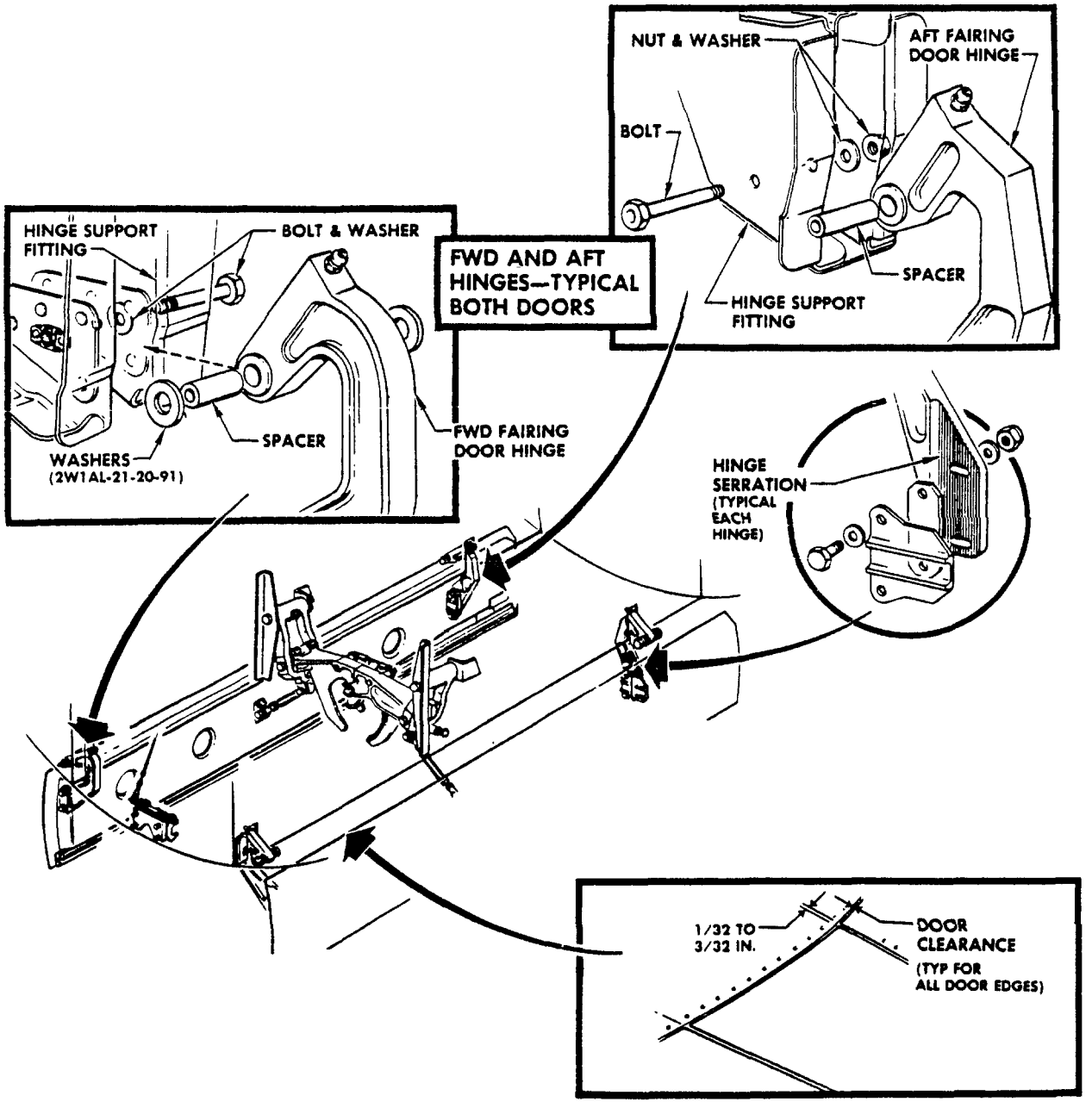

Figure 32-20-404 — Installing and Adjusting Nose Gear Fairing Doors (Sheet 1 of 2)

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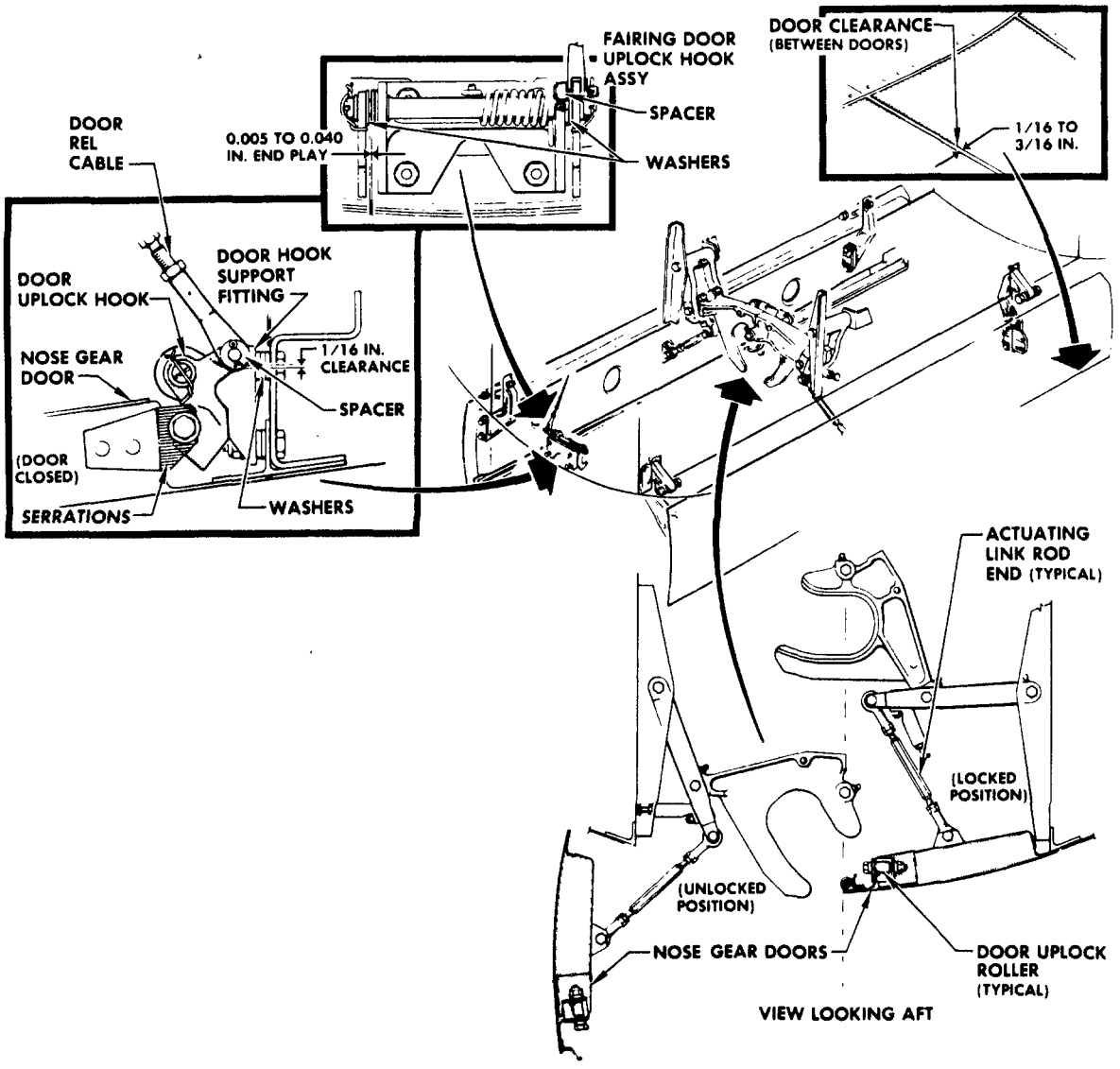

S285-2-33-32

Figure 32-20-404 — Installing and Adjusting Nose Gear Fairing Doors (Sheet 2 or 2)

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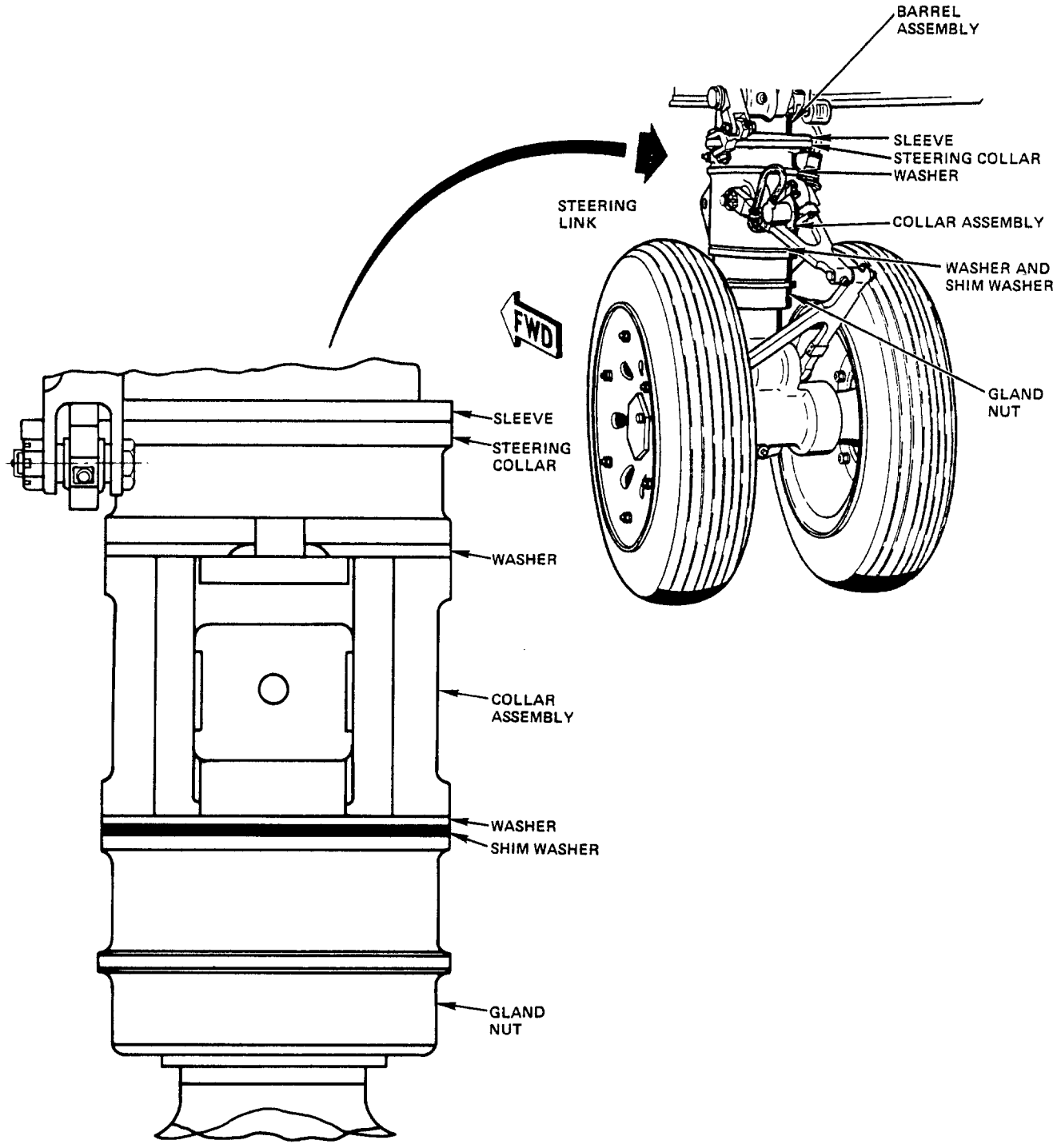

NEW

Figure 32-20-405 — Adjusting Clearance Between Nose Gear Shock Strut Steering Collar and Gland Nut

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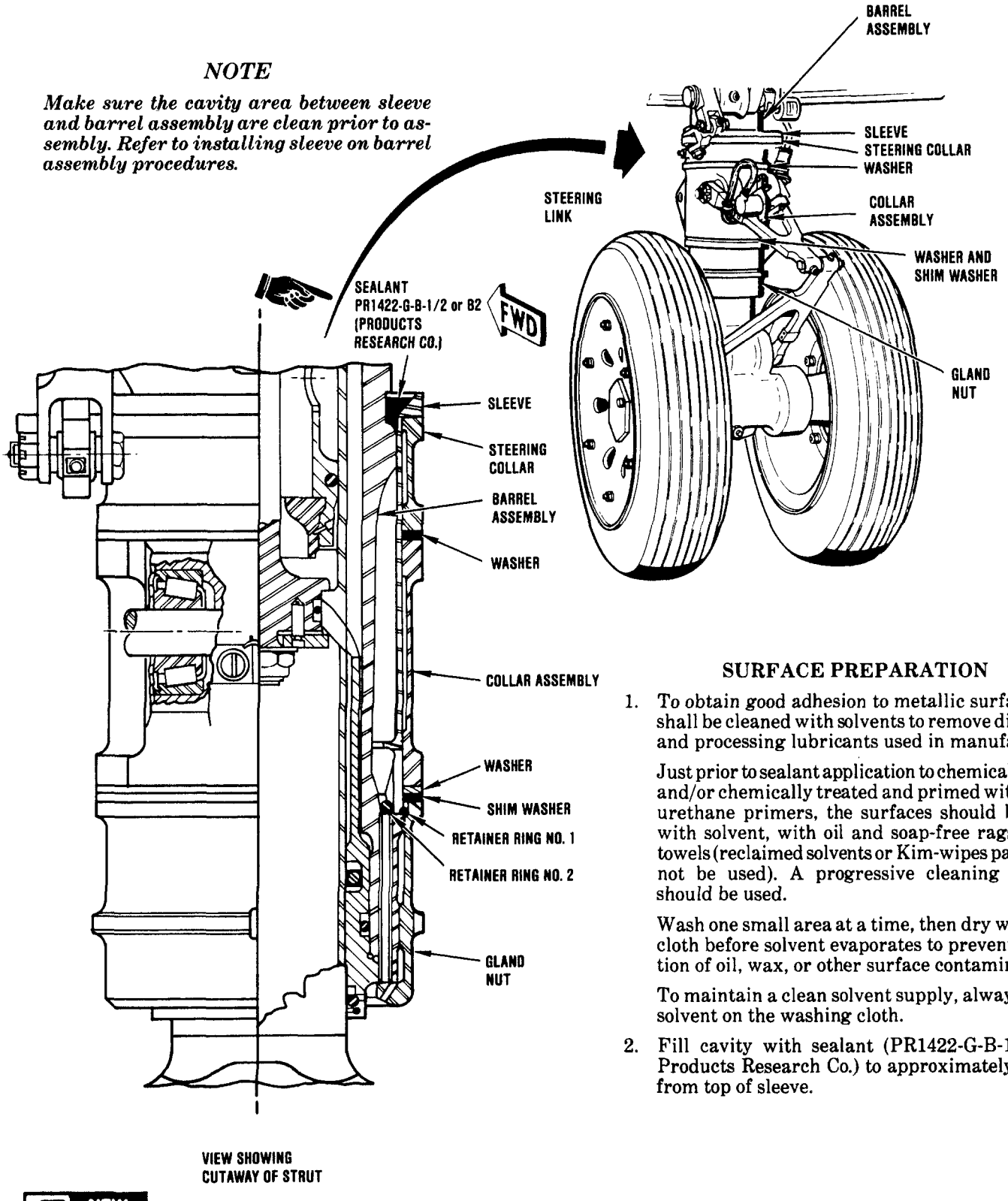
NOTE

Make sure the cavity area between sleeve and barrel assembly are clean prior to assembly. Refer to installing sleeve on barrel assembly procedures.



NEW

SURFACE PREPARATION

1. To obtain good adhesion to metallic surfaces, parts shall be cleaned with solvents to remove dirt, grease, and processing lubricants used in manufacturing.

   Just prior to sealant application to chemically treated, and/or chemically treated and primed with epoxy or urethane primers, the surfaces should be cleaned with solvent, with oil and soap-free rags or paper towels (reclaimed solvents or Kim-wipes paper should not be used). A progressive cleaning procedure should be used.

   Wash one small area at a time, then dry with a clean cloth before solvent evaporates to prevent redeposition of oil, wax, or other surface contaminants.

   To maintain a clean solvent supply, always pour the solvent on the washing cloth.

2. Fill cavity with sealant (PR1422-G-B-1/2 or B2, Products Research Co.) to approximately 0.06 inch from top of sleeve.

Figure 32-20-406 — Re-sealing Between Nose Gear Sleeve and Barrel Assembly — After Sleeve Removal and Installation

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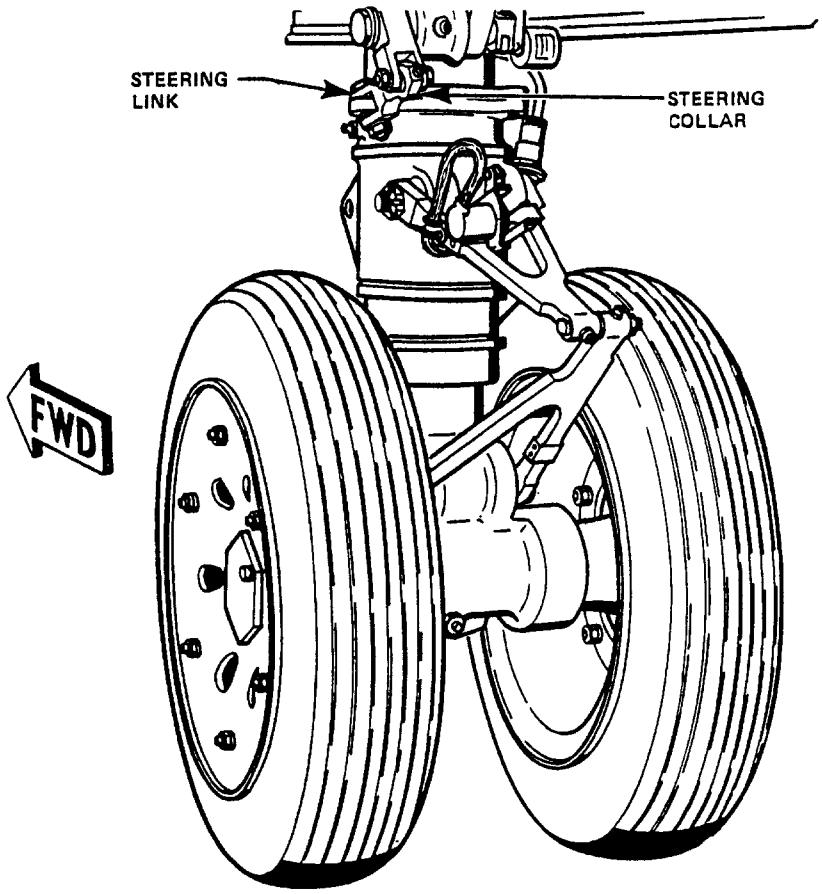

S285-2-13-71A

Figure 32-20-407 — Nose Gear Steering Link Inspection

NEW

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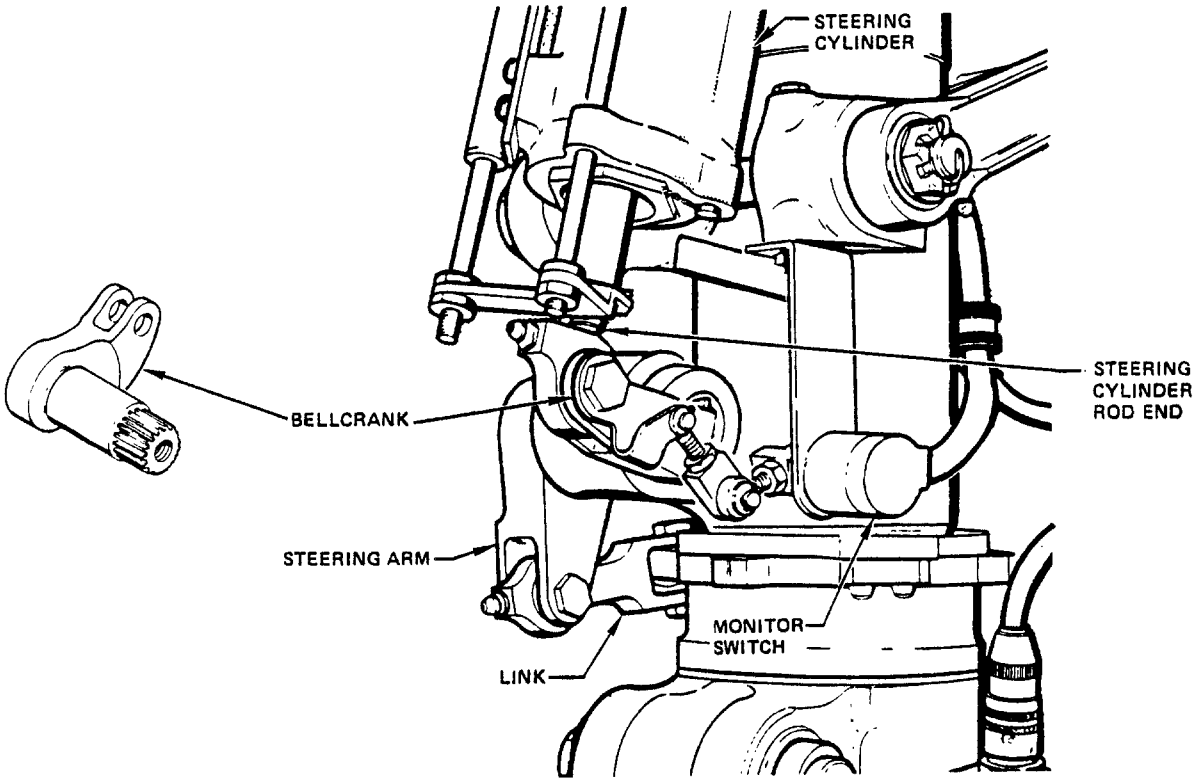

NEW

S255-2-13-72A

Figure 32-20-408 — Nose Landing Gear Shock Strut Bellcrank — P/N 3010812-1, -3, -5 and -11

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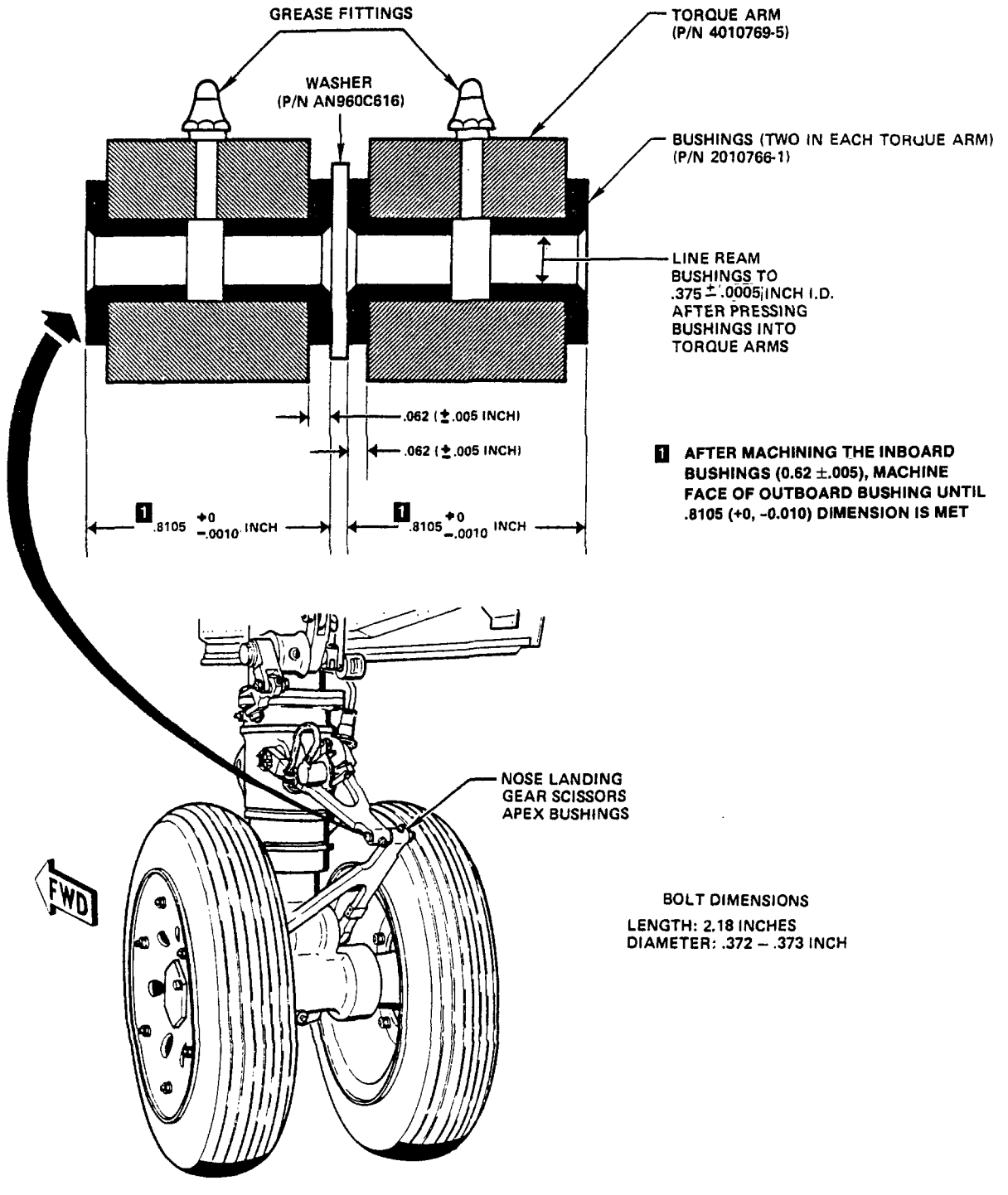

NOTES

- *Outside of bushing must be wet with zinc chromate primer, when pressing bushings into bore.*
- *Bushings must be replaced in pairs.*

S380-2-33-15

Figure 32-20-409 — Nose Landing Gear Scissors Apex Bushings Replacement

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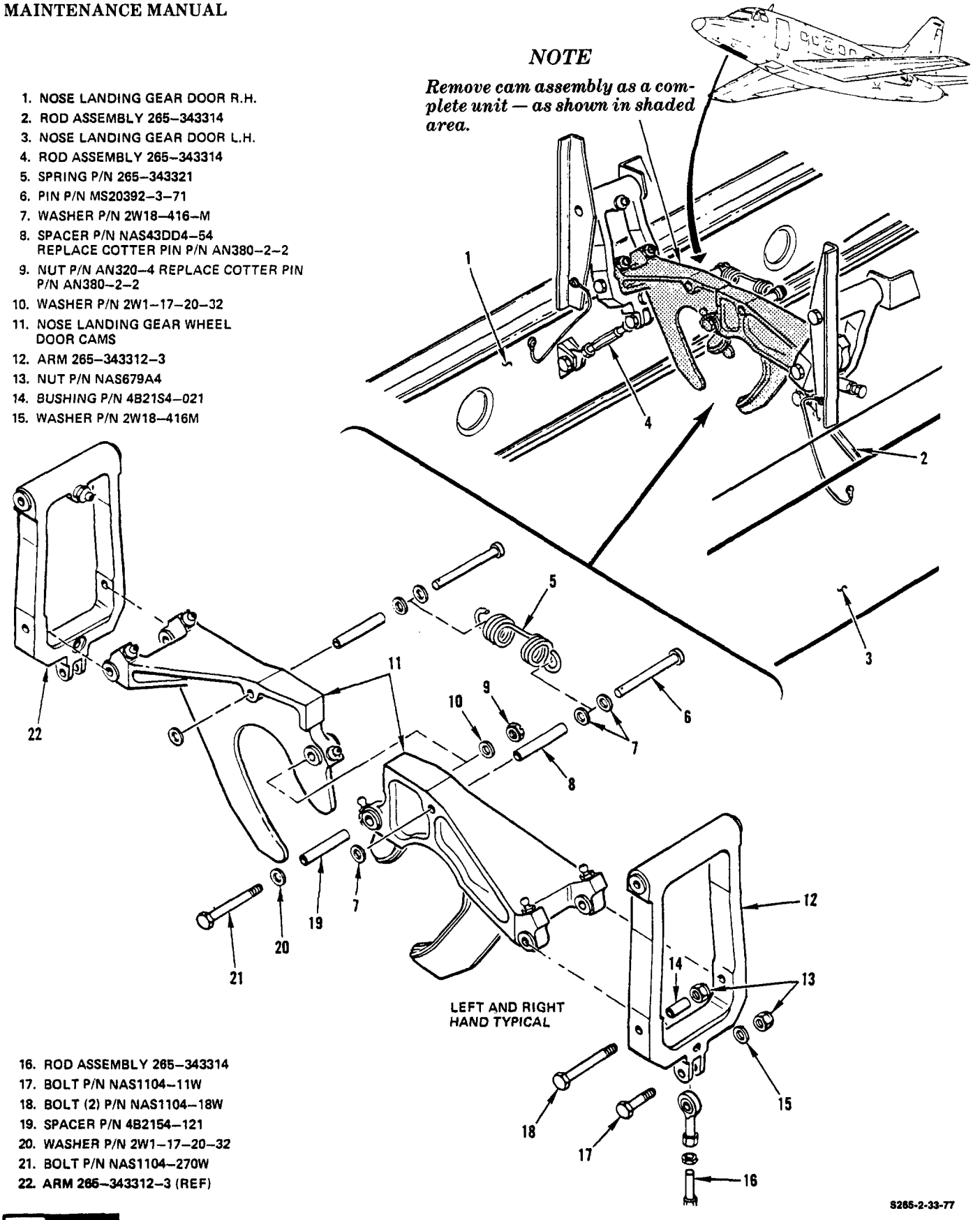
1. NOSE LANDING GEAR DOOR R.H.
2. ROD ASSEMBLY 265–343314
3. NOSE LANDING GEAR DOOR L.H.
4. ROD ASSEMBLY 265–343314
5. SPRING P/N 265–343321
6. PIN P/N MS20392–3–71
7. WASHER P/N 2W18–416–M
8. SPACER P/N NAS43DD4–54 REPLACE COTTER PIN P/N AN380–2–2
9. NUT P/N AN320–4 REPLACE COTTER PIN P/N AN380–2–2
10. WASHER P/N 2W1–17–20–32
11. NOSE LANDING GEAR WHEEL DOOR CAMS
12. ARM 265–343312–3
13. NUT P/N NAS679A4
14. BUSHING P/N 4B21S4–021
15. WASHER P/N 2W18–416M



16. ROD ASSEMBLY 265–343314
17. BOLT P/N NAS1104–11W
18. BOLT (2) P/N NAS1104–18W
19. SPACER P/N 4B2154–121
20. WASHER P/N 2W1–17–20–32
21. BOLT P/N NAS1104–270W
22. ARM 265–343312–3 (REF)

NEW

Figure 32-20-410 — Nose Gear Wheel Door Cam Assembly

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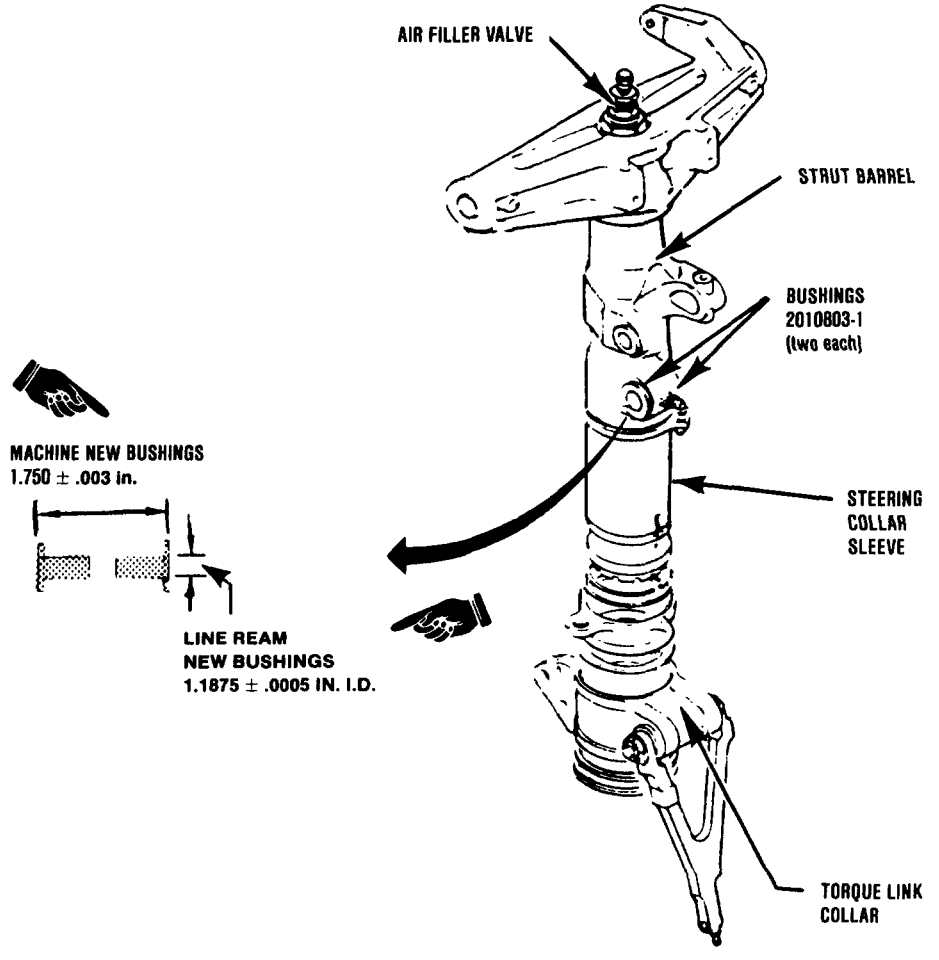

S265-2-33-89

Figure 32-20-411 — Nose Gear Steering Bellcrank Bushings Adjustment

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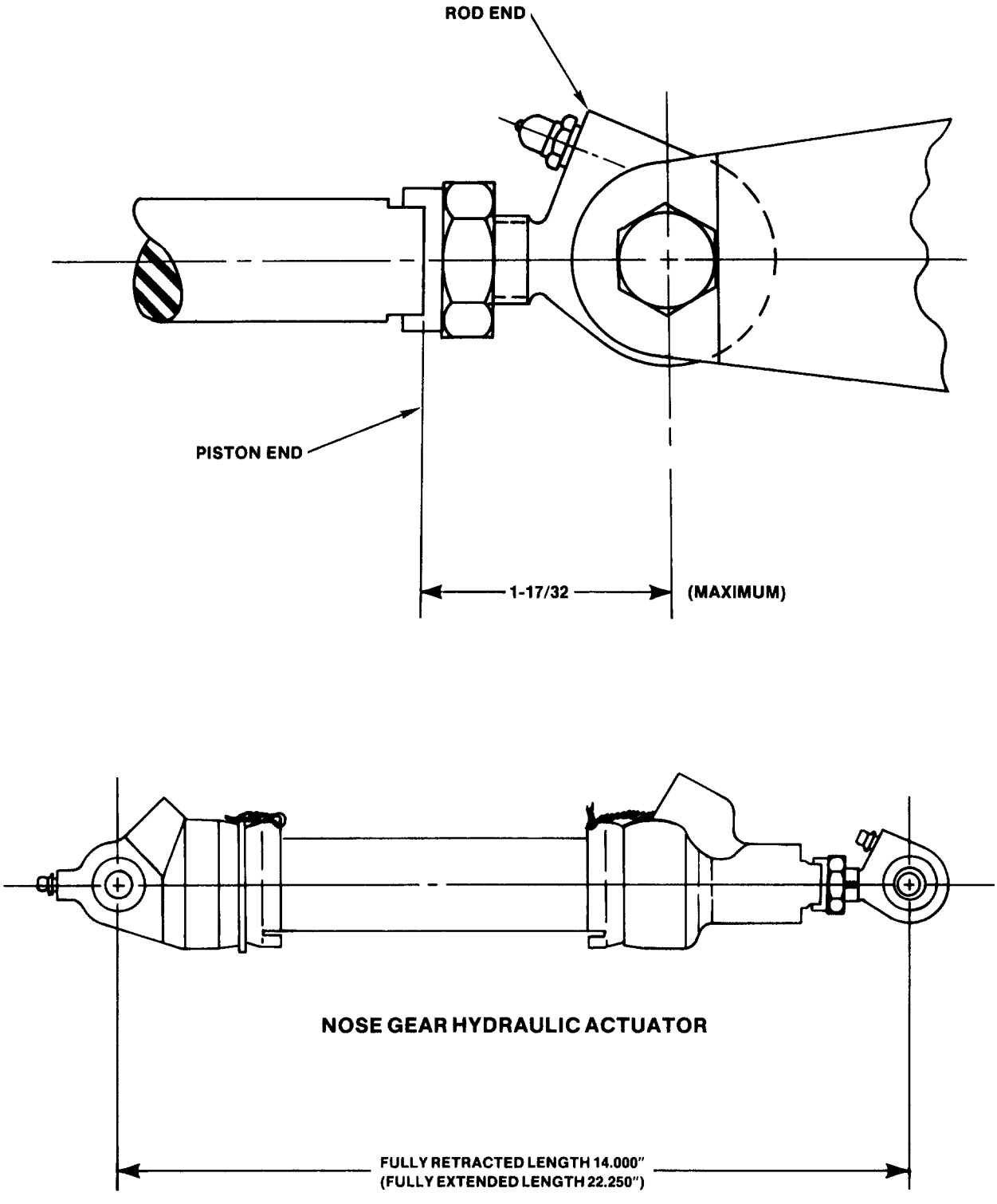

Figure 32-20-412 — NLG Actuator and Rod End Dimension

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NOSE LANDING GEAR

WEAR LIMITS

Wear limits are listed in Figure 32-20-601. This list should be used whenever an inspection or other maintenance procedure calls out a check for "adequate clearance" or "excessive wear."

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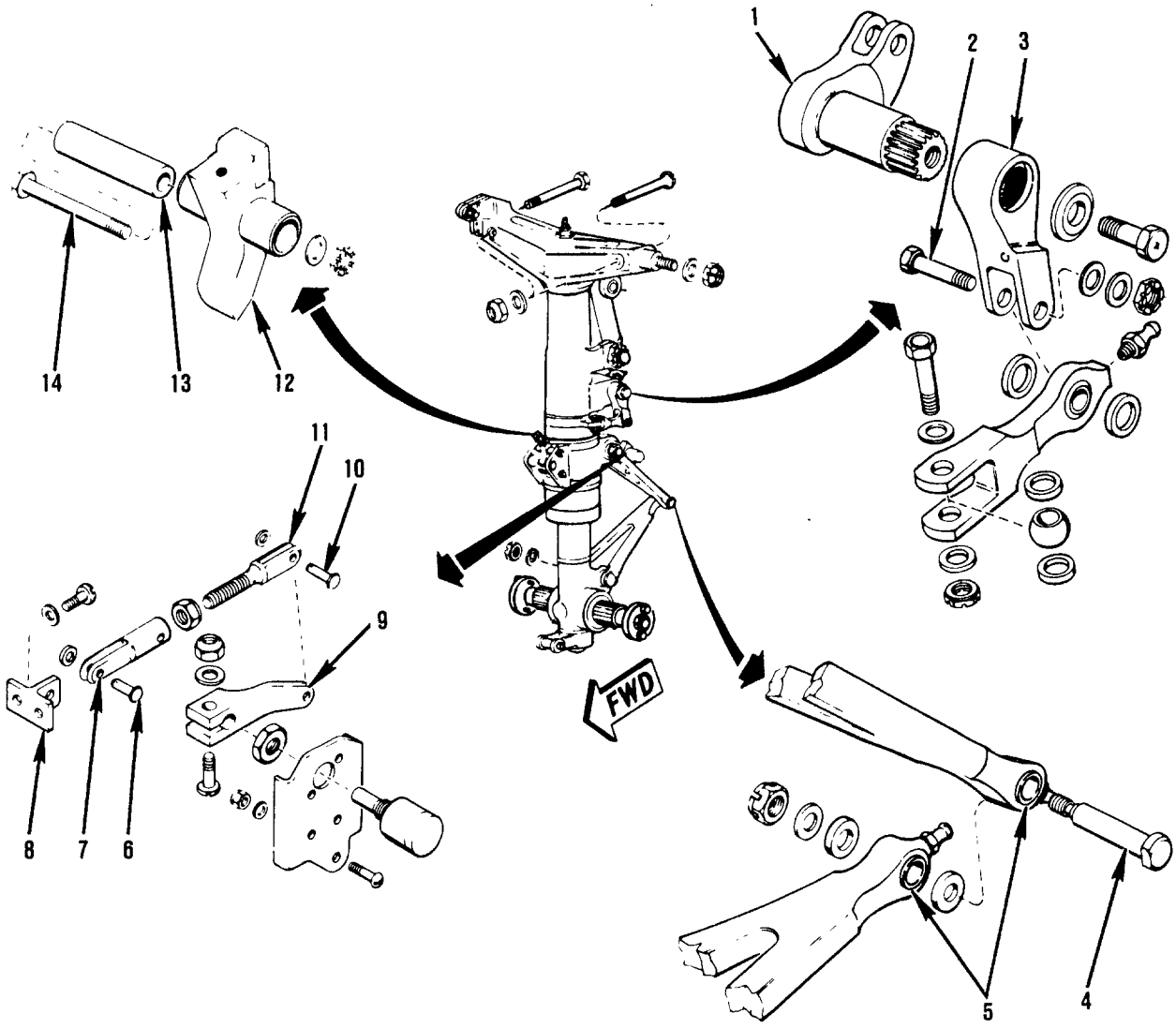

ITEMS (OD to ID fit)	DIAMETRICAL CLEARANCE (INCHES)
Bellcrank (1) OD fit to bushings (22) ID	0.0016 to 0.0031
Bolt (2) OD fit to arm (3) ID	0.0010 to 0.0027
Bolt (4) OD fit to bushing (5) ID	0.0015 to 0.0035
Pin (6) OD fit to link (7) ID	0.0000 to 0.0050
Pin (6) OD fit to bracket (8) ID	0.0000 to 0.0050
Pin (10) OD fit to arm (9) ID	0.0000 to 0.0050
Pin (10) OD fit to link (11) ID	0.0000 to 0.0050
Bushing (13) OD fit to lock (12) ID	0.0020 to 0.0040
Bolt (14) OD fit to bushing (13) ID	0.0010 to 0.0036
Bearing (16) major OD fit to barrel (21) ID	0.0035 to 0.0080
Orifice tube (17) major OD fit to piston (15) ID	0.0280 to 0.0400
Sleeve assy (20) OD fit to collar (18) ID	0.0026 to 0.0094
Sleeve assy (20) OD fit to collar (19) ID	0.0038 to 0.0082

Figure 32-20-601 — Nose Landing Gear Wear Limits (Sheet 1 of 2)

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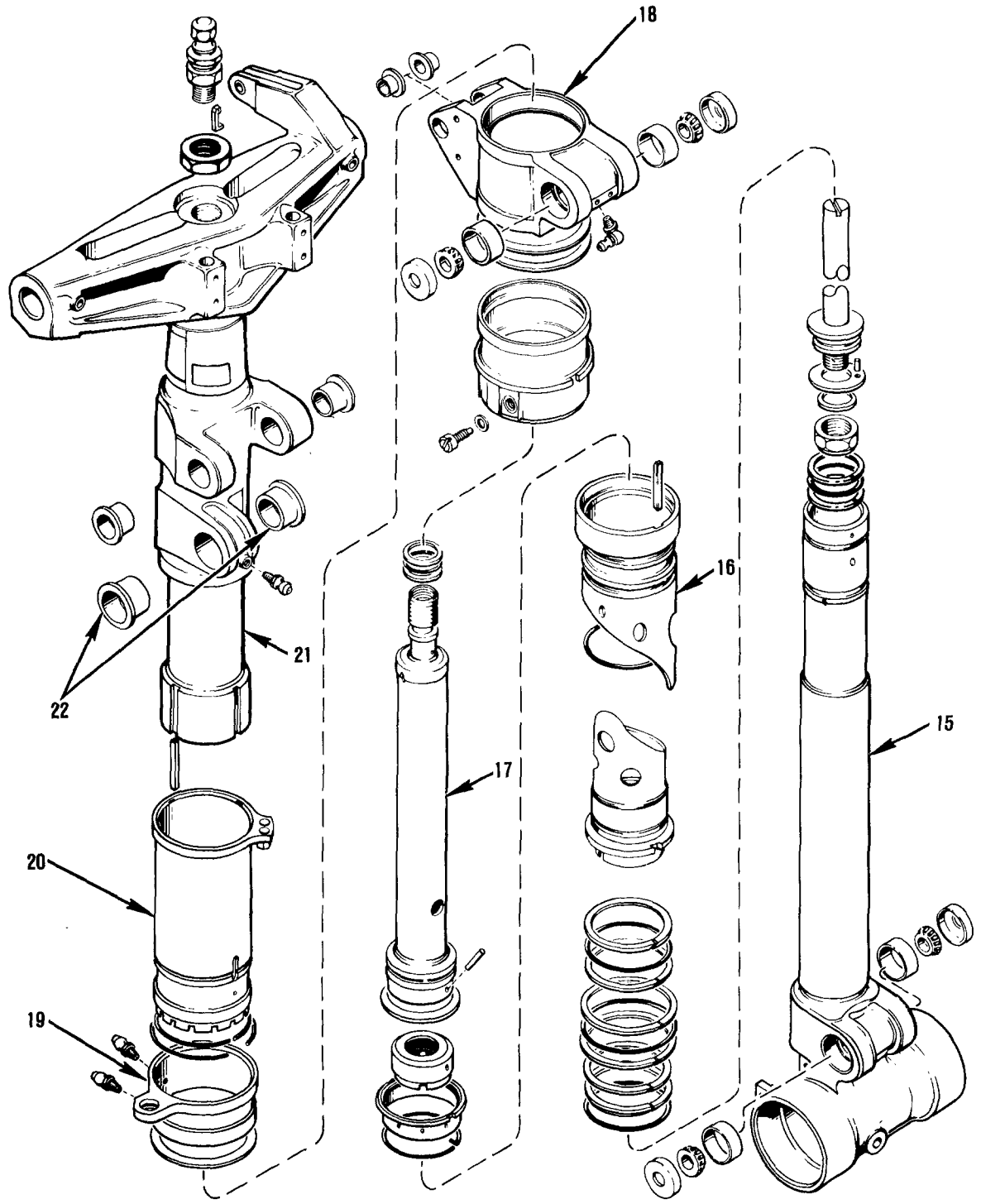

Figure 32-20-601 — Nose Landing Gear Wear Limits (Sheet 2 of 2)

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INSPECTION/CHECK (Continued)

INSPECTION OF NLG SLEEVE ASSEMBLY, BARREL ASSEMBLY, AND RELATED COMPONENTS (AIRPLANES COMPLYING WITH S/B 86-6)

EQUIPMENT/TOOLS

- Corrosion Preventive Compound, P/N C629 .. 1 Pint (MIL-C-16173D, Grade 1)
- Lectro-Solv Cleanser (or equivalent) ... As required P/N 597ES
- Sealant, P/N 1422B1/2 or B2 ..... As required
- Packing Retaining Rings, P/N MS28774-332 ...2 Ea
- Packing Retaining Rings, P/N MS28774-232 ...2 Ea
- Gasket, P/N AN6230B10 or AN6230-10 .....1 Ea
- Packing, P/N AN6227B35 or AN6230-35 .....1 Ea
- Felt Washer, P/N MS28932-1.....1 Ea
- Scraper Ring, P/N MS28776M2-24 .....1 Ea
- Shim Washer, P/N 2010790-1.....1 Ea
- Adjustable Pin Spanner Wrench (2 - 4-3/4 inches) Snap-On P/N APS363 or equivalent.

PROCEDURE

NOTE

This is a 12-month recurring inspection. After the initial compliance requirement is met, it may be to the operator's advantage to perform this inspection concurrent with the 12-month recurring inspection of the NLG shock strut since both procedures involve disassembly of the NLG.

1. Place jacks under airplane and raise airplane enough to permit landing gear retraction. Install ground safety pins.
2. Turn off battery and electrical master switches, remove external power and hydraulic pressure, and dump normal and auxiliary accumulator pressure.

WARNING

To prevent possible serious injury to personnel and/or damage to the airplane, ensure that all air is bled from shock strut before disassembly.

WARNING

Deplete normal and auxiliary accumulator pressure. This will prevent accidental operation of the auxiliary accumulator power assist retraction system and prevent serious injury and/or damage to the airplane.

NOTE

Refer to figure 32-20-602 for identification of parts during disassembly/reassembly.

3. Remove strut air filler dust cap. Slowly loosen filler valve enough to allow air to slowly escape from the shock strut. Partially compress strut to release any trapped air.
4. Remove nose gear wheels and axle. This will allow easier strut handling and will help prevent damage to the orifice piston ring during reassembly.
5. Disconnect nose gear torque arms (scissors) and secure upper torque arm to upper part of strut.
6. Place suitable container under shock strut to catch hydraulic fluid contained in strut.
7. Remove two gland nut locking screws and washers. Remove gland nut with spanner wrench and then carefully slide inner piston from strut barrel. Use care as two lower bearing keys P/N 2010795-1 can be dislodged and fall out of their keyways when the inner piston is removed.
8. Disconnect steering link from steering collar.

CAUTION

Retainer ring No. 1 retains the torque link collar and related assemblies on the sleeve assembly. Take care to prevent these assemblies from accidently sliding off when retainer ring No. 1 is removed in step 9.

9. Remove retainer ring No. 1 from lower portion of sleeve assembly taking care to manually retain the torque link collar and related assemblies to prevent them from accidently sliding off.
10. Carefully remove shim washer P/N 2010790-1, thrust washer P/N 2010778-1, torque link collar, second thrust washer P/N 2010778-1 and steering collar from sleeve assembly.
11. Using a suitable pointed tool, carefully remove sealing compound from cavity between barrel and sleeve assemblies. Use care to prevent damage to these assemblies.
12. Using a rubber or wooden mallet, gently tap the bottom of the sleeve assembly to drive it upward about 1/4 inch until it bottoms out against upper portion of barrel assembly. Use care as the sleeve assembly is aluminum and can be easily damaged.

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INSPECTION/CHECK (Continued)

INSPECTION OF NLG SLEEVE ASSEMBLY, BARREL ASSEMBLY, AND RELATED COMPONENTS (AIRPLANES COMPLYING WITH S/B 86-6) (Cont'd)

NOTE

The sleeve assembly must be driven upwards until it bottoms out against the upper portion of barrel assembly in order to allow release of retainer ring No. 2 in step 13.

13. Using a screwdriver or other suitable instrument, reach through the two elongated holes in the sleeve assembly and pry retainer ring No. 2 up out of the lower retaining groove and push it upwards until it locks in the upper retaining groove located at the top of the elongated holes. Using a pocket knife or other suitable instrument, carefully rotate the retainer ring until it can be verified that both ends of the ring are locked into the upper retaining groove.

CAUTION

Retainer ring No. 2 must be fully locked into the upper retaining groove of the sleeve assembly or damage to the ring and sleeve may occur during sleeve removal.

14. Slide sleeve assembly from barrel assembly. This may require gently tapping the sleeve assembly with a rubber or wooden mallet. Use care as the sleeve assembly is aluminum and can be easily damaged.

15. Using Lectro-Solv cleanser 597ES (or equivalent), clean sleeve assembly, steering collar, and torque link collar for inspection.

16. Inspect the inner and outer surfaces of sleeve assembly for corrosion, pitting and other obvious damage. Inspect the bearing surfaces of the sleeve assembly for galling. If damage is found, contact Sabreliner Corporation's engineering department for further instructions. If sleeve assembly checks OK, apply a thin coat of corrosion preventative compound MIL-C-16173D, Grade 1, to the interior surface of the sleeve assembly.

17. Inspect the steering and torque link collars, especially the inner bearing surfaces for galling, corrosion and other obvious damage. If damage is found, contact Sabreliner Corporation's engineering department for further instructions.

18. Using Lectro-Solv cleanser 597ES (or equivalent), clean lower portion of barrel assembly as shown in Figure 32-20-602 (Sheet 2 of 2) and inspect this area for corrosion, pitting and other obvious damage. If damage is found, contact Sabreliner's engineering department for further instructions. If barrel assembly checks OK, apply corrosion preventative compound MIL-C-16173D, Grade 1 to the inspected area.

19. Install retainer ring No. 2 into the upper retaining groove inside the sleeve assembly (upper groove is located at the top of the elongated holes). Orient the two ends of the retainer ring so that they can both be seen through one of the elongated holes in the sleeve assembly.

CAUTION

It is possible to install sleeve assembly backwards, 180° from the proper position. The correct position is with the steering collar stop oriented to face the right side of the airplane.

20. Install sleeve assembly on barrel assembly and gently tap it upwards with a rubber or wooden mallet until it bottoms out against the upper portion of the barrel assembly. Use care as the sleeve assembly is aluminum and can be easily damaged. Ensure that the sleeve assembly is oriented with the steering collar stop facing the right side of the airplane.

21. Using a screwdriver or other suitable instrument, reach through the two elongated holes in the sleeve assembly and pry retainer ring No. 2 out of the upper retianing groove and push it downwards until it locks into the lower retaining groove located at the bottom of the elongated holes. Verify that the retainer ring is fully seated in the lower retaining groove by viewing the ring through both of the elongated holes; both ends of the ring must be visible through one of the elongated holes to ensure that it is fully seated. Using a rubber or wooden mallet, gently tap the sleeve assembly downward until the No. 2 retainer ring restricts the downward movement.

22. Rotate the sleeve assembly as necessary to align the two keyways located in the bottom of the sleeve assembly with the two keyways in the bottom of the barrel assembly. Ensure that the sleeve assembly remains oriented with the steering collar stop facing the right side of the airplane. Slide the two lower bearing keys P/N 2010795-1 into these keyways and attempt to rotate the sleeve assembly to apply pressure to the keys to keep them in place.

NOTE

The adjustment for proper clearance between the gland nut and thrust washer is a critical adjustment in proper rigging of the nose gear. Perform a clearance check regardless of whether or not inspection requires replacement of parts.

23. Perform procedures to adjust for proper clearance between installed gland nut and thrust washer P/N 2010778-1. See 32-20.

NEW

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INSPECTION/CHECK (Continued)

INSPECTION OF NLG SLEEVE ASSEMBLY, BARREL ASSEMBLY, AND RELATED COMPONENTS (AIRPLANES COMPLYING WITH S/B 86-6) (Cont'd)

CAUTION

The upper and lower bearings on the lower piston serve as centering cams. When reinserting the lower piston into the barrel assembly, it is possible to install the lower bearing 180° out-of-phase. This condition would prevent full extension, center locking and proper retraction of the strut.

24. Slide inner piston assembly into bore of the barrel assembly. Orient point of lower bearing to face forward and align the two keyways in the lower bearing flange with the two lower bearing keys P/N 2010795-1 in sleeve assembly. Carefully slide inner piston up into barrel assembly to the point of which lower bearing is firmly seated. Start gland nut on sleeve assembly. Using spanner wrench, torque gland nut to 15-25 ft-lbs.
25. If slots in lower sleeve assembly are not aligned with screwholes in gland nut, back off gland nut until first set of two slots in lower sleeve assembly are aligned with threaded screw holes in gland nut.
26. Install gland nut lockscrews and washers. Safety with lockwire.

CAUTION

Torque arms (scissors) must be free to rotate at apex pivot. Do not overtighten nut.

27. Reconnect and safety torque arms (scissors). Tighten nut to obtain a clearance of 0.001 to 0.002 inch between torque arms at apex pivot.
28. Roconnect and safety steering link to steering collar.

CAUTION

To ensure a water-tight seal between the sleeve and barrel assemblies, take care to observe the surface preparation instructions given in step 29 and the sealant mixing instructions provided by the manufacturer.

29. Apply sealant between sleeve assembly and barrel assembly per Figure 32-20-602. Just prior to sealant application, clean the cavity between the barrel and sleeve assemblies with Lectro-Solv cleanser P/N 597ES (or equivalent) to remove dirt, grease and processing lubricant used in manufacturing. The surfaces should be cleaned with oil and soap-free cloths lightly dampened with solvent (reclaimed solvent or Kim-wipes paper should not be used.) Wash one small area at a time, then dry with a clean cloth before solvent evaporates to prevent redisposition of oil, wax or other surface contaminents. Dry cavity with forced, dry, clean air before applying sealant.

CAUTION

When servicing nose landing gear shock strut, take care to observe proper hydraulic fluid level.

30. Service nose landing gear shock strut per chapter 12; then check strut for leaks.
31. Remove ground safety pins and perform operational check of landing gear.

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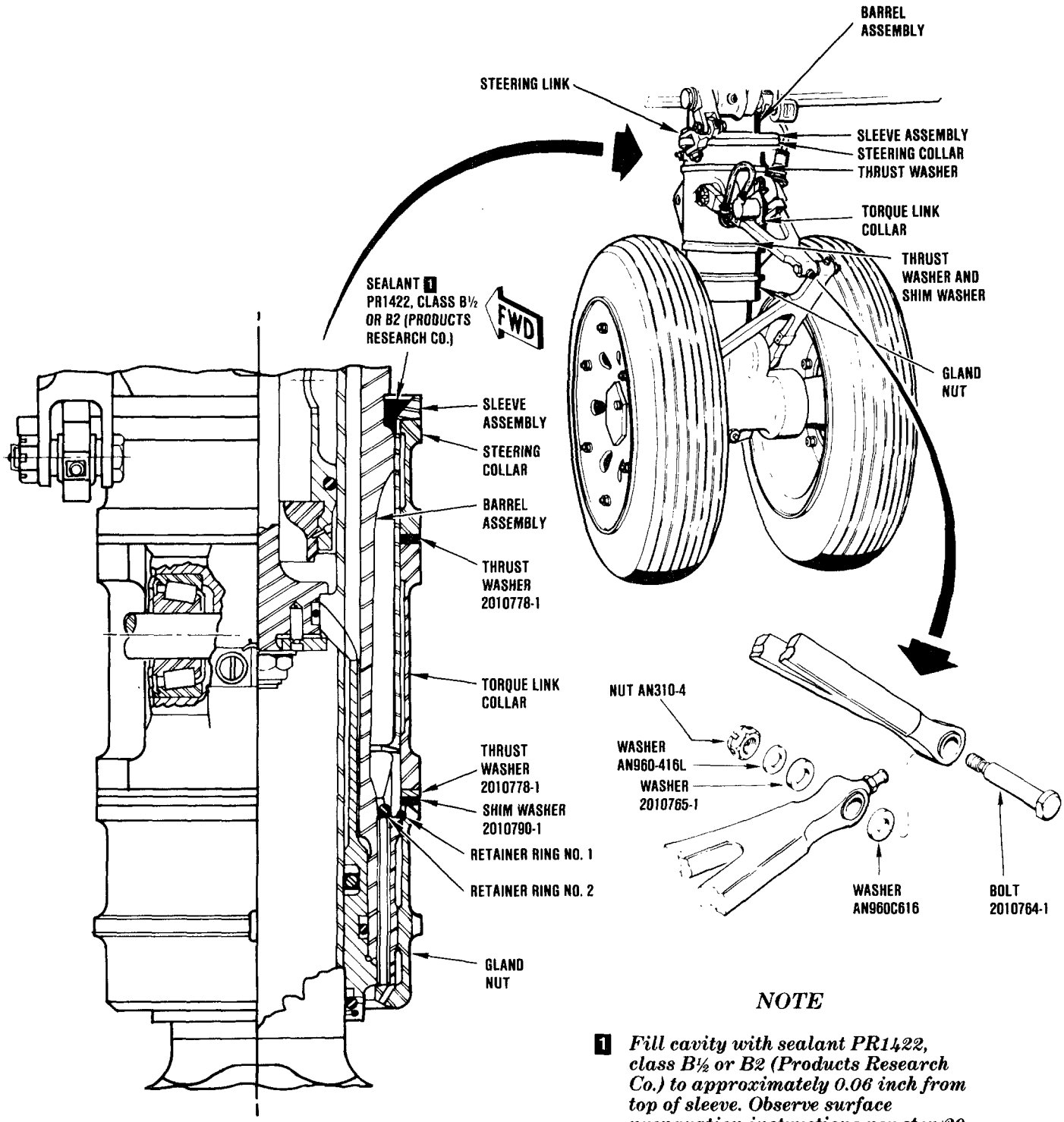

VIEW SHOWING CUTAWAY OF STRUT

NOTE

1 *Fill cavity with sealant PR1422, class B½ or B2 (Products Research Co.) to approximately 0.06 inch from top of sleeve. Observe surface preparation instructions per step 29 of Accomplishment Instructions prior to sealant application.*

NEW

Figure 32-20-602 — Disassembly/Reassembly of Nose Landing Gear Strut (Sheet 1 of 2)

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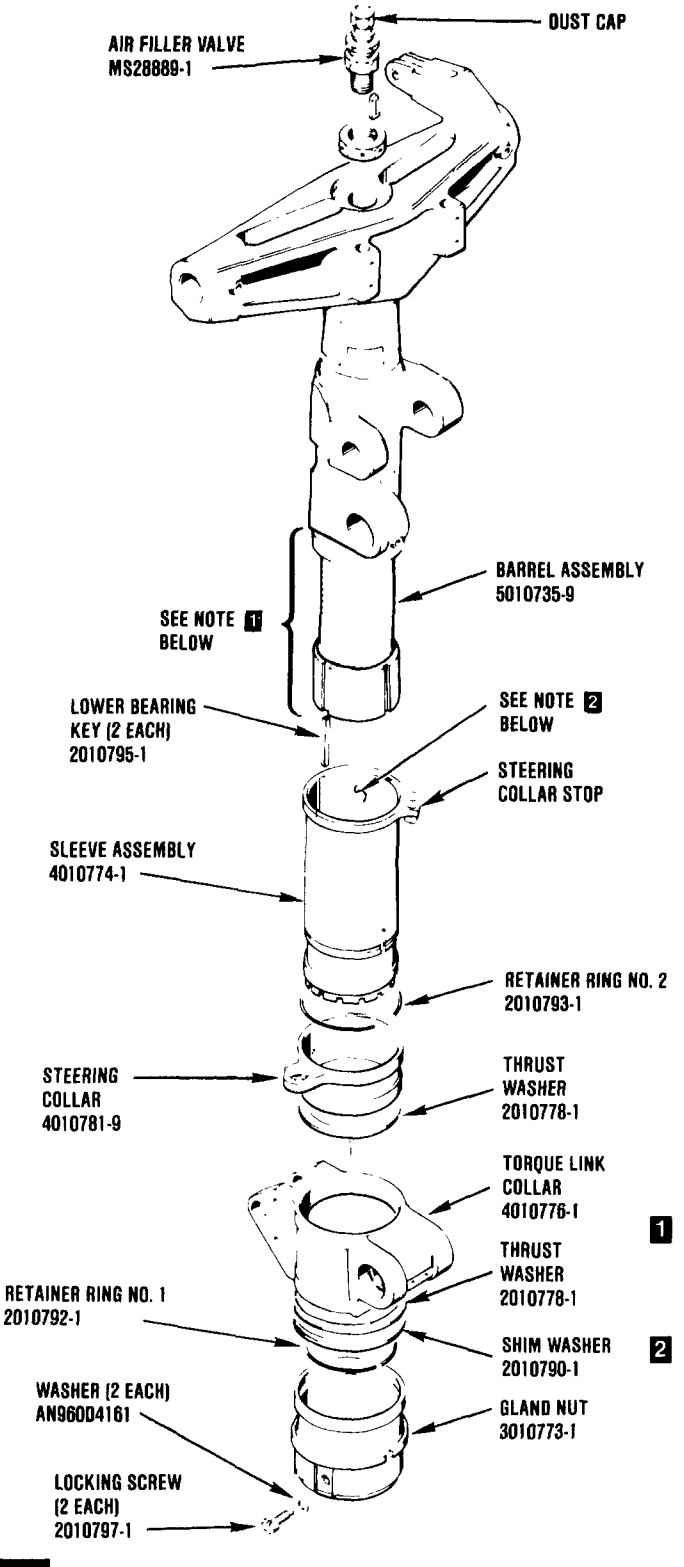

NOTES

1. *Inspect lower portion of barrel assembly and apply corrosion preventative compound MIL-C-16173D, Grade 1 to inspected area.*
2. *Apply a thin coat of corrosion preventative compound MIL-C-16173D, Grade 1, to interior surface of sleeve assembly*

NEW

Figure 32-20-602 — Disassembly/Reassembly of Nose Landing Gear Strut (Sheet 2 of 2)

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INSPECTION/CHECK (Continued)

INSPECTION OF NOSE LANDING GEAR AXLE AND OUTER BEARING SEAL (Ref: S/B 93-1)

Equipment/Tools

- Aircraft Jacks and Jack Pads.
- Magnetic Inspection Equipment
- Proper Solvent (Per Corrosion Control Manual SR-76-023)

Procedure

1. Raise nose wheel off the ground by using aircraft jacks at the nose fuselage jack point and at the two wing jack points.
2. Remove nose wheels, seals, bearings, and axle in accordance with applicable Maintenance Manual procedures.
3. Inspect the 3010741-3 seals for evidence of improper installation. This will be characterized by wear on the face of the seal that has the cup portion. See Figure 32-20-603 (Sheet 1 of 3).

NOTE

Seals with wear indications deeper than .020 inch shall be discarded and replaced with new parts.

4. Using adequate solvent, clean the entire internal and external surfaces. Inspect the entire internal and external surfaces for corrosion. If corrosion is found, contact Sabreliner Engineering for disposition.
5. Perform a Magnetic Particle inspection of the threaded areas for cracks using both circular and longitudinal techniques in accordance with MIL-STD-1949. Current application requirements are as follows: See Figure 32-20-603 (Sheet 2 of 3)
   - Longitudinal Magnetization: 900 D.C. amps.
   - Circular Magnetization: D.C. current = $\frac{6400}{N}$ where N = number of turns in coil.

NOTE

4010745-3 or -5 axles with verified crack indications shall be discarded and replaced with a 4010745-5 axle.

6. Install axle, bearings, seals and nose wheels in accordance with applicable Maintenance Manual procedures.

CAUTION

Ensure proper installation of 3010741-3 seals, cup portion facing outboard. See Figure 32-30-603 (Sheet 3 of 3). Improper installation could result in substantially reduced fatigue life of the nose landing gear axle.

7. Check for proper installation and safeties. (See applicable Maintenance Manual Instructions.)
8. Lower airplane and remove jacks.

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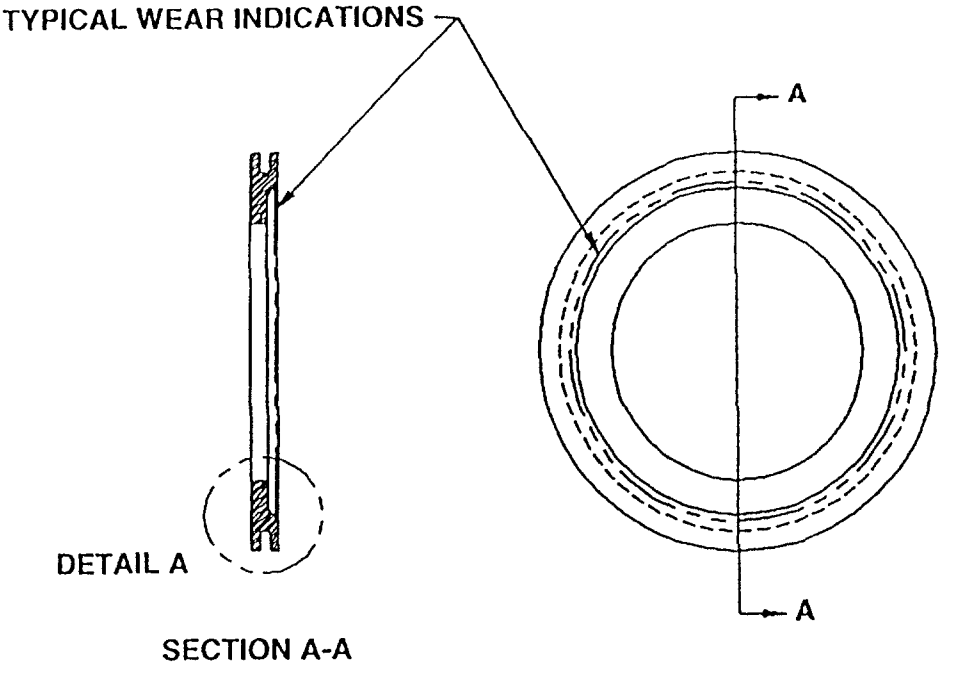
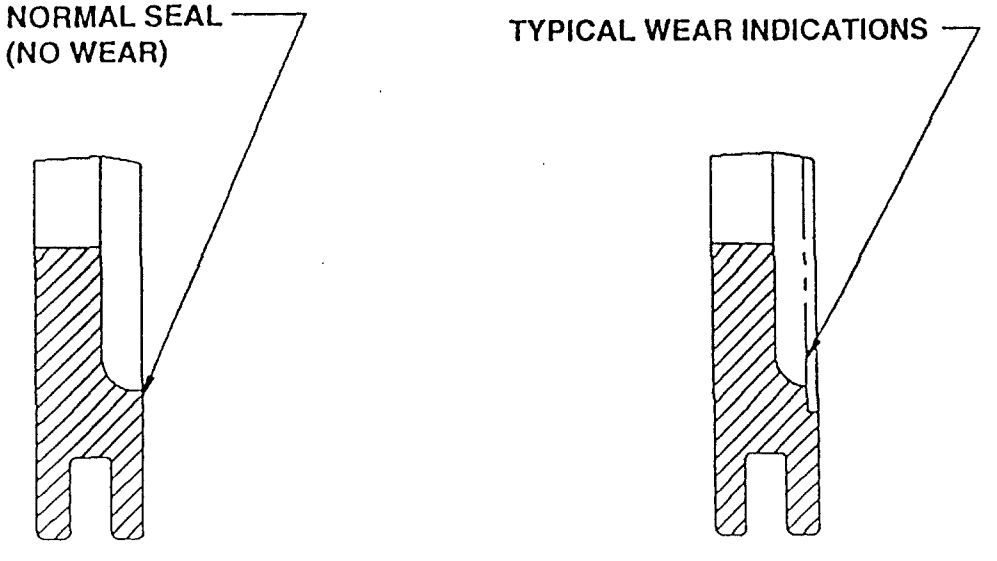


3010741-3 SEAL



DETAIL A

Wheel Bearing Outer Seal Wear

NEW

Figure 32-20-603 — Inspection of NLG Axle and Outer Bearing Seal. (Sheet 1 of 3) Ref: S/B 93-1

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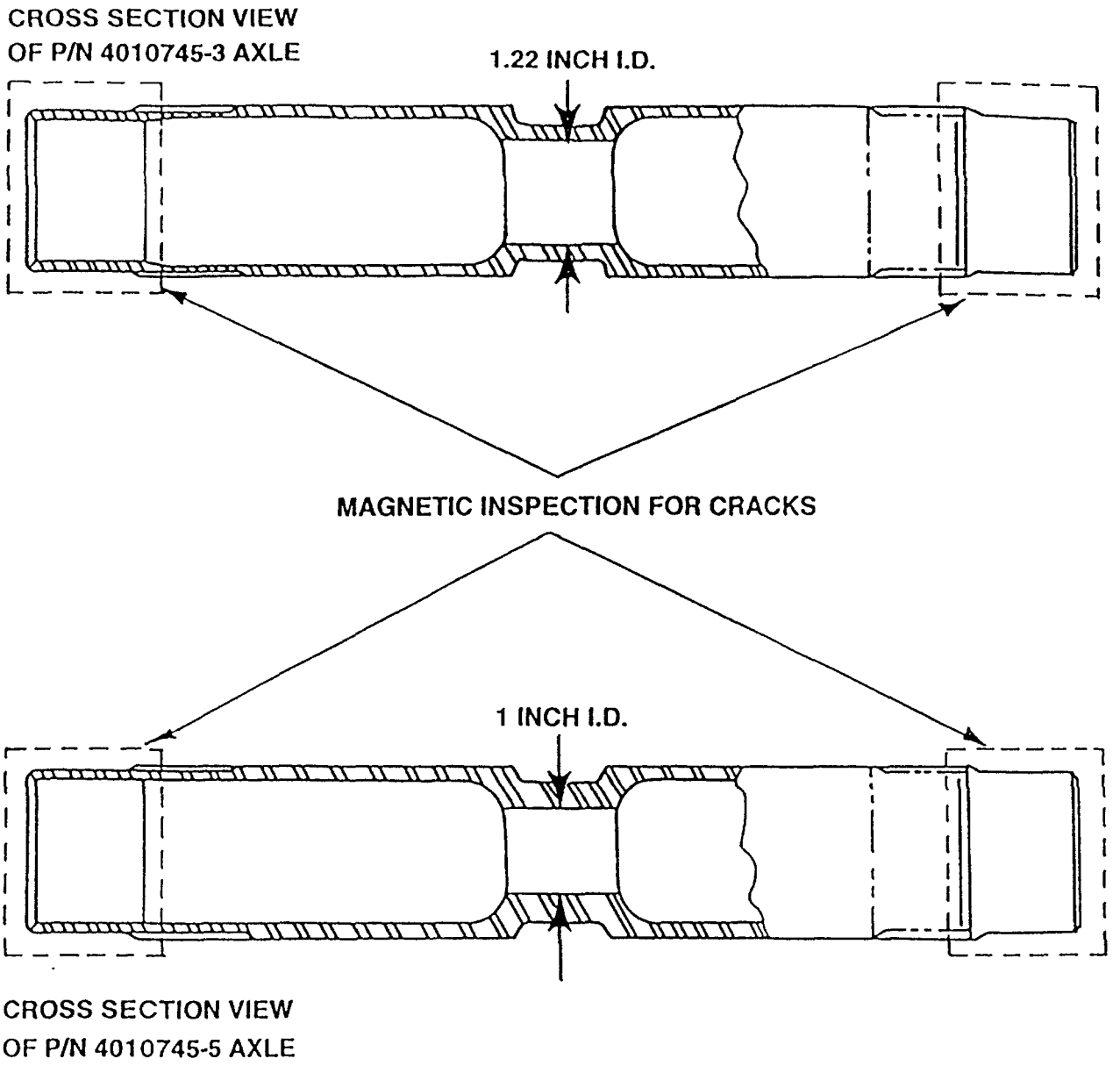

Nose Landing Gear Axle Inspection Zones

Figure 32-20-603 — Inspection of NLG Axle and Outer Bearing Seal.
(Sheet 2 of 3) Ref: S/B 93-1

NEW

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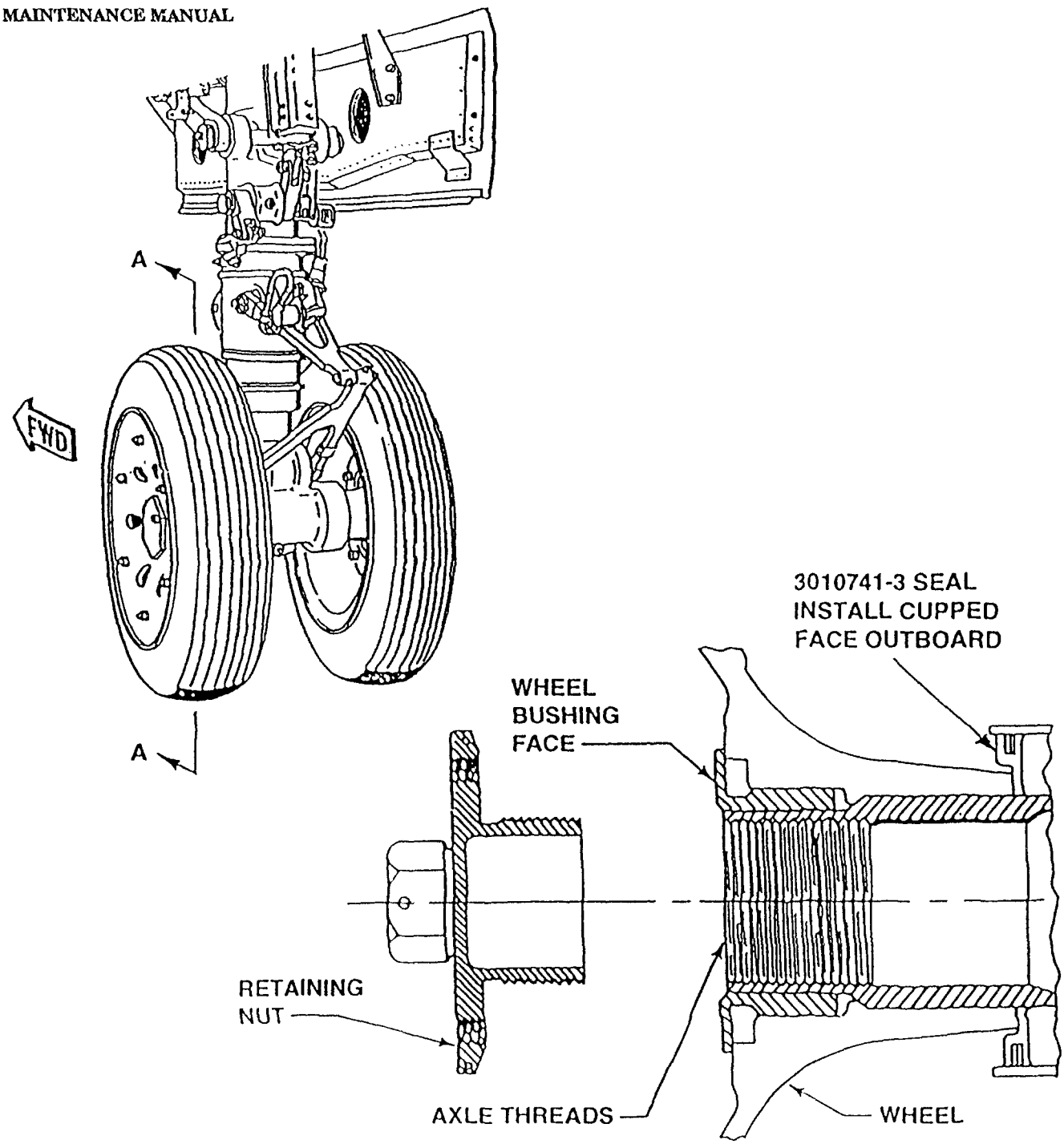

SECTION A-A
(TIRE OMITTED FOR CLARITY)

Wheel Bearing Outer Seal Installation

NEW

Figure 32-20-603 — Inspection of NLG Axle and Outer Bearing Seal.
(Sheet 3 of 3) Ref: S/B 93-1

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NOSE LANDING GEAR

WEAR LIMITS

Wear limits are listed in figure 32-20-801. This list should be used whenever an inspection or other maintenance procedures calls out a check for "adequate clearance" or "excessive wear."

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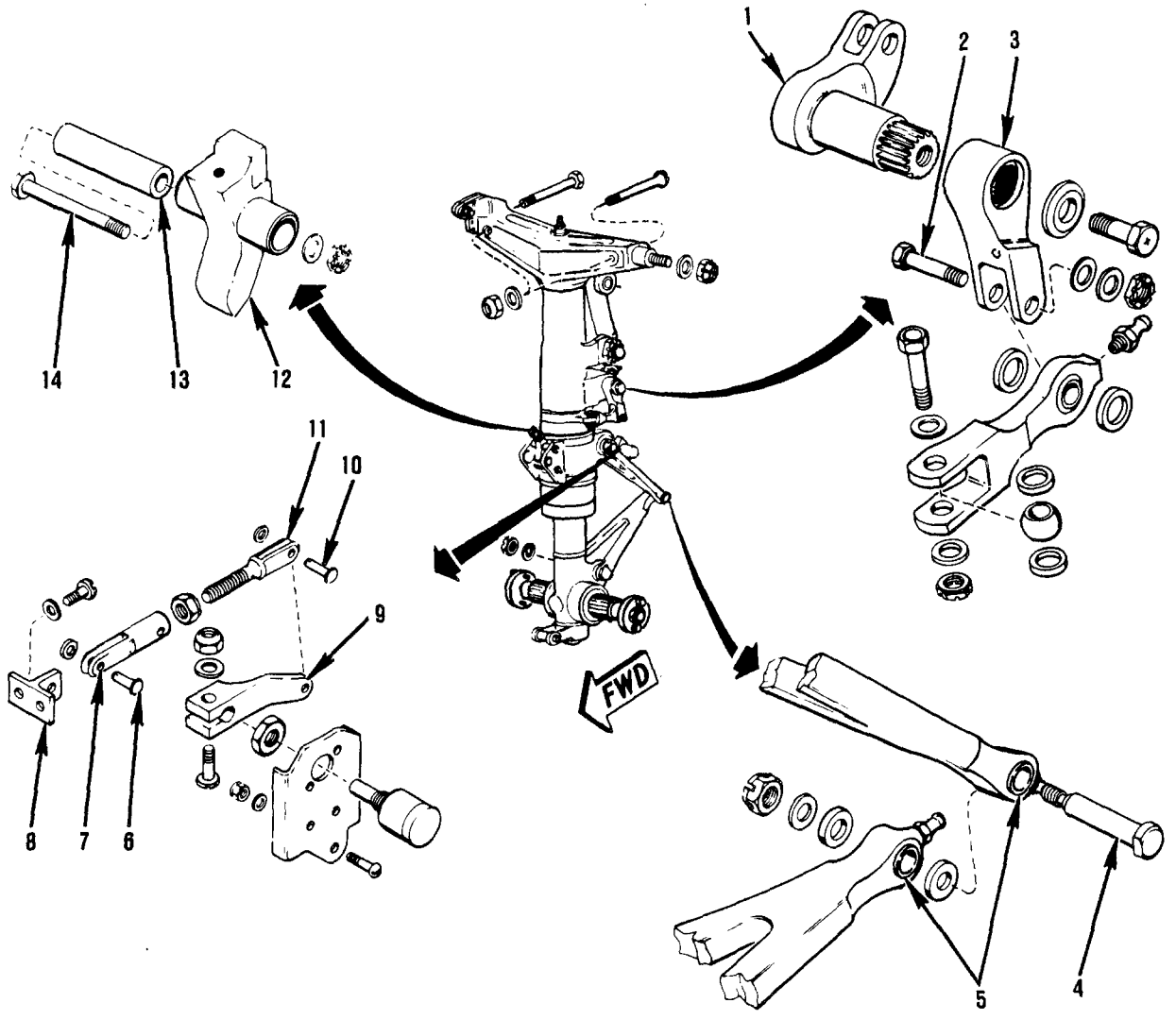

ITEMS (OD to ID fit)	DIAMETRICAL CLEARANCE (INCHES)
Bellcrank (1) OD fit to bushings (22) ID	0.0016 to 0.0031
Bolt (2) OD fit to arm (3) ID	0.0010 to 0.0027
Bolt (4) OD fit to bushing (5) ID	0.0015 to 0.0035
Pin (6) OD fit to link (7) ID	0.0000 to 0.0050
Pin (6) OD fit to bracket (8) ID	0.0000 to 0.0050
Pin (10) OD fit to arm (9) ID	0.0000 to 0.0050
Pin (10) OD fit to link (11) ID	0.0000 to 0.0050
Bushing (13) OD fit to lock (12) ID	0.0020 to 0.0040
Bolt (14) OD fit to bushing (13) ID	0.0010 to 0.0036
Bearing (16) major OD fit to barrel (21) ID	0.0035 to 0.0080
Orifice tube (17) major OD fit to piston (15) ID	0.0280 to 0.0400
Sleeve assy (20) OD fit to collar (18) ID	0.0026 to 0.0094
Sleeve assy (20) OD fit to collar (19) ID	0.0038 to 0.0082

NEW

Nose Landing Gear Wear Limits
Figure 32-20-801 (Sheet 1 of 2)

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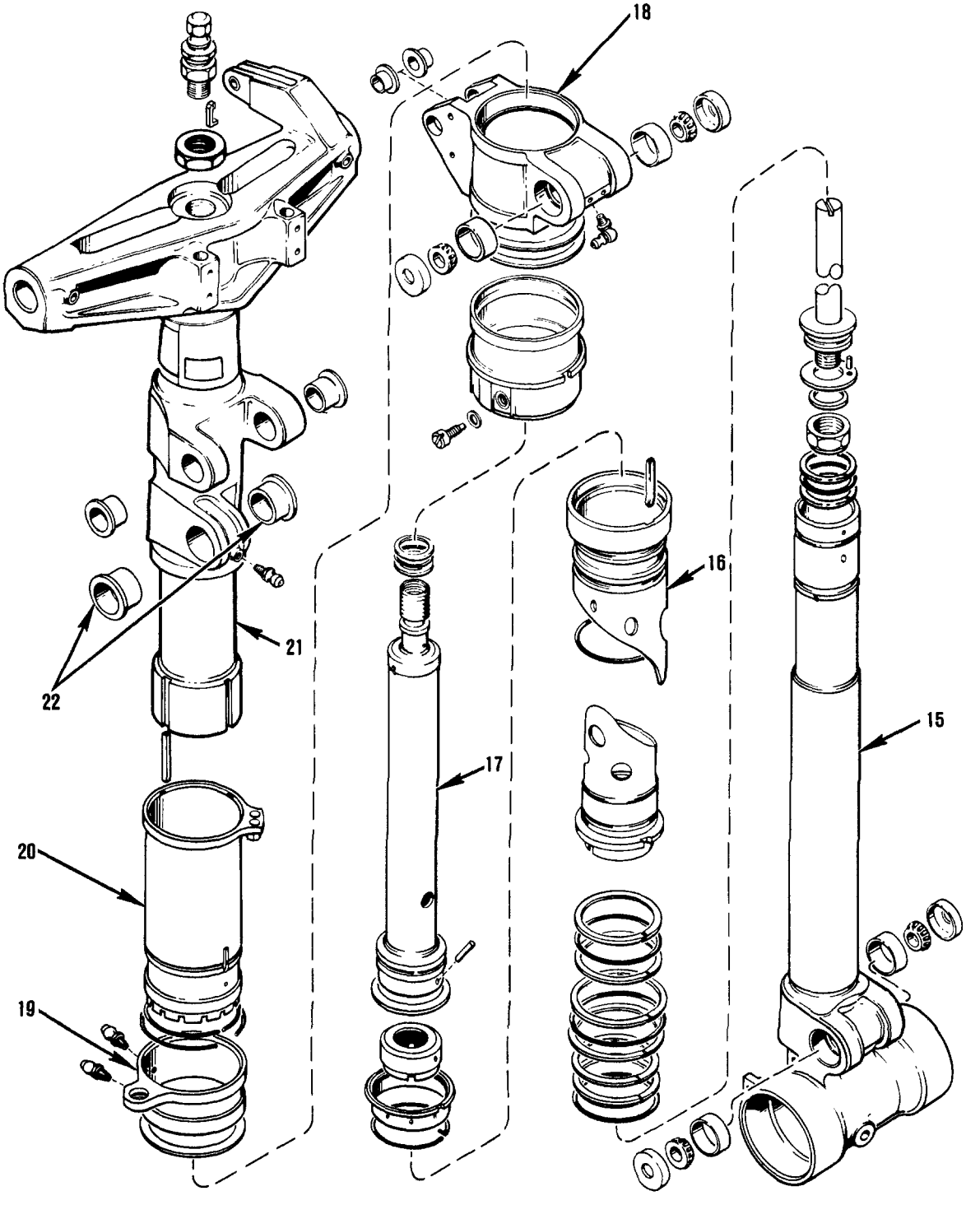

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Nose Landing Gear Wear Limits
Figure 32-20-801 (Sheet 2 of 2)

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EXTENSION AND RETRACTION

DESCRIPTION AND OPERATION

GENERAL

The extension and retraction system consists of the landing gear position selector switch, landing gear position selector valve, landing gear dump valve, mechanical sequence switches, and landing gear actuating cylinders. A cable-operated emergency extension system is provided, which manually unlocks the landing gear and landing gear strut door uplock hooks, and positions the landing gear dump valve to prevent hydraulic lock when the gear extends. Operation of the landing gear system is divided into three operations: normal retraction, normal extension, and emergency system operation.

NORMAL RETRACTION — S/N 370-2 THRU -6 NOT CHANGED BY S/B 72-6

Deleted.

NORMAL RETRACTION — S/N 370-1 AND SUBS AND 380-1 AND SUBS

With the airplane on the ground, hydraulic and electrical power on, and landing gear control handle down, the landing gear position selector valve down solenoid is energized. When the landing gear position selector valve down solenoid is energized, hydraulic pressure is routed to the extend side of the actuators in the landing gear system. In this condition, an electrical circuit through the landing gear downlock switches is completed to the three gear position indicator lights. A solenoid in the landing gear control handle, which is controlled by the left and right main gear load switches, is de-energized when the struts are compressed. The solenoid, when de-energized, engages a pin which physically prevents movement of the landing gear control handle to the up position.

When the airplane becomes airborne (main gear struts extended), the locking solenoid in the landing gear control handle retracts and the handle can be moved to the up position. A manual override is provided above the control handle which mechanically retracts the lockpin if an emergency should require gear retraction on the ground. When the control handle is moved to the up position, electrical circuits are completed through the gear uplock switches and the main landing gear strut door position switch to the red warning lights in the landing gear handle. The lights come on, indicating that the landing gear and the strut doors are not in the position selected. At the same time, a circuit is completed through the gear uplock switches to the retract solenoid of the landing gear position selector valve. Hydraulic pressure is routed to the unlock side of the gear downlock cylinders, and to the retract side of the landing gear actuating cylinders. Pressure is also routed to the closed side of the gear uplock cylinders. The main gear uplocks are held open by overcenter toggles.

As each downlock pin is pulled, the downlock switches are actuated, and the electrical circuit to the corresponding green gear position indicator light is broken, causing the light to go out. This indicates that the gear is not down and locked. If at this time the airplane altitude is below 12,000 (±600) feet, the airspeed is below 140 [1] or 150 [2] knots, and if either throttle is retarded below 95% rpm, the landing gear audible warning will sound in the headsets and the cockpit warning horn will sound. This also indicates that the gear is not down and locked. When the downlocks are retracted, the landing gear begins to retract. When the nose gear nears the up position, mechanical linkage attached to the nose wheel fairing doors is cammed by the nose gear strut toward the closed position. Near the fully retracted position, the nose gear uplock roller cams the nose gear uplock hook open. A bungee on the uplock mechanism snaps the hook under the roller when the nose gear reaches the fully retracted position. When the nose gear reaches the fully retracted position, the nose wheel fairing doors are pulled to the closed position and mechanically locked. On the main gear, the uplock roller strikes the upper cam face of the main gear uplock hook. This breaks the uplock mechanism overcenter, permitting the uplock cylinder to rotate the hook under the uplock roller to the full overcenter position. When the last landing gear reaches the full up-and-locked position, the gear uplock switches are actuated. When actuated, the gear uplock switches break the circuit to the warning light in the landing gear handle, causing the light to go out, and breaks the circuit to the landing gear position selector valve gear-up solenoid. The gear-up solenoid is de-energized, positioning the solenoid valve spool to neutral, and shutting off hydraulic pressure at the valve. All hydraulic lines to and from the landing gear system actuating cylinders are now open to return through the landing gear position selector valve.

When all the gears are up and locked, if either throttle is retarded below 95% rpm, the airplane descends below 12,000 (±600) feet, and airplane airspeed is below 140 [1] or 150 [2] knots with the gear handle in the up position, the red warning light in the gear control handle will flash on and off and an audible warning will be heard from the headsets and the cockpit warning horn. These warnings indicate that the landing gear is not down and locked.

[1] S/N 370-2 thru -6, -8 -9

[2] S/N 370-1 and -7, 380-1 and Subs

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DESCRIPTION AND OPERATION (Continued)

NORMAL EXTENSION — S/N 370-2 THRU -6 NOT CHANGED BY S/B 72-6

Deleted.

NORMAL EXTENSION — S/N 370-1 AND SUBS AND 380-1 AND SUBS

When the landing gear control handle is moved down, a circuit to the red warning lights in the landing gear control handle is completed and the lights come on. If at this time, either throttle is retarded below 95% rpm, if the airspeed is below 140 [1] or 150 [2] knots, and if the altitude is below 12,000 (±600) feet, an audible warning will be heard in the headsets and from the cockpit warning horn. The audible warning will continue until all gears are down and locked, until the warning horn cutout button in the cockpit is depressed (cutout button inoperative when flaps are extended beyond 80%), or until both throttles are advanced to 95% rpm. When the circuit to the gear-down solenoid in the landing gear position selector valve is completed, the solenoid is energized. Hydraulic pressure is routed to the unlock side of all gear uplock cylinders, to the lock side of all gear downlock cylinders, and to the gear-down side of all gear actuating cylinders. Hydraulic pressure applied to the main gear uplock hook actuating cylinders overcomes the overcenter lock mechanism and rotates the hooks to the unlocked position unlocking the strut fairing doors. Hydraulic pressure, when applied to the nose gear lock actuating cylinder, disengages the uplock hook, mechanically unlocking the nose gear fairing doors. All gears are now released from their uplock hooks. This permits hydraulic pressure to extend the gear.

As the nose gear is extended, the nose gear fairing doors are mechanically opened to the full open overcenter position by the nose gear strut. When the nose gear nears the down position, the twin downlock pins ride circular cams on the unfolding drag brace. When the gear is fully down, the lockpins are seated into the locked position by springs. When the nose gear is down and locked, an electrical circuit is completed to the nose gear position indicator light, causing the light to come on, indicating that the nose gear is down and locked. When the main gear nears the down and locked position, the side-brace ends cam the downlock pins open. As the gear reaches the full down position, the spring-loaded downlock pins seat into receivers in the side brace and lock the gear down. With each main gear in the down and locked position, an electrical circuit is completed to each main gear position indicating light, causing the light to come on, indicating that the gear is down and locked. The electrical circuits to the red warning lights in the landing gear control handle and to the audible warning system are opened. This causes the red lights in the handle to go out and the warning horn and headsets audible signal to stop, indicating that all landing gears are down and locked. Hydraulic pressure remains applied to the down side of all gear actuating cylinders and to the lock side of all downlock cylinders, as long as the landing gear control handle is down and electrical and hydraulic power is applied.

EMERGENCY EXTENSION

To extend the gear using the emergency extension system, the landing gear control handle is placed at down and the landing gear emergency release handle at the left side of the cockpit below the instrument panel is pulled. When the emergency release handle is pulled, the normal electrical circuitry to the landing gear handle is broken, and fairing door uplock hooks are released. The landing gear dump valve provides a path from the actuating cylinders for return fluids to the return lines, preventing a hydraulic lock. The landing gears are extended to the down and locked position by gravity; the nose gear is aided to the down and locked position by the air-stream. The L.G. ELEC RESET switch, on the instrument panel below the landing gear emergency release handle, must be reset after an emergency extension is performed. The reset switch is used to return the landing circuitry to normal sequence of operation. If the reset switch is not reset, the landing gear will not retract when the next normal operation of the landing gear is attempted.

It is possible to retract the landing gear while the airplane is on the ground, if required during emergency. This is accomplished by pushing in the landing gear downlock override button above the landing gear control handle and moving the control handle up in the normal manner. Depressing the override button manually disengages a pin which is spring-loaded to a position that holds the landing gear handle down. During normal operation, the pin is retracted by a solenoid when the landing gear struts are all extended and the landing gear load switches are actuated.

AUXILIARY ACCUMULATOR POWER ASSIST RETRACTION

On S/N 370-1, -7, 380-1 and subs and 370-2 thru -6, -8 and -9 changed by SOK 11, the auxiliary system shut off valve is automatically opened (power removed from the valve) in the event of a hydraulic pump failure at take-off during landing gear retraction (see Figures 32-00-3, -4 and -5). In the event of hydraulic pump failure during takeoff and the landing gear system in the retraction cycle, when the normal system hydraulic pressure drops to 2000 (±100) psi, electrical power is routed through the hydraulic low pressure switch to the coil of the hydraulic low pressure monitor relay (K780), which is then energized. At the same time, electrical power is routed through the RH and LH landing gear uplock switches (S596 and S594),

[1] S/N 370-2 thru -6, -8 -9

[2] S/N 370-1 and -7, 380-1 and Subs

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DESCRIPTION AND OPERATION (Continued)

AUXILIARY ACCUMULATOR POWER ASSIST RETRACTION (Cont'd)

through the energized closed contacts of relay K780 to the coil of the landing gear valve (up) monitor relay K781. Relay K781 energizes which in turn removes electrical power from the auxiliary system shutoff valve allowing the valve to open which releases the hydraulic pressure stored in the auxiliary accumulator into the normal hydraulic system thereby completely retracting the main landing gear. As soon as the RH and LH landing gear uplocks are actuated, electrical power is removed from relay K781, which then de-energizes and applies electrical power to the auxiliary system shutoff valve. The shutoff valve is then closed and traps the remaining auxiliary system pressure in the auxiliary system accumulator.

LANDING GEAR POSITION SELECTOR SWITCH

The landing gear position selector is mounted in a panel below the instrument panel and to the left of the center pedestal. The switch consists of mechanically actuated electrical contacts controlled by movement of the landing gear control handle. A safety feature within the selector switch prevents movement of the landing gear control handle to the up position if the airplane is not air-borne. This is accomplished by a spring-loaded solenoid-operated pin extending in the path of the control handle when it is moved toward the up position. The pin is electrically pulled through action of the solenoid when the airplane is air-borne. A landing gear downlock override button, on the face of the landing gear control panel, provides a means of manually retracting the pin if it is necessary to retract the landing gear before the airplane is air-borne. A mechanical lever attached to the landing gear control handle mechanically locks the landing gear control handle when it is moved to the up position. The plastic landing gear control handle contains two landing gear position warning lights that come on when the landing gear is in an unsafe position. On NA-370 airplanes changed by S/B 72-6 and NA-380-1 and subs the red warning lights in the landing gear control handle will also come on if the main gear strut doors are in an unsafe condition.

LANDING GEAR POSITION SELECTOR VALVE

The landing gear position selector valve is mounted on a bracket on the left side of the nose wheel well. (See Figure 32-00-1). The valve is a four-way valve having four hydraulic ports which are electrically controlled by solenoids at opposite ends of the valve. When the landing gear control handle is moved, an electrical circuit to one of the solenoids is completed. This causes a spring-loaded pilot valve spool between the solenoids to be pulled toward the energized solenoid. Necessary ports are then connected to allow entry of hydraulic fluid under pressure from the hydraulic system. Fluid is then directed to the end of the spring-loaded secondary spool, repositioning the secondary spool. This connects the necessary ports to direct hydraulic pressure to retract the landing gear. Return fluid is routed through the properly positioned spools in the valve to a system return line. When all gears and doors are up and locked, the solenoid is de-energized. This allows springs at each end of the valve spools to return the spools to the neutral position, blocking hydraulic pressure at the entry to the valve. Movement of the landing gear control handle down energizes the opposite solenoid, causing the valve spools to slide in the opposite direction. Hydraulic pressure is then routed to extend the landing gear.

During emergency extension of the landing gear, the landing gear position selector valve is bypassed by a manually operated dump valve. Actuation of the landing gear emergency release handle positions the dump valve to allow hydraulic fluid, on the retraction side of the actuating cylinders, to be routed to the return lines. (See Figure 32-00-3.)

LANDING GEAR DUMP VALVE

The landing gear dump valve is in the left forward side of the nose wheel well at FS 76. The dump valve operates in a 3000 psig hydraulic system and is normally in the closed position. The dump valve is a two-position, manually operated, three-port valve. The dump valve provides an emergency means of lowering the landing gear in case of hydraulic or electrical failure. The dump valve is connected to the landing gear emergency release cable (Figure 32-00-2) and is operated to the dump position when the release cable is pulled. When the dump valve is moved to the dump position, it provides a path for return fluid from the landing gear.

INFLATABLE WHEEL SEAL PRESSURE REGULATOR MANUAL DUMP AND RELIEF VALVE S/N 370-1 THRU -9 NOT CHANGED BY S/B 72-8

Deleted.

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EXTENSION AND RETRACTION

MAINTENANCE PRACTICES

REPLACEMENT OF LANDING GEAR POSITION SELECTOR VALVE

Removing Position Selector Valve

1. Remove electrical and hydraulic power.
2. Disconnect electrical plug at top of valve.
3. Disconnect and cap valve hydraulic lines.
4. Remove valve attach screws and remove valve.

Installing Position Selector Valve

1. Position valve on mounting bracket. Install and tighten valve attach screws.
2. Uncap, connect, and tighten valve hydraulic lines.
3. Connect valve electrical disconnect plug.
4. Check normal retraction and normal and emergency extension of landing gear. Check valve and line connections for evidence of hydraulic leaks.

REPLACEMENT OF LANDING GEAR POSITION SELECTOR SWITCH

Removing Landing Gear Position Selector Switch

1. Remove electrical and hydraulic power.
2. Disconnect switch electrical plug forward of center pedestal.

NOTE

The outboard end of the aileron control rod on the pilot's control column should be disconnected to provide clearance to remove the switch. Do not change rod adjustments.

3. Remove screws securing switch to panel and remove switch.

Installing Landing Gear Position Selector Switch

1. Position switch on panel, and install and tighten switch mounting screws.
2. Reconnect aileron control rod on pilot's control column.
3. Connect switch electrical disconnect plug.
4. Check operation of landing gear and landing gear position selector switch.

REPLACEMENT OF LANDING GEAR DUMP VALVE

Removing Landing Gear Dump Valve

1. Remove electrical and hydraulic power.
2. Disconnect nose gear section of emergency release cable at both ends.

NOTE

Do not disconnect emergency release cable at turnbuckles, unless rigging procedures are being used.

3. Disconnect hydraulic lines. Cap lines and fittings on valve.
4. Remove screws, and lower bracket assembly about two inches.
5. Remove cotter pin and bolt that attach cage (on emergency release cable) to dump valve arm.
6. Remove dump valve arm from dump valve. (Mark arm and dump valve for reinstallation.)
7. Remove fittings from valve. Note location for reinstallation.
8. Remove dump valve from bracket.
9. Install dust plugs in valve ports.

Installing Landing Gear Dump Valve

1. Carefully inspect previously used parts before installing on landing gear dump valve.
2. Install dump valve on bracket.
3. Install fittings on dump valve. Do not tighten jam nuts at this time.
4. Install dump valve arm on dump valve.
5. Position dump valve on bracket to approximate location.
6. Install emergency release cable cage to dump valve arm.
7. Install dump valve and bracket assembly to bulkhead.
8. Connect nose gear section of emergency release cable at both ends.
9. Connect hydraulic lines. Tighten jam nuts on fittings.

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MAINTENANCE PRACTICES (Continued)

ADJUSTING LANDING GEAR EMERGENCY RELEASE SYSTEM

See Figure 32-30-401.

CHECKING OPERATION OF LANDING GEAR EMERGENCY SYSTEM

Checking Emergency Extension Cycle

Equipment/Tools

- Hydraulic test stand capable of delivering 4 gpm at 3000 psi
- Set of jacks (three, 5-ton minimum)
- Jacking and mooring kit
- 28 VDC external electrical power
- Deleted
- A push-pull scale capable of measuring more than 55 pounds

WARNING

When working in or around wheel well areas, use ground maintenance locks on MLG doors and MLG actuators and pin the nose gear down to prevent inadvertant retraction. Failure to do this could result in serious injury or death.

1. Perform steps of General Procedure Before Landing Gear System Operation. (Refer to 32-00-3.)
2. Engage (push in) the following circuit breakers:
   - LANDING GEAR — CONT
   - LANDING GEAR — POS
3. Start hydraulic test stand. Regulate hydraulic test stand pressure flow to 4 gpm. Increase hydraulic pressure to 3000 psi. Move landing gear control handle up and retract landing gear.
4. With landing gear up and locked, shut off electrical and hydraulic power.
5. Attach a push-pull scale to landing gear emergency release handle. Pull scale straight aft. The force applied to the landing gear emergency release handle should not exceed 40 pounds. All gears should extend. An external aft force of 20 pounds maximum at the centerline of the strut on the axle is permissible to lock the nose gear. An external side force of 18 pounds maximum at the axle hub is permissible to lock the main gear.
6. Push in L.G. ELEC RESET switch and reapply electrical and hydraulic power. The landing gear should retract.
7. Place landing gear control handle at DOWN. The landing gear should extend to the down-and-locked position.
8. Make at least one complete normal cycle of landing gear.

Checking Emergency Retraction

Emergency retraction of the landing gear is identical to the normal retraction cycle, with the exception that the LDG GEAR — DOWNLOCK OVERRIDE button on the landing gear position selector switch is pushed in; therefore, the emergency retraction check consists of checking operation of the landing gear downlock override switch. To make this check proceed as follows:

1. Remove hydraulic pressure.
2. Place jacks under airplane and raise jacks until landing gear clear ground.
3. Disconnect main gear load switches to simulate a ground load condition. Ensure that the LDG GEAR handle cannot be raised.
4. Position the LH load switch to the air (unloaded) position and the RH load switch to the ground (loaded) position.
5. Check that the LDG GEAR control handle cannot be moved to the UP position (with slight pressure) without depressing the override button.
6. Position the LH load switch to the ground (loaded) position and the RH load switch to the air (unloaded) position.
7. Check that the LDG GEAR control handle cannot be moved to the UP position (with slight pressure) without depressing the override button.
8. Position both LH and RH load switches to the ground (loaded) position.
9. Check that the LDG GEAR control handle cannot be moved to the UP position (with slight pressure) without depressing the override button.
10. Manually push in and hold LDG GEAR — DOWNLOCK OVERRIDE button on landing gear control panel.
11. Check that landing gear control handle can be moved to the up position.
12. Release LDG GEAR — DOWNLOCK OVERRIDE button and move landing gear control handle down. Ensure that the LDG GEAR handle cannot be raised.
13. Reconnect MLG load switches.
14. Lower the airplane and remove jacks.

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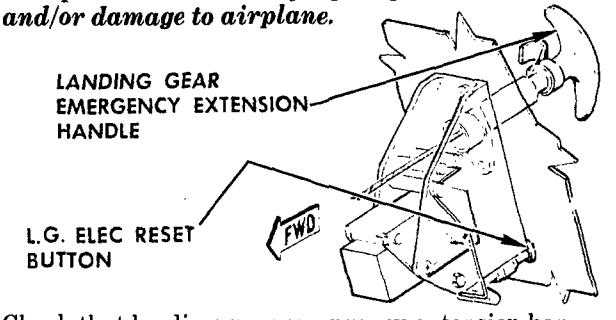
Equipment/Tools

- Set of jacks (three, 5-ton minimum)
- Jacking and mooring kit
- Hydraulic test stand capable of delivering 8 gpm at 3000 psi
- 28 VDC external electrical power
- Continuity light
- Push-pull scale capable of measuring 50 pounds

WARNING

Before applying landing gear hydraulic pressure for any reason (checks, adjustments, etc.), delete normal and auxiliary accumulator hydraulic pressures. This will prevent the auxiliary accumulator power assist retraction system to operate and prevent serious injury to personnel and/or damage to airplane.

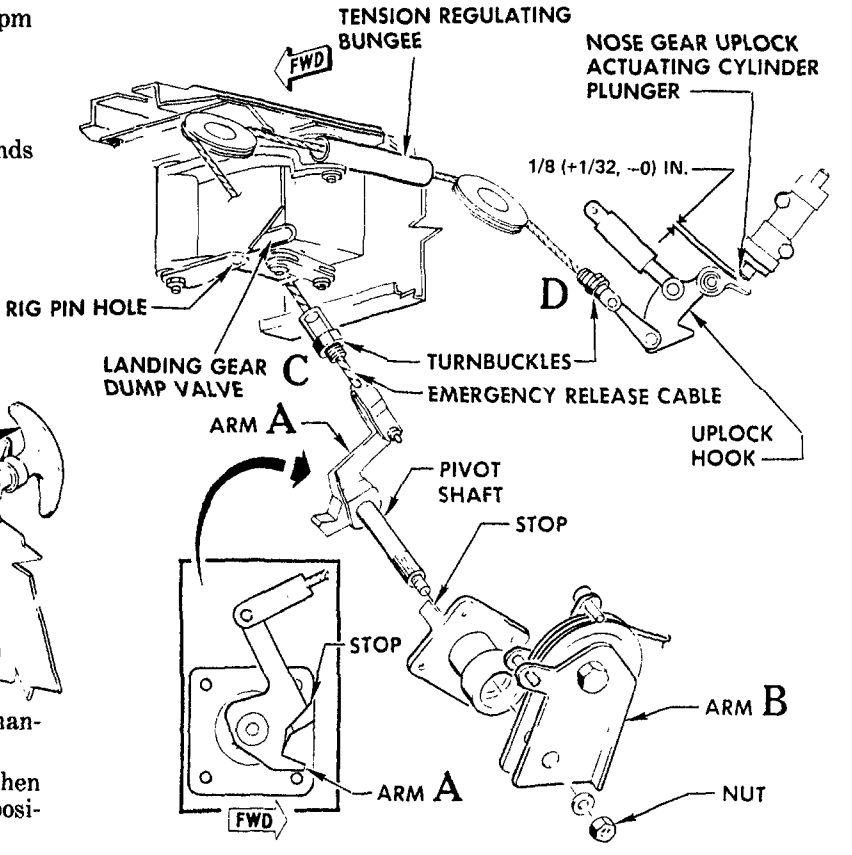


1. Check that landing gear emergency extension handle is in full-forward stowed position.
2. Check that arm (A) is against stop as shown; then remove access F11 and check that arm (B) is positioned as shown.

NOTE

Before any adjustment of arms, loosen turnbuckle at station 262 on left side of airplane.

3. Recheck arm positions. If arms are not aligned properly, remove nut from shaft. Slide arm (A) inboard, freeing serrations on pivot shaft. Rotate arm (B) relative to arm (A) to obtain required angular (12-3/4 (±2) degrees) relationship. Slide arm (A) outboard, engaging pivot shaft in serrations of arm (B), and install nut.
4. With arm (A) against surface of stop, adjust turnbuckle (C) to provide a "drop fit" with rig pin in landing gear dump valve arm. Double safety turnbuckle with 0.040 inch diameter lockwire.



NOTE

Main gear must be retracted with nose gear pinned down and hydraulic pressure on to do step 5.

5. Pin nose gear down and place gear handle up to retract main gear.
6. Check that uplock cylinder is fully retracted, then adjust turnbuckle "D" to obtain 1/8 (+ 1/32, -0) inch between top edge of gear uplock hook and tip of gear uplock actuating cylinder piston (hydraulic pressure applied).

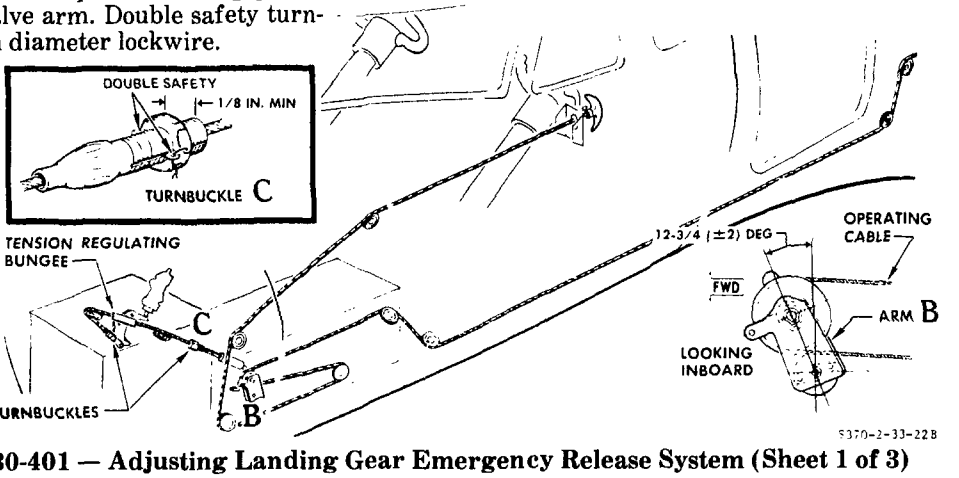


Figure 32-30-401 — Adjusting Landing Gear Emergency Release System (Sheet 1 of 3)

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7. Adjust turnbuckle at station 262 on left side of fuselage to position pulley arm attached to return spring 0.005 to 0.015 inch away from pulley arm stop. Cable tension should read 13 (±1.5) pounds. Secure turnbuckle using 0.040 safetywire.
8. Adjust individual cable to left and right main gear uplock hooks to position arm (P/N 265-335566-1) 0.005 to 0.015 inch from its stop. Secure turnbuckle using 0.040 safetywire.

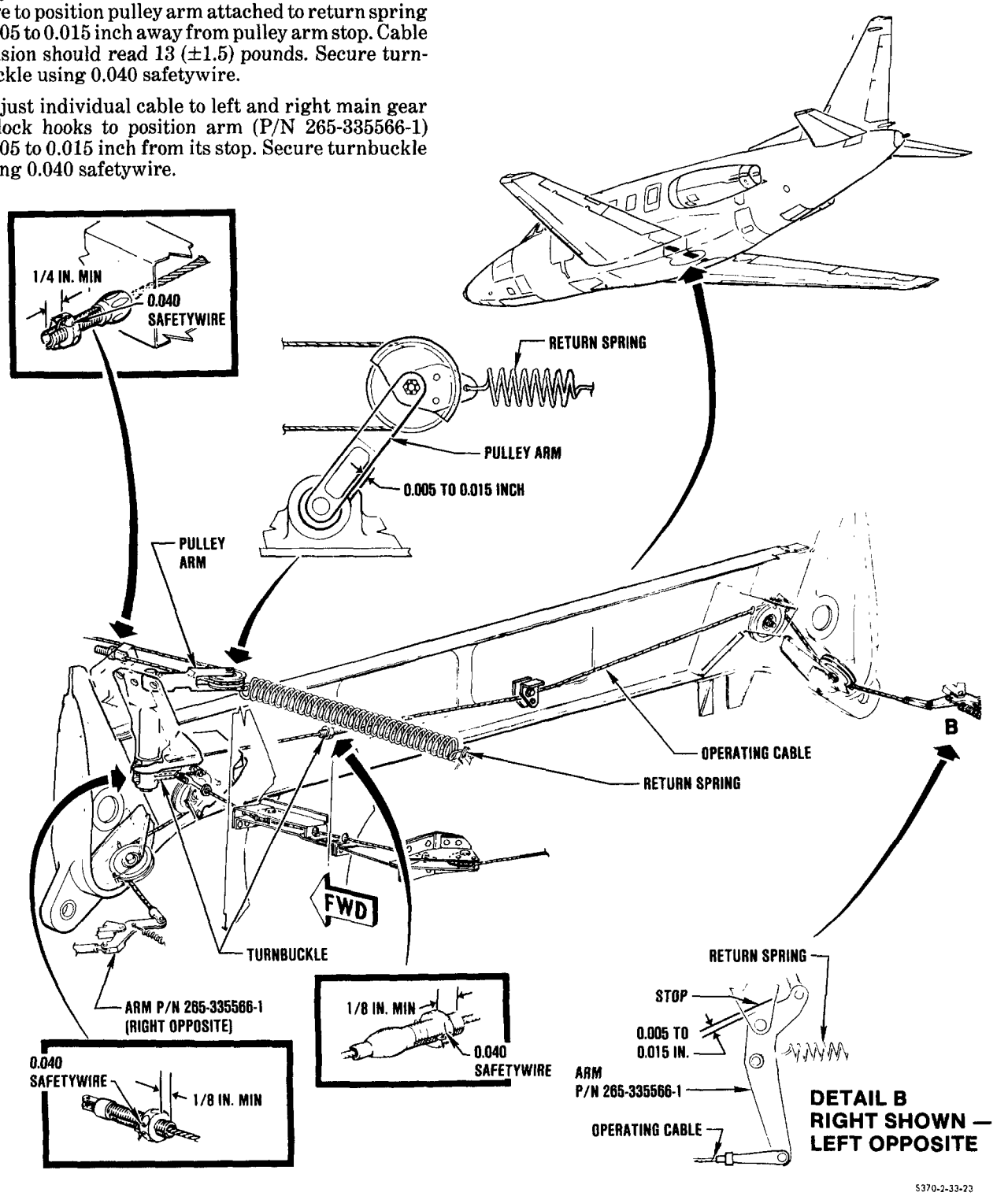

Figure 32-30-401 — Adjusting Landing Gear Emergency Release System (Sheet 2 of 3)

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9. Deleted.

10. Working forward of left instrument panel, loosen landing gear emergency and steering switch attach screws. Adjust position of switch so that electrical circuit to landing gear position switch is closed. Check adjustment of landing gear emergency and steering switch by connecting a continuity light to switch terminals. The continuity light should come on when the switch is correctly adjusted. Remove continuity light and check that switch attach screws are tight.

NOTE

Whenever the emergency gear release system is actuated, the landing gear electrical reset button will have to be depressed to return the gear circuitry to a normal condition.

11. Check all cables, bell cranks, and linkage for installation of all safeties (0.040 safetywire).

12. Check normal and emergency operation of landing gear system.

13. Check that force to operate landing gear emergency release handle does not exceed 55 pounds.

Figure 32-30-401 — Adjusting Landing Gear Emergency Release System (Sheet 3 of 3)

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WHEELS AND BRAKES

DESCRIPTION AND OPERATION

GENERAL

The main and nose landing gear wheels are of the split type, held together with bolts. These bolts must be removed for installation or removal of tires. The dual main wheels and dual nose wheels are equipped with extra-high-pressure tubeless tires. The wheel brake system consists of the self-adjusting disk brake assemblies, an anti-skid system, a brake control system, and the normal and emergency hydraulic power systems.

The brake assemblies are mounted on the inboard side of each main gear wheel assembly. The brakes are applied in the conventional manner, by pressure to the toe portion of the rudder pedals. The rudder pedals are connected by a mechanical linkage to the brake control valves. The mechanical linkage actuates two combination manual-power brake valves, each of which controls the brake pressure to one pair of brake assemblies. Hydraulic pressure for the brake system is supplied by the hydraulic power system. An emergency brake system can be set into operation by actuation of a Teleflex-controlled emergency brake control knob in the cockpit. Actuation of the emergency brake control knob provides brake applications through a separate set of brake lines. The brakes may be left in a parked condition (brakes applied) by actuating the parking brake control handle, in the cockpit, while the rudder pedals are held depressed. The parking brake is released by momentarily depressing the rudder pedals.

NOSE WHEELS

Dual wheels are mounted on the nose gear strut axle. The wheels are of the split type, consisting of two cast-magnesium sections bolted together. The wheels are Type VII, 18 by 4.4 inches. The tires are of the 18 by 4.4-inch extra-high-pressure tubeless type. The two halves of the wheel must be separated for installation or removal of the tires.

MAIN WHEELS

The main wheel assembly is a 22 × 5.75-12, forged aluminum wheel designed for use with tubeless tires. The wheel is the divided type with the wheel halves (subassemblies) held together with bolts, washers, and self-locking nuts. A plastic hub spacer is installed between the wheel hubs to prevent fretting on the hub mating surfaces.

An O-ring packing, mounted in a groove in the inboard wheel half, seals the wheel at the juncture of the two halves. A valve subassembly installed in the inboard wheel half provides for tire inflation and deflation.

The inboard half of the wheel contains five hardened steel disk drive keys and three fusible plugs. The drive keys engage the keyways in the rotating disks of the brake and transmit the brake torque to the wheel and tire. The fusible plugs release the tire pressure if the wheel becomes overheated.

The drive cap (hubcap) contains a drive tube that engages and drives the wheel-driven unit of the anti-skid control system.

BRAKE ASSEMBLIES

The brake is a multiple-disk assembly consisting principally of a brake housing, a torque tube, a disk stack, and backplate. The torque tube is bolted between the housing and backplate, with the disk stack installed on the torque tube.

The disk stack consists of a pressure plate installed nearest the housing, two rotating disks, and one stationary disk. The stationary disk and pressure plate are keyed to the torque tube; the rotating disks are keyed to, and rotate with, the wheel.

Steel wear pads, attached with rivets, serve as friction surfaces of the pressure plate, backplate, and stationary disk. A sintered friction material is bonded to the wear surfaces of the rotating disks.

Six cavities are provided in the brake housing for the brake pistons. The cavities are interconnected by drilled fluid passages, and each cavity contains a groove for an O-ring packing and backup ring. The pistons are made of a phenolic-asbestos compound, which prevents heat transfer from the disk stack to the brake housing.

The housing also contains six return mechanisms, which are connected to the pressure plate with return pins. A grip and tube subassembly within each return mechanism automatically adjusts as the brake disks wear, thus maintaining a constant operating clearance.

The brake housing contains two inlet ports and one bleeder port. The arrangement of these ports permits the brake to be used in both right- and left-hand positions on the airplane.

The brake is actuated by hydraulic pressure moving the pistons against the pressure plate, which in turn forces the rotating and stationary disks to slide on the torque tube until compressed against the backplate. The rotating disks, keyed to the main wheel, rotate in friction contact with the steel wear pads of the pressure plate, stationary disk, and backplate.

When hydraulic pressure is reduced or released, the return spring mechanisms in the brake housing pull the pressure plate toward the housing and away from the brake disks. The pistons are retracted into piston cavities by action of the pressure plate.

As the brake disks wear, the return pins and tubes in the return mechanisms are advanced farther through the grips by the forward movement of the pressure plate. Since the amount of return is built into the return mechanisms, a constant operation clearance is maintained.

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DESCRIPTION AND OPERATION (Continued)

BRAKE CONTROL VALVES

The brake control valves are combination manual-power metering-type units operated by actuation of the rudder pedals. Each rudder pedal is connected to its respective valve by mechanical linkage. Each valve controls the brake pressure to its relative pair of brake units at the wheels. Hydraulic pressure, normally at 3000 psi, is supplied to the brake control valves. The brake control valves are mounted on a bulkhead in the nose wheel well above the fairing door lock mechanism. (See Figure 32-40-1.)

If normal hydraulic system pressure or auxiliary hydraulic system pressure is not available, the brake control valve can be changed to operate as a manual brake control valve by actuation of the emergency brake control knob in the cockpit. This changes the valve port connections within the valve. Hydraulic fluid obtained from the brake reservoir is forced by a master cylinder through only a portion of the brake control valve, to the brake assembly, bypassing the normal metering valve and the parking brake valve. When the manual portion of the brake control valve is used, it is necessary to pump the rudder pedals a few times to generate enough pressure to effectively apply the brakes. The brakes are released in the same manner as they are released when normal hydraulic system pressure or auxiliary hydraulic system pressure is used, except that the return fluid flows through the same passages in the brake control valve and back to the brake reservoir as when the brakes were applied.

PARKING BRAKE CONTROL VALVE

The parking brake control valve, in the nose wheel well, is a mechanically actuated valve that permits the brakes to be held on, once they are applied. (See Figure 32-40-1.) The valve is controlled by a Teleflex cable and a knob in the cockpit. When the parking brake control knob is actuated, the control valve is positioned to trap hydraulic fluid under pressure in the brake assemblies. Two spring-loaded compensators keep a constant pressure on the trapped fluid, when the brakes are parked, and allow for thermal expansion of the fluid. The valve arm is spring-loaded to the normal position, and is locked at on position by a hydraulically operated lockpin. Application of brake pressure, through the rudder pedals, retracts the lockpin and allows the valve arm to return to the normal position.

BRAKE RESERVOIR

The twin cylindrical-shaped brake reservoir, constructed of aluminum alloy, can supply about 15 cubic inches of hydraulic fluid to the brake control valves. The brake reservoir is attached to a bracket at the top aft end of the nose wheel well. Four lines welded to the bodies of the reservoir provide a means of connecting the reservoir to the hydraulic system: Three of the lines indirectly route fluid to the control valves; one of the lines returns fluid through a separate line to the hydraulic system reservoir. Pressurization of the hydraulic system reservoir during engine or hydraulic test stand operation supplies hydraulic fluid to the brake reservoir at a pressure of about 35 psi. This ensures enough hydraulic fluid to operate the brake control valve. Of the three lines indirectly routing fluid to the brake control valves, two supply hydraulic fluid to the brake control valves, and a third returns from the brake control valve and anti-skid control valves to the brake reservoir.

HYDRAULIC SHUTTLE VALVES

The wheel brake system contains four hydraulic shuttle valves, one attached to the inboard side of each brake assembly. (See Figure 32-40-2.) The shuttle valves are pressure-operated units that direct normal, auxiliary, or emergency hydraulic pressure to the brake assembly. Each shuttle valve is hydraulically connected to the brake control valve through parallel hydraulic lines. One line, connected to the parking brake valve and thence to the brake control valve, supplies either auxiliary or normal pressure to the brake. The other line is connected directly to the brake control valve and supplies emergency hydraulic pressure to the brake. This provides a parallel source of hydraulic pressure to the brake assembly. Hydraulic pressure is admitted to the brake assembly from the shuttle valve through a common manifold (swivel bolt), also serving as the shuttle valve attach point to the brake assembly.

AUTOMATIC WHEEL STOPPER CYLINDERS — NA-370 AIRPLANES NOT CHANGED BY S/B 72-8

Deleted.

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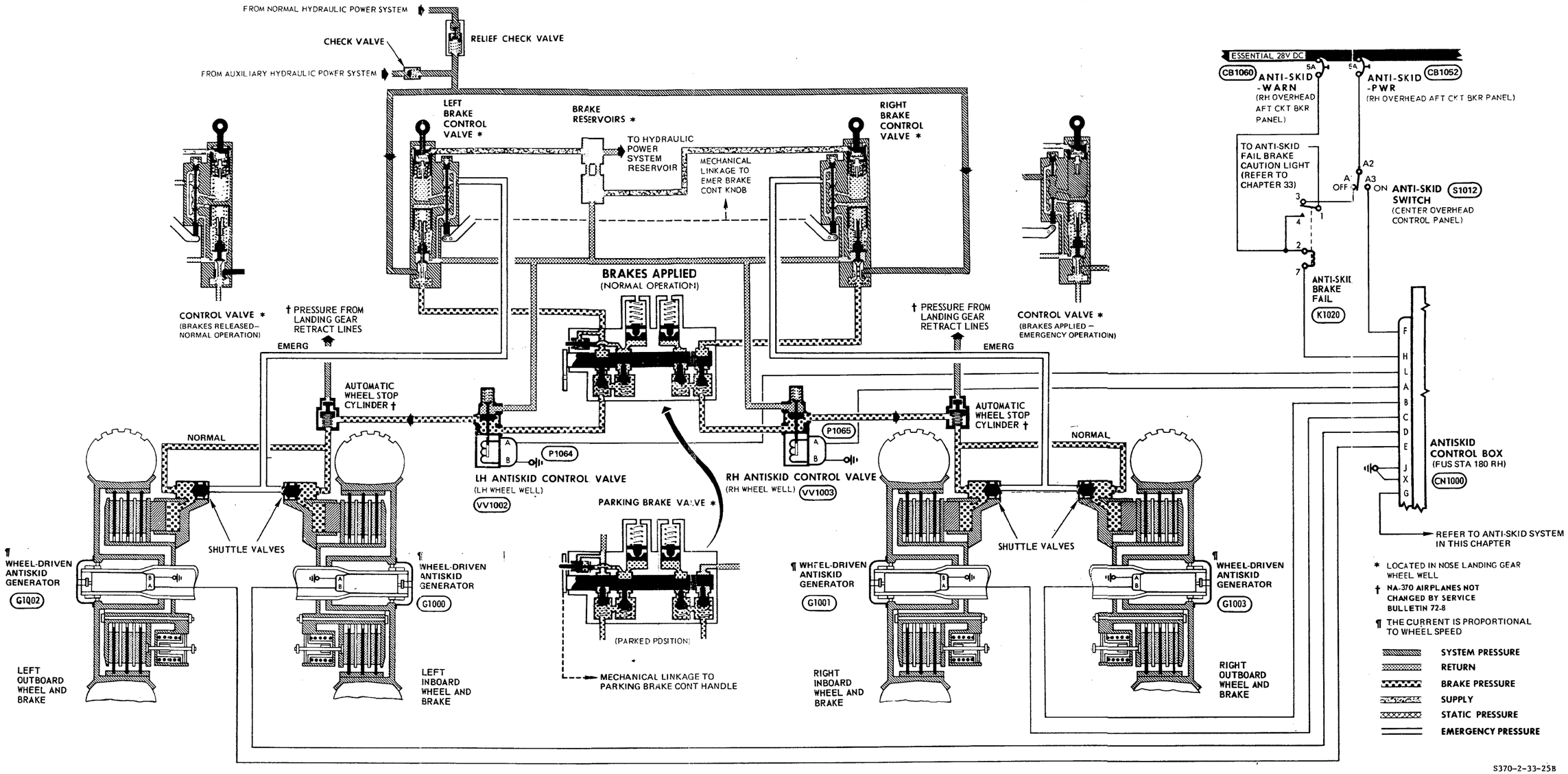
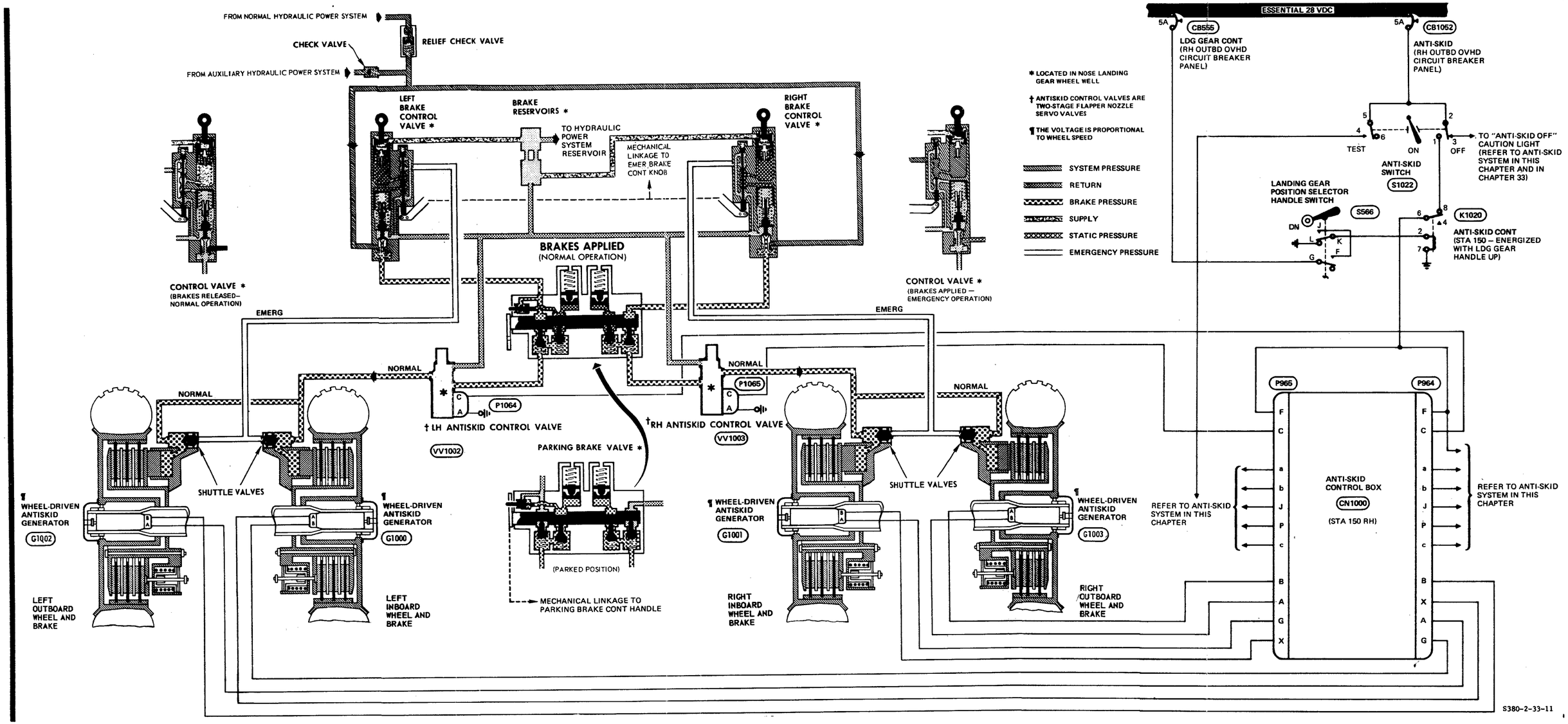
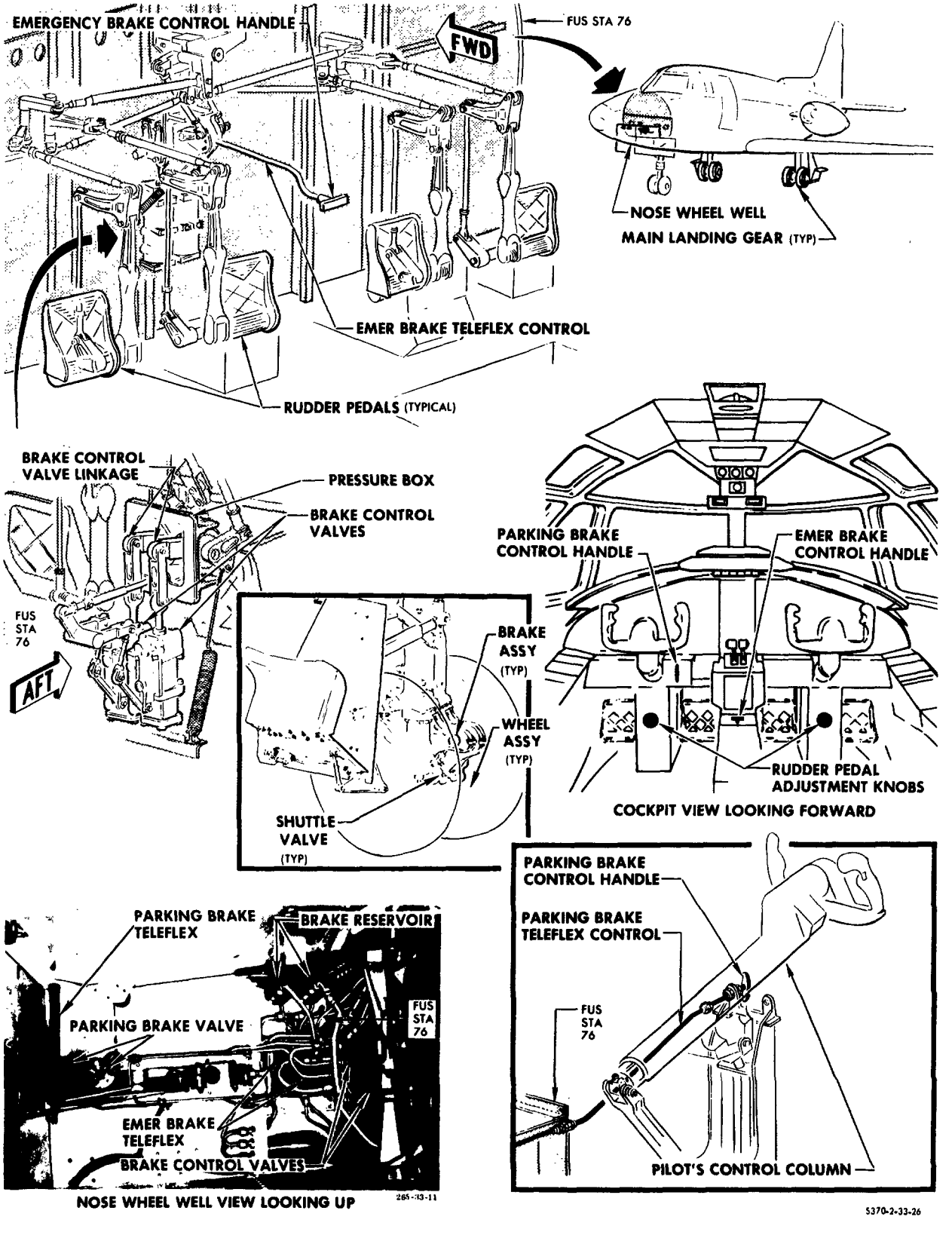


Brake System Schematic - NA-370-2 Through -6, -8 and -9 Airplanes
Figure 32-40-1



Brake System Schematic - NA-370-1 and -7, NA-380-1 and Later
Figure 32-40-1A



Wheel Brake System
Figure 32-40-2

WHEELS AND BRAKES - TROUBLE SHOOTING

1. Trouble Shooting Wheel Brake System

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. GRABBING OF BRAKES.		
(1) Air in hydraulic system.	Check for air in hydraulic system.	Bleed wheel brake system.
(2) Damaged stationary and/or rotating disks, backplate assembly, pressure plate assembly, cylinder sleeves, or pistons.	Visually inspect brakes for any sign of damage, loose parts, and corrosion. (See figure 32-40-301.)	Remove and replace brake assembly.
B. INSUFFICIENT BRAKING POWER.		
(1) Air in hydraulic system.	Check for air in hydraulic system.	Bleed wheel brake system.
(2) Tire pressure too low or high.	Check tires for correct air pressure.	Inflate tires to correct air pressure.
(3) Worn disks and/or backplate wear pads; damaged or worn backplate, pressure plate, cylinder sleeves, and/or piston.	Apply and hold brakes applied. Measure distance return pin recedes into brake assembly. (See figure 32-40-301.)	Replace brake assembly.
C. AIRPLANE PULLS TO RIGHT OR LEFT.		
(1) Insufficient brake clearance.	Check that brake clearance is within tolerance. (See figure 32-40-301.) If brake clearance is below tolerance, it will cause airplane to pull to right or left.	Replace brake assembly.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
(2) Brake return springs and pins do not retract pressure plate assembly and pistons.	Apply, then release brakes. Check that disks on both brake assemblies are fully released. This can be checked by manually moving brake disks.	Replace brake assembly.
(3) Warped or distorted brake disks.	Check brake clearance. If clearance is uneven around brake assembly, brake disks are warped or distorted.	Replace brake assembly.
(4) Brake control valve not releasing all hydraulic pressure.	Install hydraulic pressure gage at wheel brake. Depress and release rudder pedal. Hydraulic pressure should return to approximately 35 psi when pedal is released. If pressure does not return to this value, brake control valve is defective.	Replace brake control valve.
D. AIRPLANE PULLS TO RIGHT OR LEFT WHEN BRAKES ARE APPLIED.		
(1) Brake running clearance uneven between right and left brake.	Check brake running clearance. Brake running clearance between left and right brake should be nearly the same. If brake clearance between left and right brake is not reasonably close, it could cause airplane to pull to one side when brakes are applied.	Replace brake assembly having the closer clearance.
(2) Defective brake control valve.	Check hydraulic pressure at wheel brake assembly. With the same applied pedal load, the difference in brake pressure at the wheel brakes should not exceed 100 psi.	Replace brake control valve.
(3) Defective parking brake control valve.	Connect a hydraulic pressure gage to brake assembly. Depress brake pedal and record indicated pressure. If pressure is not	Replace parking brake control valve.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
	within tolerance, disconnect hydraulic line at parking brake control valve leading to brake control valve. Connect a gage to disconnected line, then depress rudder pedal. If hydraulic pressure at gage is within tolerance, the parking brake control valve is defective.	
(4) Emergency brake Teleflex control linkage improperly adjusted.	Check adjustment of emergency brake Teleflex control linkage. (See figure 32-40-404.) Improper adjustment of control linkage could block internal passages in the brake control valve.	Adjust emergency brake control linkage. (See figure 32-40-404.)
(5) Hydraulic fluid leakage at brake assembly.	Apply brake and visually check brake assembly for hydraulic leaks. If there is evidence of hydraulic leaks, the brake should be disassembled and inspected.	Replace brake assembly.
E. VIOLENT BRAKING OR LOCKING OF BRAKES.		
(1) Brake control valves not operating correctly.	Install gages at wheel brake assemblies. Depress rudder pedal slowly and evenly. Brake pressure at wheels should increase smoothly and evenly. If pressure is erratic, brake control valve is defective.	Replace brake control valve.
(2) Brake clearance below standard.	With all pressure removed from brake, check disks by hand to see if they are free.	Reset adjusters to recommended clearances.
F. LEFT OR RIGHT BRAKE CANNOT BE APPLIED WHEN USING EMERGENCY BRAKE SYSTEM.		
(1) Emergency brake control linkage	Shut off normal hydraulic system pressure. Discon-	Adjust emergency brake control linkage

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
at brake control valve out of adjustment or brake control valve is defective.	nect emergency brake control linkage at brake control valve. Manually actuate brake valve actuating arm to emergency position. Have an assistant pump, then hold rudder pedal depressed. Check that brake is applied. If brake is applied, emergency brake control linkage is incorrectly adjusted. If brake is not applied, brake control valve is defective.	or replace brake control valve.
G. EXCESSIVE BRAKE WEAR (ONE SIDE OVER THE OTHER).		
(1) Insufficient brake clearance (brakes dragging).	Check that brake clearance is within tolerance. (See figure 32-40-301.) If break clearance is below tolerance, it will cause excessive brake wear	Replace brake assembly.
(2) Brake control valve not releasing all hydraulic pressure.	Install hydraulic pressure gage at wheel brake. Depress and release rudder pedal. Hydraulic pressure should return to approximately 35 psi when pedal is released. If pressure does not return to this value, brake control valve is defective.	Replace brake control valve.
(3) Defective brake control valve.	Check hydraulic pressure at wheel brake assembly. With the same applied pedal load, the difference in brake pressure at the wheel brakes should not exceed 100 psi. (Refer to paragraph 2 in 32-40-4.)	Replace brake control valve.

WHEELS AND BRAKES — MAINTENANCE PRACTICES

1. BLEEDING WHEEL BRAKE SYSTEM

A. Bleed system as follows:

1. Open hydraulic ground test panel access door.
2. Connect hydraulic test stand to hydraulic test connections.
3. Open access doors to hydraulic reservoir.
4. Check hydraulic reservoir fluid level. Fill reservoir if necessary. The reservoir fluid level must remain above the refill level during bleeding procedure.
5. Remove dust screw from bleeder fitting on brake to be bled. Connect a bleeder hose to bleeder fitting. Place end of hose in a container partially filled with hydraulic fluid. The bleeder hose should be kept submerged in fluid during the bleeding operation.
6. Regulate test stand to 4 gpm at 3000 psi.

6A. Check that ANTI-SKID switch is OFF.

7. Have an assistant depress rudder pedal for brake to be bled. Open brake bleeder valve and watch fluid flow from bleeder hose. When fluid flow is clear and free of air bubbles, close bleeder valve and release rudder pedal.
8. Bleed emergency brake lines as follows:
   a. Pull out emergency brake knob.
   b. Pump rudder pedal of brake that is to be bled until brake pedal is firm.
   c. Have an assistant open bleeder valve. Watch fluid flow from bleeder hose. When fluid flow stops, shut off bleeder valve; then release rudder pedal.
   d. Repeat procedure in steps (b) and (c) until flow of fluid from bleeder hose is clear and free of air bubbles.
   e. Push in emergency brake knob.
9. Remove brake bleeder hose and replace dust screw.
10. Repeat steps (5) through (9) if other brakes are to be bled.
11. Service hydraulic system reservoir if necessary. Shut down and disconnect hydraulic test stand and close all access doors.

2. CHECKING BRAKE WEAR

For quick check and final wear check, refer to the "Description and Operation" section of the applicable Aircraft Braking Systems Corporation (ABSC) Component Maintenance Manual (CMM). See chart below.

ABSC BRAKE P/N	CMM MANUAL	EFFECTIVITY
9561043	AP-328 [1]	Model NA265-70
5001897	AP-422 [1]	Model NA265-80
5001897-1	AP-422 [1]	Model NA265-80

[1] With AP-763 NDT Supplement

NOTE

Some CMM's provide two (2) methods of checking brake wear. A "Quick Check" method which is more easily accomplished by measuring the adjusting pin protrusion, but is less accurate due to manufacturing tolerances of the various components; and a "Final Wear Check" procedure measuring the distance between the brake pressure plate and the brake housing as shown in the applicable CMM. The "Final Wear Check" procedure should be used to ascertain the brake's airworthiness.

3. INSPECT MLG WHEELS AND WHEEL ASSEMBLIES

Inspect/overhaul MLG and NLG wheel assemblies in accordance with ABSC (Aircraft Braking Systems Corporation) Service Letter No. GS-SL-36, Revision No. 2 dated Jan 17/94 and the applicable ABSC Component Maintenance Manual with the AP763 NDT CMM Supplement.

Equipment/Tools

- ED-520 Eddy Current Instrument (or equivalent). Available from Magnaflux Corporation, Tel: 312/867-8000

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Figure 32-40-301
titled "Brake Wear and Clearance"
has been deleted from this page.

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1. Replacement of Brake Control Valves

A. Removing Brake Control Valve

The parts referred to in the following procedure are identified in figure 32-40-401. The brake control valves are in the nose wheel well. The following procedure is typical for either brake control valve.

(1) Open fuselage access doors 6 and 14. Remove fiberglass cover over brake control valve actuating rod ends, and disconnect and cap hydraulic lines.

(2) Disconnect emergency brake control push rod at arm on brake control valve.

(3) Depress brake pedal; then, through access door 6 or 14, disconnect brake control valve actuating rod end from link assembly, and release brake pedal.

(4) Remove brake valve attach bolts and remove brake control valve.

B. Installing and Adjusting Brake Control Valve

See figure 32-40-401.

C. Removing Parking Brake Control Valve

The parking brake control valve is at the forward end of the nose wheel well. All parts in the following procedure are identified in figure 32-40-401.

(1) Disconnect and cap hydraulic lines.

(2) Disconnect Teleflex unit at valve arm.

(3) Remove valve mounting bolts and remove valve.

D. Installing Parking Brake Control Valve

NOTE: Identification of parts and illustration of individual step procedures are shown in figure 32-40-401. Individual step procedures are identified by step numbers enclosed within arrow symbols on the illustration.

(1) Position parking brake control valve in nose wheel area with valve arm facing forward.

(2) Install and tighten valve attach bolts.

(3) Uncap, connect, and tighten hydraulic lines. Tighten fitting jam nuts.

(4) Connect and adjust Teleflex unit to valve arm. (See figure 32-40-401.)

(5) Bleed wheel brake system. (Refer to 32-40-2.)

(6) Check operation of parking brake control valve. Check parking brake control valve for hydraulic leaks.

(7) Shut down and disconnect hydraulic test stand. Check hydraulic reservoir fluid level. Service reservoir if necessary.

(8) Check parking brake control valve for installation of all safeties, and install all access doors.

2. Installing and Adjusting Brake Control Linkage

See figure 32-40-402.

3. Replacement of Teleflex Control Cables

A. Removing Parking Brake Teleflex Control Cable

See figure 32-40-403, and proceed as follows:

(1) Remove clevis pin from rod end of outler slider.

(2) With index mark on cable locking belt, parallel with outer slider unit, loosen nut on cable locking belt.

(3) Turn outer slider unit counterclockwise until cable is free of threads in cable locking belt.

(4) From cockpit end of conduit, loosen nut attaching handle to conduit and pull Teleflex cable assembly from conduit.

(5) Seal cable in a clean bag to protect it from foreign material. If a new cable is being installed, save the old cable and use it to measure new cable.

B. Installing and Adjusting Parking Brake Teleflex Control Cable

See figure 32-40-403.

C. Removing Emergency Brake Teleflex Control Cable

(1) Place emergency brake control handle in stowed position (full forward).

(2) Disconnect telescopic unit from bell crank.

(3) While holding head of cable locking bolt, loosen nut one full turn.

(4) Turn outer slider of telescopic unit counterclockwise until cable is free of cable locking bolt.

(5) Loosen nut, attaching handle assembly (in cockpit) to conduit, and remove Teleflex cable.

D. Installing and Adjusting Emergency Brake Teleflex Control Cable

See figure 32-40-404.

4. Replacement of Main Wheel and Brake Assembly

A. Removing Main Wheel and Brake Assembly

(1) The following equipment is needed to perform this procedure.

- Jack - 5-ton capacity
- Pin - fuselage jacking (265-810127)
- Adapter, wing jacking (265-810129)
- Jack - tripod (set of three)

(2) To prepare airplane for wheel and/or brake assembly removal, refer to Chapter 7 and use either one of the following jacking methods.

- Use jack as specified. Jack up main gear strut until wheel clears ground.
- Use jack adapters and tripod jacks as specified and jack up entire airplane until all wheels clear ground.

NOTE: If a wheel is being replaced because of a tire change, do only steps (a) through (f). When only the wheel assembly is being removed, it is suggested that the brakes be applied and the parking brake set. This will hold the brake rotor disks in position when the wheel is removed and will aid when the replacement wheel is being installed.

- All parts referred to in the following procedure are identified in figure 32-40-405.

(3) To remove wheel and brake assembly, proceed as follows:

(a) If outboard wheel is to be removed, remove wheel cover assembly.

(b) Release air pressure from tire of wheel to be removed.

(c) Remove snap ring and dust cap from wheel axle.

(d) Remove axle nut retainer bolts and remove axle nut.

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(e) Remove washer and wheel bearing.

(f) Slide wheel assembly from strut axle.

(g) Remove seal and bearing from inner side of wheel.

(h) Install protective covers over center of wheel assembly and bearings.

(i) Cut lockwire and remove shuttle valve swivel bolt.

(j) Cut lockwire and remove the six attach bolts attaching brake assembly to flange on strut. Remove brake assembly following removal of attach bolts.

B. Installing Main Wheel and Brake Assembly

WARNING: BE SURE THAT OUTBOARD WHEEL FAIRINGS ARE INSTALLED AFTER MAINTENANCE. ADDITIONAL DRAG CREATED IN FLIGHT BY UNFAIRED WHEELS CAN ADVERSELY AFFECT FUEL CONSUMPTION.

To install main landing gear wheel and brake assemblies, see figure 32-40-405.

5. Replacement of Brake Shuttle Valve

A. Removing Brake Shuttle Valve

The brake shuttle valve is on the side of each wheel and brake assembly. Parts in the following procedure are identified in figure 32-40-405.

(1) Disconnect and cap hydraulic lines at shuttle valve.

(2) Cut lockwire at swivel bolt. Remove swivel bolt and shuttle valve.

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B. Installing Brake Shuttle Valve

(1) Install swivel bolt through shuttle valve and into brake assembly. Do not tighten swivel bolt at this time.

(2) Install union fitting at forward end of shuttle valve. Tighten fitting jam nut.

(3) Install a 90-degree elbow fitting in upper end of shuttle valve. Do not tighten fitting jam nut.

(4) Uncap, connect, and tighten hydraulic lines. Tighten fitting jam nut.

(5) Tighten shuttle valve swivel bolt, and safety with lockwire.

(6) Bleed wheel brake system.

(7) Check shuttle valve and line connections for evidence of hydraulic leaks.

(8) Shut down and disconnect hydraulic test stand. Check hydraulic system reservoir fluid level. Service reservoir if necessary. Install all access doors.

6. Replacement of Nose Gear Wheel

A. Removing Nose Gear Wheel (Parts referred to in this procedure are identified in figure 32-40-406.)

(1) On NA-370-2 through -6, -8 and -9 airplanes, raise nose wheels off the ground using jack point on nose gear strut. On NA-370-1 and -7, and NA-380-1 and later airplanes raise nose gear wheels off the ground using jacks at all three fuselage jack points.

(2) Release air pressure from tire of wheel to be removed.

(3) Cut lockwire and remove retaining nut lockbolts. Remove safety retainer and spacer; then remove retaining nut.

(4) Slide wheel off axle.

B. Installing Nose Gear Wheel (See figure 32-40-406.)

7. Replacement of Nose Gear Wheel Bearings

A. Removing Nose Gear Wheel Bearings

To remove the nose gear wheel bearings, the nose of the airplane must be raised. On NA-370-2 through -6, -8 and -9 airplanes, raise the nose gear wheels off the ground using jack point on nose gear strut. On NA-370-1 and -7 and NA-380-1 and later airplanes, raise nose gear wheels off the ground using jacks at all three fuselage jack points.

(1) Remove both nose gear wheels.

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(2) Cut lockwire and remove bolt from aft bottom side of nose gear strut. This bolt is directly aft of the nose gear strut jack point.

(3) Remove nut and bolt that secure nose gear strut uplock roller, jack point, and axle retaining lug assembly. Remove lug assembly.

(4) Install a wheel retaining nut in axle and, using a mallet, tap lightly to drive axle and one bearing assembly from strut axle housing.

(5) Using brass drift and mallet, remove remaining bearing assembly from strut axle housing.

(6) Using brass drift and mallet, tap bearing from axle shaft.

(7) Clean axle and bearing assemblies, using solvent. Dry thoroughly, using compressed dry air and clean lint-free cloth.

(8) Check axle, nose wheels axle splines and bearing assemblies for damage and wear.

B. Installing Nose Gear Wheel Bearings

(1) Lubricate wheel bearings.

(2) Install one bearing on axle shaft, with tapered portion of bearing facing axle center. Place bearing seal on axle and cup portion of seal facing away from axle center. (See figure 32-40-410.)

(3) Insert axle assembly through axle housing of strut and center axle.

(4) Install opposite bearing and seal as directed in step (2).

(5) Place axle retaining lug assembly in strut, align hole, and install aft retaining bolt.

(6) Install bolt and nut securing forward portion of axle retaining lug assembly.

(7) Safety aft retaining bolt to forward retaining bolt with lockwire.

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Installing Nose Gear Wheel Bearings (Cont'd)

8. With axle assembly in about center, install nose gear wheels. (See Figure 32-40-406.)

NOTE

There will be a certain amount of free travel to the axle assembly on installation. This free travel will center out when both wheels are installed and the retaining nuts properly tightened.

MOUNTING TUBELESS TIRES

See Figure 32-40-407.

NOSE WHEEL AND TIRE STATIC BALANCE

See Figure 32-40-408.

REPLACING MAIN LANDING GEAR WHEEL FUSIBLE PLUGS

See Figure 32-40-409.

REPLACEMENT OF HYDRAULIC BRAKE AND AUTOMATIC LANDING GEAR RETRACTION ACTUATOR (AUTOMATIC WHEEL STOPPER CYLINDERS) — NA-370 AIRPLANES NOT CHANGED BY S/B 72-8

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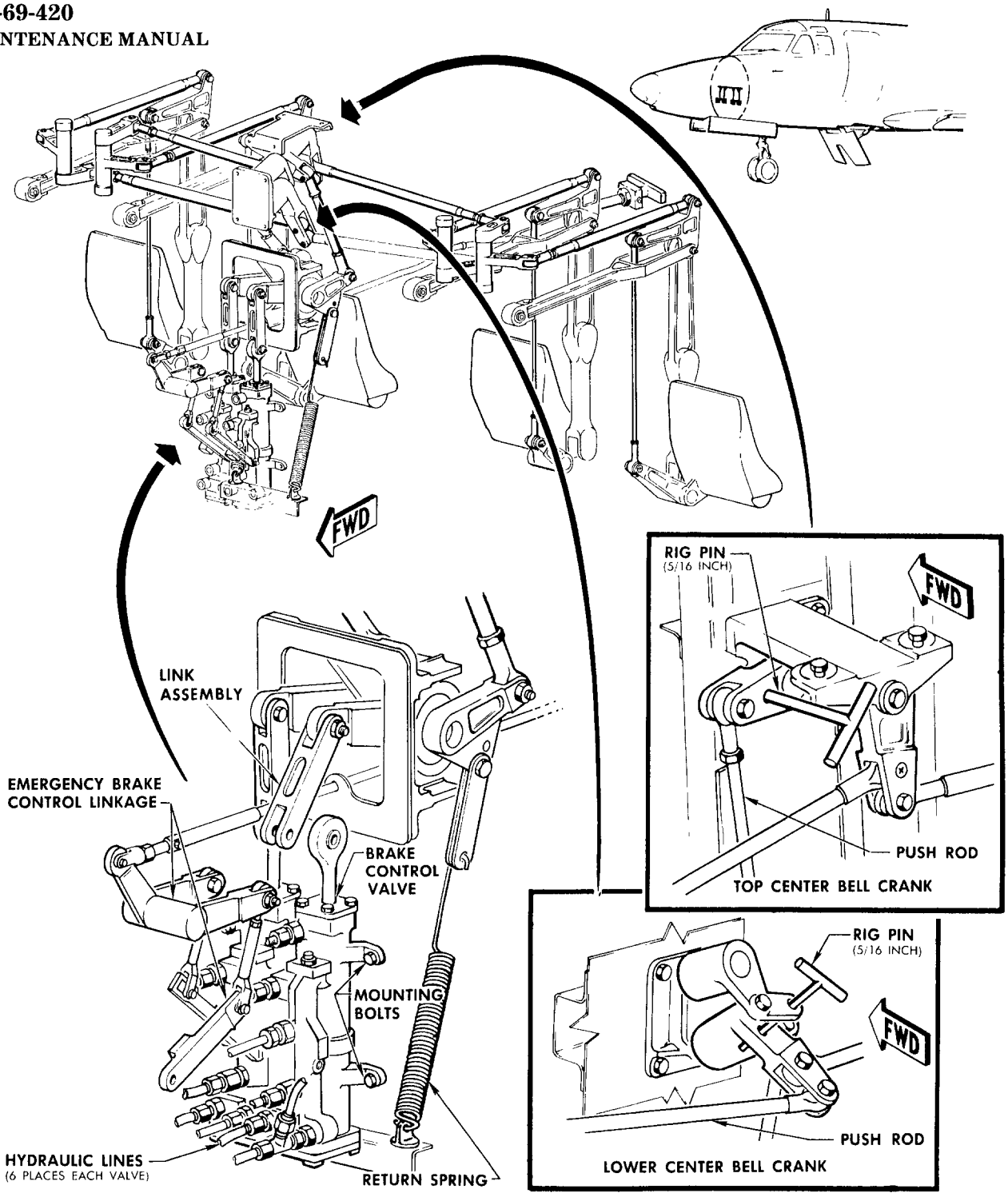

1. Depress brake pedal, and connect brake control valve rod end to link assembly. Release brake pedal, and install mounting bolts. Do not tighten mounting bolts at this time.
2. Uncap, connect, and tighten hydraulic lines. Tighten brake control valve mounting bolts and hydraulic fitting jam nuts.
3. At center bell crank, disconnect push rod that is mechanically connected to brake control valve (top center bell crank if left brake linkage is being adjusted, lower center bell crank if right brake linkage is being adjusted).

Figure 32-40-401 — Installing and Adjusting Brake Control Valve (Sheet 1 of 2)

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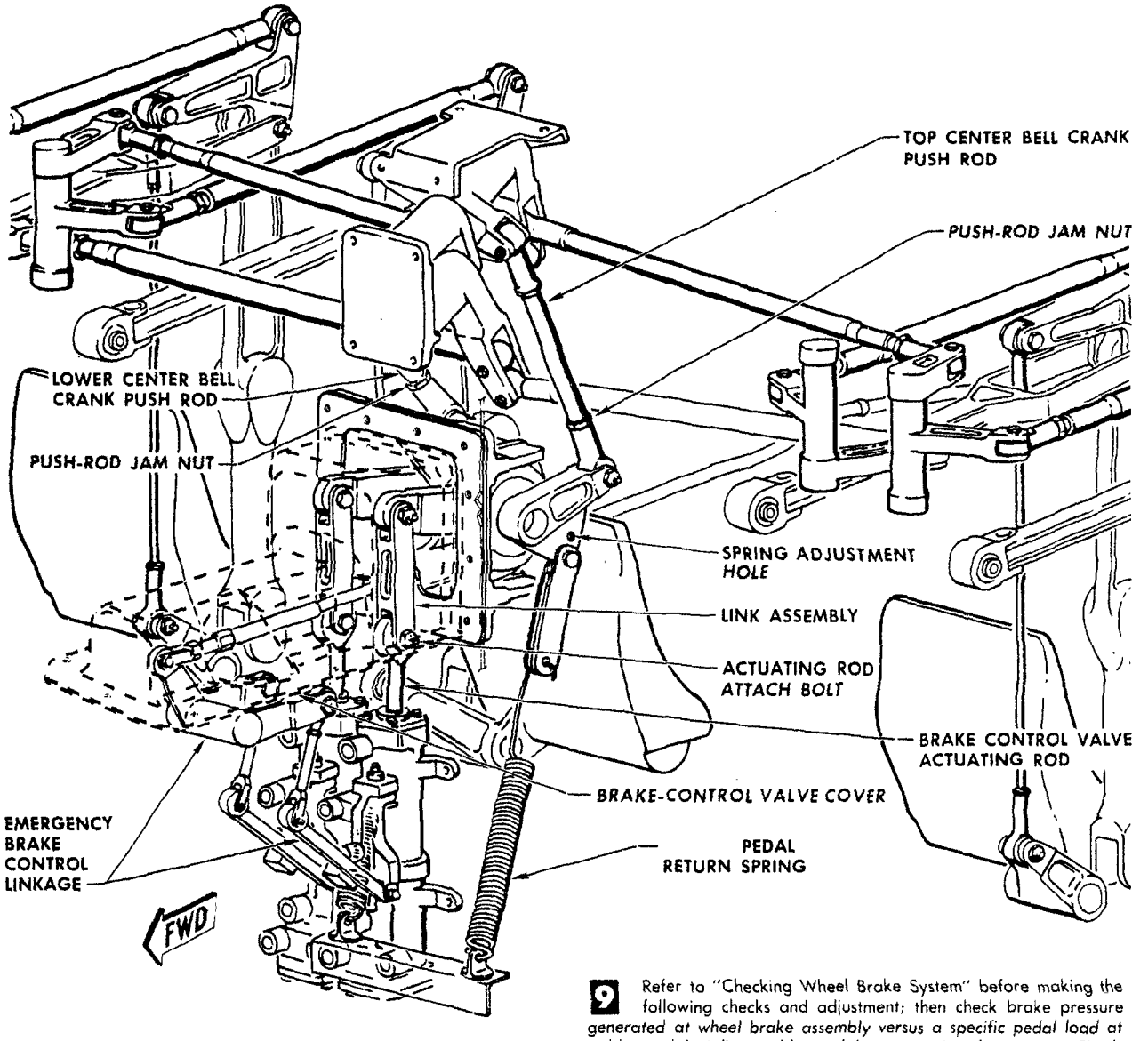

4 Install 5/16-inch diameter rig pin through top center bell crank when left brake linkage is being adjusted and through lower center bell crank when right brake linkage is being adjusted.

5 *Adjust length of push rod that was disconnected in step 3, to* obtain a "slip fit" of push-rod attach bolt; then, connect push rod to center bell crank. Tighten push-rod jam nut, and safety rod-end attach screw.

6 Remove rig pin from center bell crank.

7 *Connect and adjust emergency brake control linkage. (Refer* to "Adjusting Emergency Brake Control Linkage.")

8 Bleed wheel brake system. (Refer to "Bleeding Wheel Brake System.")

9 Refer to "Checking Wheel Brake System" before making the following checks and adjustment; then check brake pressure *generated at wheel brake assembly versus a specific pedal load at* rudder pedal. Adjust rudder pedal return spring if necessary. Check maximum pressure at brake assembly when rudder pedal is fully depressed.

10 Check emergency (manual) operation of brake control valve. *(Refer to "Checking Wheel Brake System.")*

11 Shut down and disconnect hydraulic test stand. Check hydraulic reservoir fluid level. Service reservoir if fluid level is below refill level.

12 Check brake control valve to determine whether there are *hydraulic leaks and to determine whether brake control* linkage is safely installed.

13 Install brake control valve cover; then install fuselage access doors.

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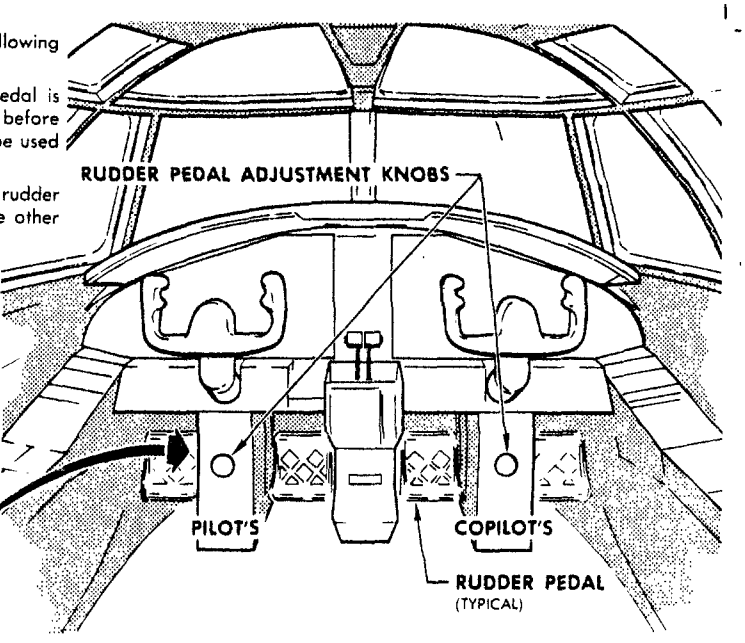
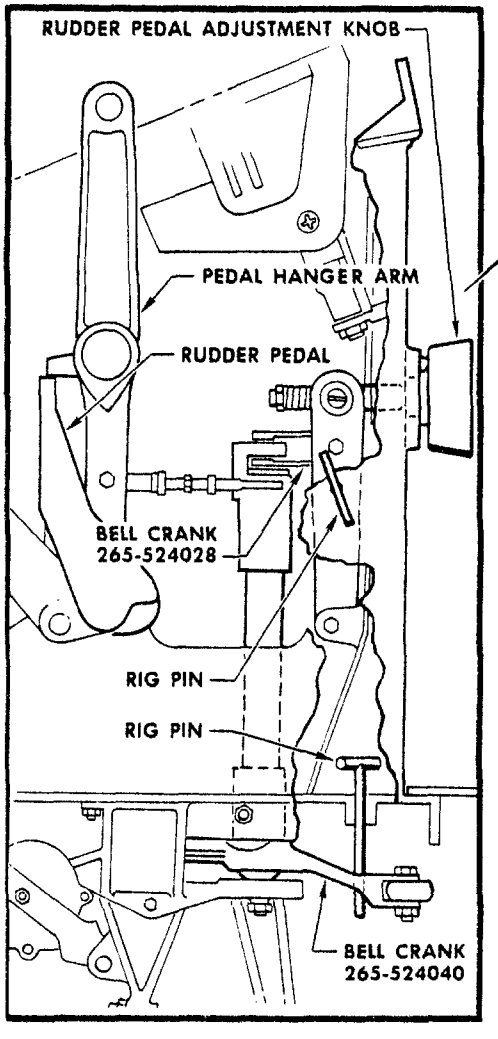
NOTE Rig pins 5/16 inch in diameter are used in the following procedure.

- Because the brake control linkage from each rudder pedal is similar, it is suggested that one rudder pedal linkage be rigged before another pedal linkage is rigged. The same rig pins can then be used to rig the other rudder pedal linkages.

The following steps apply to the rigging of one particular rudder pedal. The same procedure may be used to rig each of the other rudder pedals.



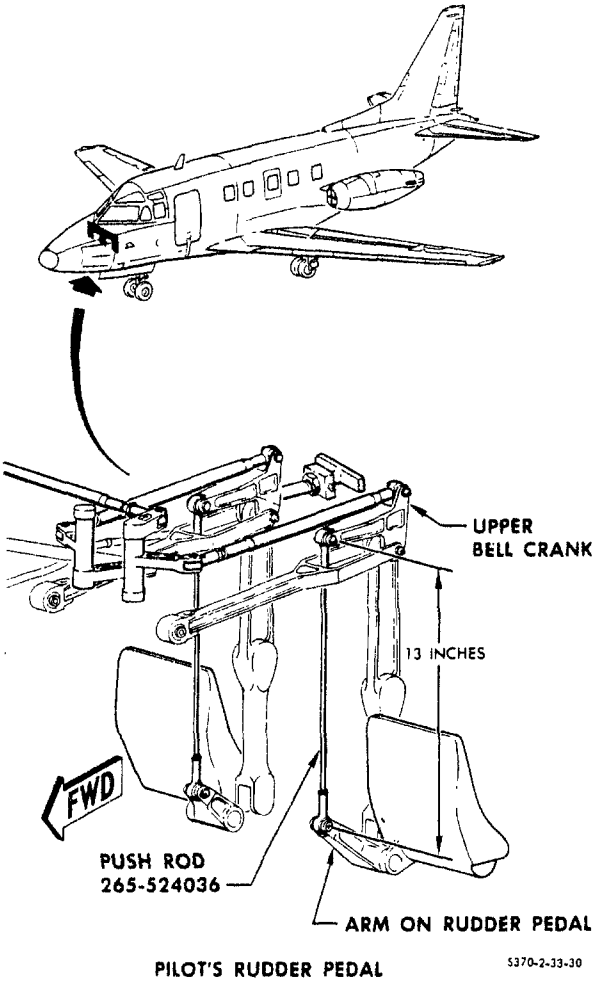


1 Position rudder pedal in mid-adjustment position by rotating pilot's or copilot's rudder pedal adjustment knob.

2 Hold rudder pedal in mid-adjustment position by installing a rig pin through bell crank 265-524028, forward of pedal adjustment knob.

3 Install a rig pin through bell crank 265-524040, to prevent rudder pedal forward and aft movement. The bell crank is below the floor; however, the rig pin is installed from the cockpit by reaching around in front of the pedestal.

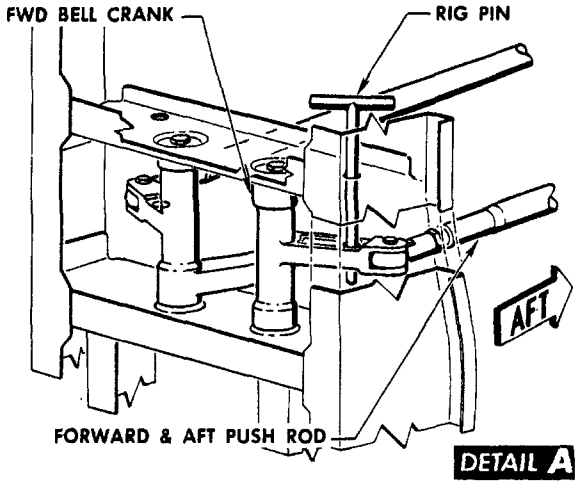
4 Adjust length of push rod 265-524036 to 13 inches (center-to-center of rod-end attach bolt holes). Connect rod to arm on rudder pedal and to upper bell crank. Install rod so that its adjustable end is connected to arm on rudder pedal. Install rod attach bolts.



PILOT'S RUDDER PEDAL

5370-2-33-30

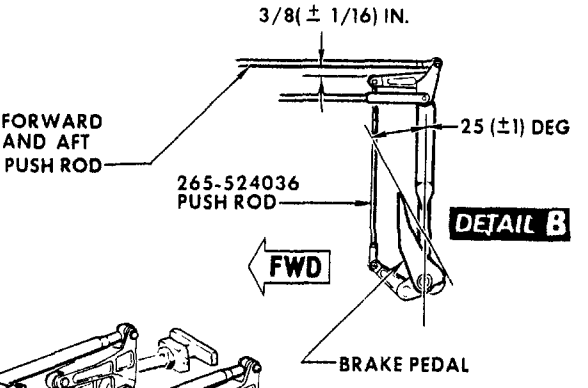
Installing and Adjusting Brake Control Linkage
Figure 32-40-402 (Sheet 1 of 3)

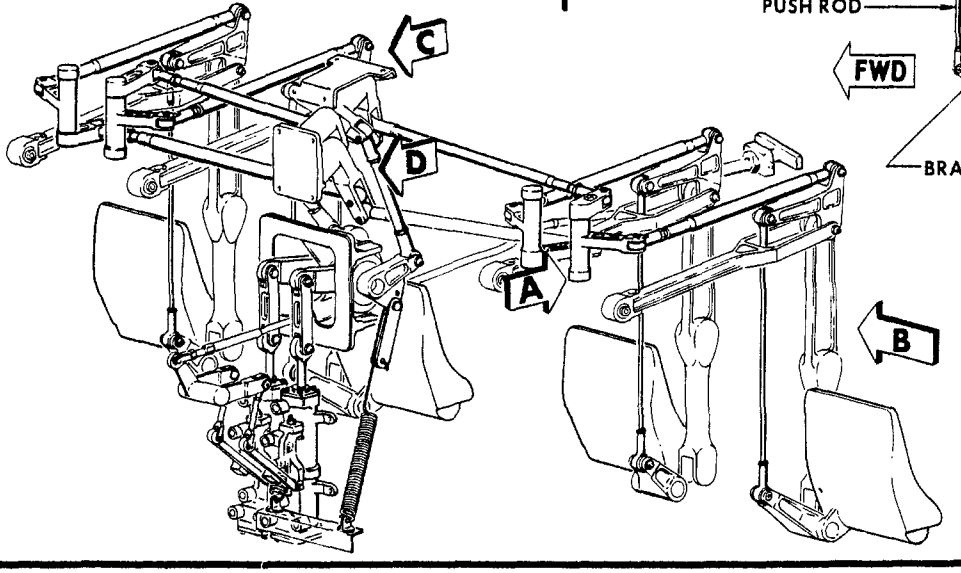


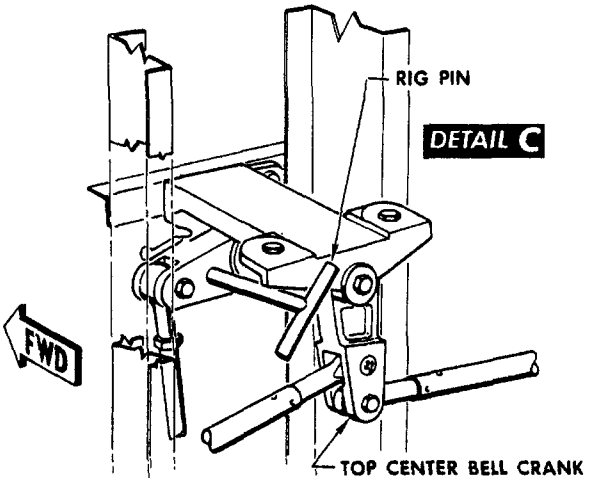
5 Install rig pin through forward bell crank.

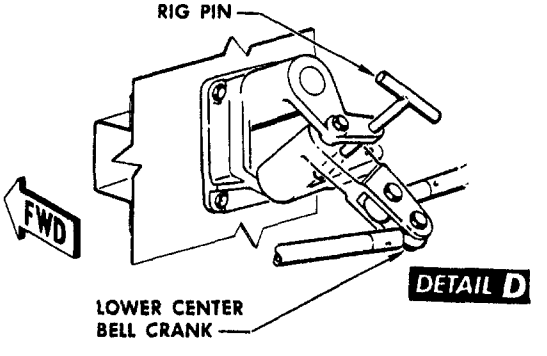
6 Connect forward-and-aft push rod to forward bell crank. Install rod so that its adjustable rod end is forward.

7 Adjust forward-and-aft push rod so that 3/8 (±1/16) inch clearance is between bottom of forward-and-aft push rod and top of push rod 265-524036 when forward-and-aft push rod is connected to the upper bell crank. Tighten forward-and-aft push rod end jam nut, and safety rod attach bolts. Adjust 265-524036 rod to provide pedal angle of 25 (±1) degrees.









8 Install a rig pin through top center bell crank when adjusting left rudder pedal linkage, and through lower center bell crank when adjusting right rudder pedal linkage.

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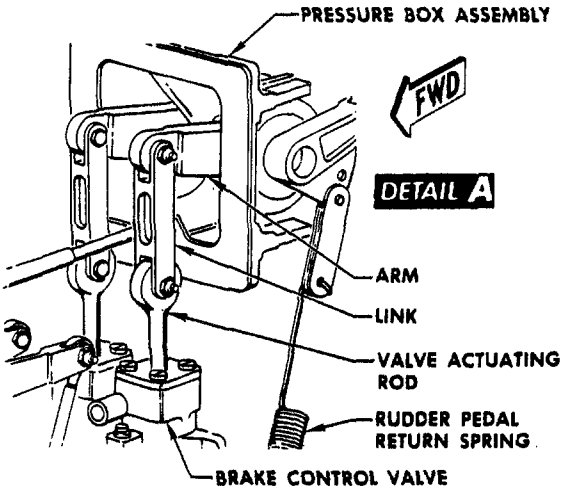
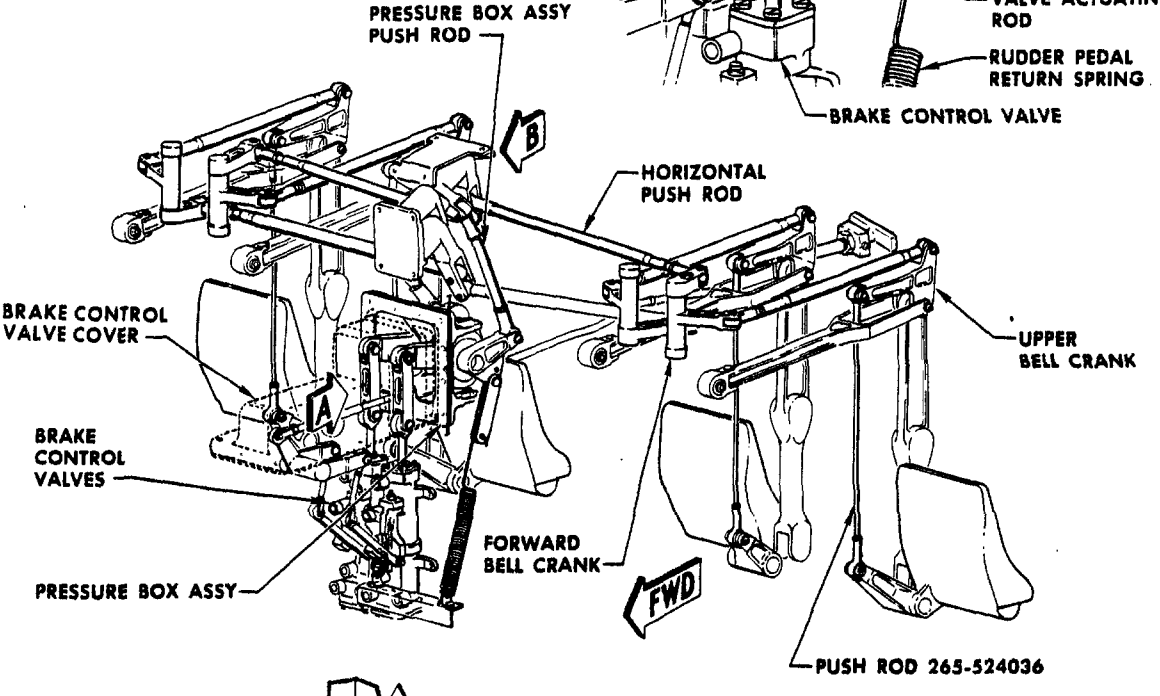
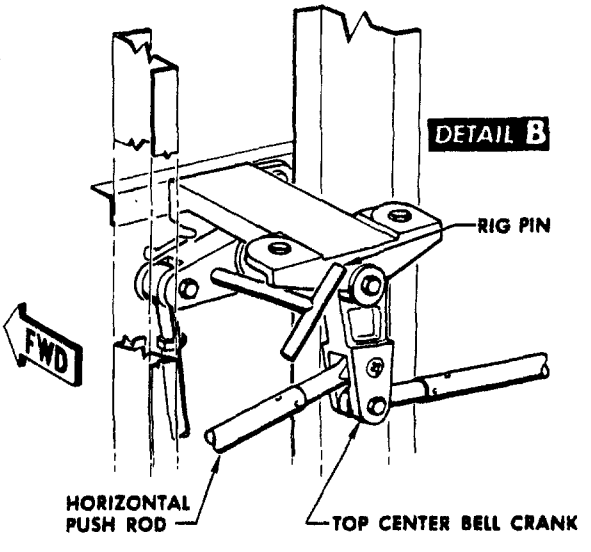
9 Connect horizontal push rod to forward bell crank. Adjust length of rod to obtain a "slip fit" of rod attach bolt when horizontal push rod is connected to center bell crank (top center bell crank when left rudder pedal linkage is being adjusted, lower center bell crank when right rudder pedal linkage is being adjusted). Tighten rod-end jam nut and safety rod attach bolts.

10 Connect link between brake control valve actuating rod and arm on pressure box assembly. Tighten and safety link attach bolts. Install rudder pedal return spring.

11 Connect pressure box assembly push rod to arm on pressure box assembly.







12 Adjust length of horizontal push rod to obtain a "slip fit" of attach bolt when push rod is connected to center bell crank (top center bell crank when left rudder pedal linkage is being adjusted, lower center bell crank when right rudder pedal linkage is being adjusted). Tighten horizontal push-rod jam nut, and safety rod-end attach screws.

13 Remove all rig pins. Connect hydraulic test stand to hydraulic system. Regulate test stand to 6-8 gpm at 3000 psi. Bleed wheel brake system if hydraulic lines or hydraulic units have been replaced. (Refer to "Bleeding Wheel Brake System.")

14 Check brake pressure generated at brake assembly versus a specific load applied to rudder pedals. Adjust rudder pedal return springs if pedal loads are high or low. (Refer to "Checking Wheel Brake System.")

15 Check whether brake control linkage has been safely and correctly installed. Disconnect hydraulic pressure gages at wheel brake bleeder fittings. Bleed wheel brake system. Disconnect hydraulic test stand.

16 Install brake control valve cover; then install fuselage access doors.

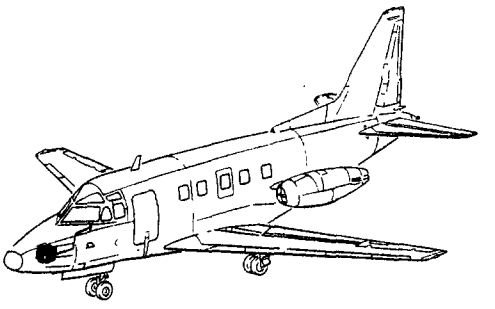
S370-2-33-32

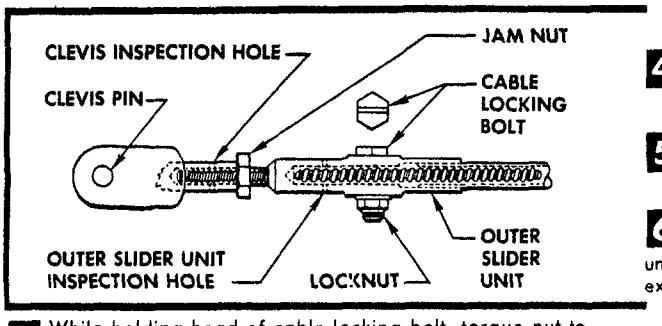
Installing and Adjusting Brake Control Linkage
Figure 32-40-402 (Sheet 3 of 3)

1 Cut new cable the same length as the old cable. (Use any suitable wire-cutting device to cut cable.)

2 Using a file or suitable tool, burr and dress cut end of cable until it is shaped to a bullet-nosed configuration. The shaping operation should be performed in a manner that tends to tighten the cable windings.

3 Insert cable (leading from cockpit) into conduit, and connect handle assembly to conduit.





4 Loosen cable locking bolt, on outer slider unit, one full turn.

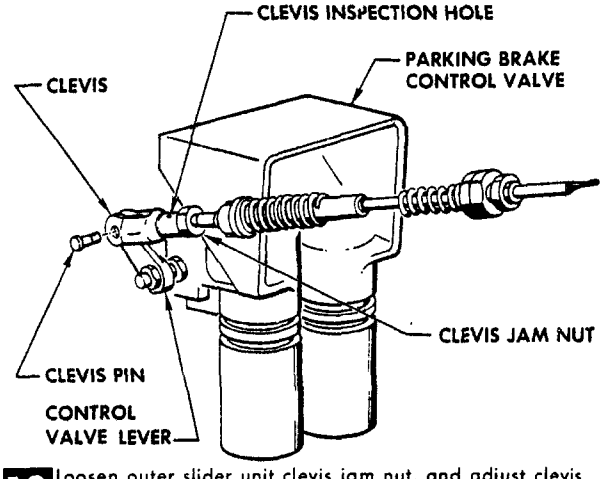
5 Insert cable into outer slider unit.

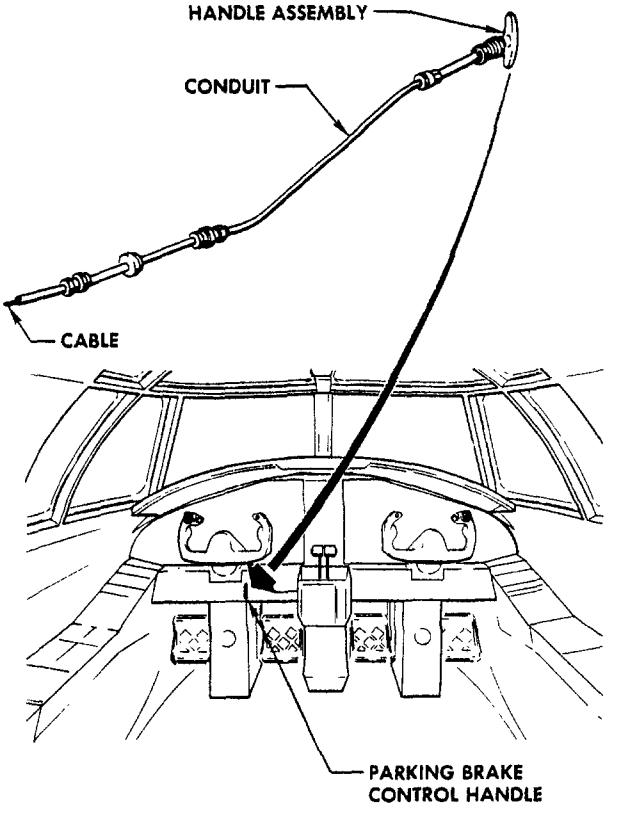
6 Turn index mark on cable locking bolt until index mark is parallel with outer slider unit; then, by rotating outer slider unit in a clockwise direction, thread cable through bolt until cable extends two turns past inspection hole of outer slider unit.

7 While holding head of cable locking bolt, torque nut to 12-15 inch-pounds.

8 Place parking brake control handle at stowed (full-forward) position.

9 Position parking brake control valve lever at rotated-full-inboard (unparked) position.





10 Loosen outer slider unit clevis jam nut, and adjust clevis until "slip fit" of clevis pin is obtained. Be sure threads are visible through inspection hole of clevis.

11 Install clevis pin, washer, and cotter pin. Tighten jam nut on clevis.

12 Check operation of parking brake control valve. (Refer to "Checking Wheel Brake System.")

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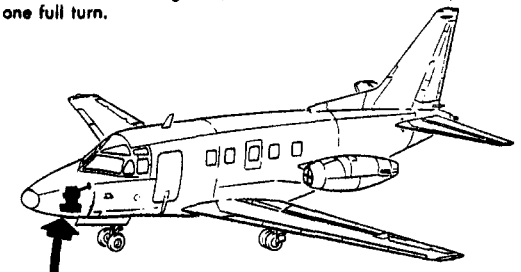
Installing and Adjusting Parking Brake Control Cable
Figure 32-40-403

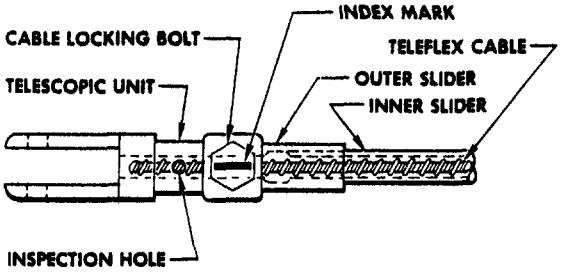
1 Cut new cable the same length as the old cable. (Use suitable wire-cutting device to cut cable.)

2 Using a file or suitable tool, burr and dress cut end of cable until it is shaped to a bullet-nosed configuration. The shaping operation should be performed in a manner that tends to tighten the cable windings.

3 Insert Teleflex cable (from cockpit) in conduit, and connect handle assembly to conduit. Place handle in stowed position (full forward).

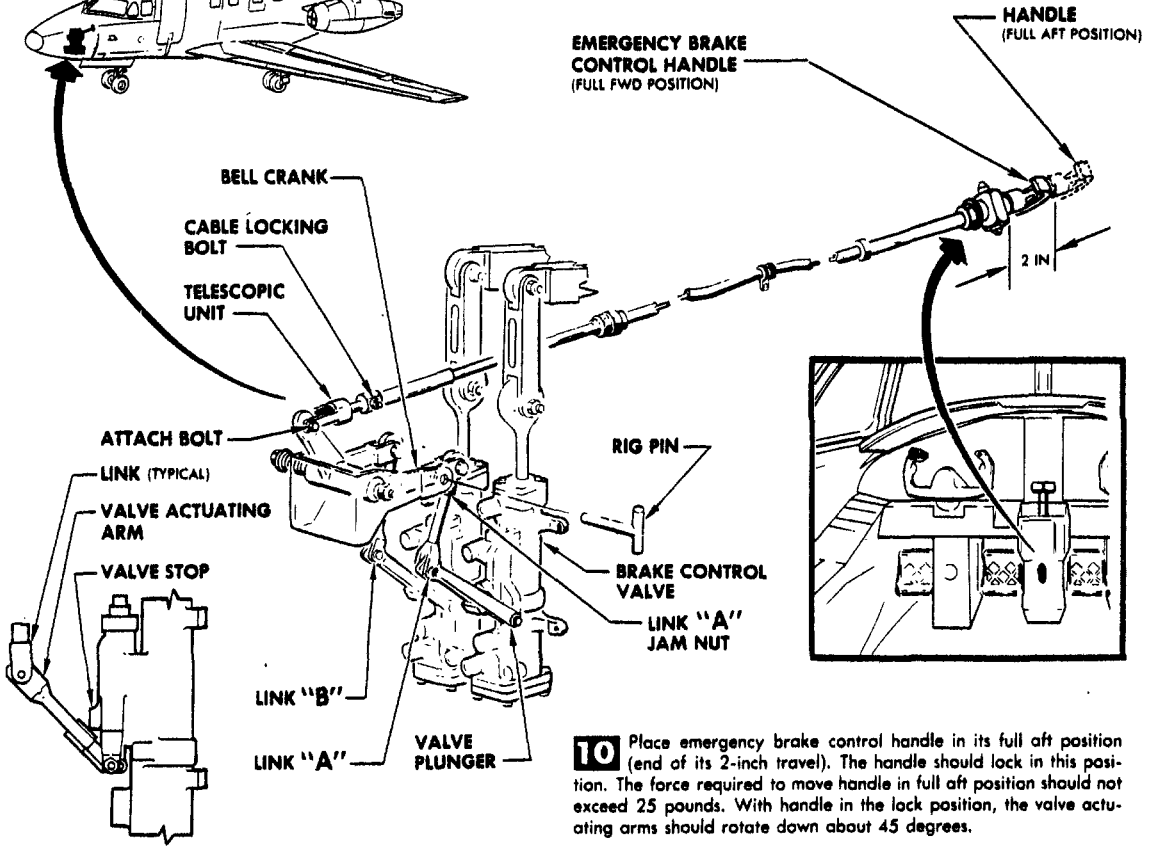
4 Loosen cable locking bolt, on outer slider of telescopic unit, one full turn.





5 Insert cable in outer slider.

6 Turn index mark on cable locking bolt, so that index mark is parallel with outer slider; then thread cable through bolt in a clockwise direction, until cable is approximately two turns past inspection hole of outer slider.



7 Place handle in stowed position (full forward). Insert rig pin in bell crank.

8 Attach telescopic unit to bell crank; then install bolt, nut, and cotter key. While holding head of cable locking bolt, torque nut to 12-15 inch-pounds.

9 With valve actuating arms against valve stops, adjust links "A" and "B," to provide a "slip fit" of bolts at links and actuating arms. Increase length of links "A" and "B" one full turn of rod-end fitting. Tighten jam nuts on links.

10 Place emergency brake control handle in its full aft position (end of its 2-inch travel). The handle should lock in this position. The force required to move handle in full aft position should not exceed 25 pounds. With handle in the lock position, the valve actuating arms should rotate down about 45 degrees.

11 Disengage handle by turning it counterclockwise 90 degrees, and pushing it forward. Place handle in stowed position.

12 Check operation of emergency brake valve.

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Installing and Adjusting Emergency Brake Control Cable
Figure 32-40-404

NOTES

- ***If a wheel has been replaced because of a tire change, omit steps 1 through 6.***
- ***Wheel bearings should be packed with grease before each wheel installation.***

1. Spread a thin film of grease (Specification MIL-G-81322) on axle and spacer. Position spacer over strut axle and against flange of strut.
2. Position brake assembly against flange on strut. Lubricate attach bolt threads with antiseize compound (Specification MIL-T-5544).
3. Align boltholes in brake assembly housing with boltholes in flange on strut axle. Install attach bolts, washers (as required) and nuts. Lubtork to 150 inch-pounds.

NOTE

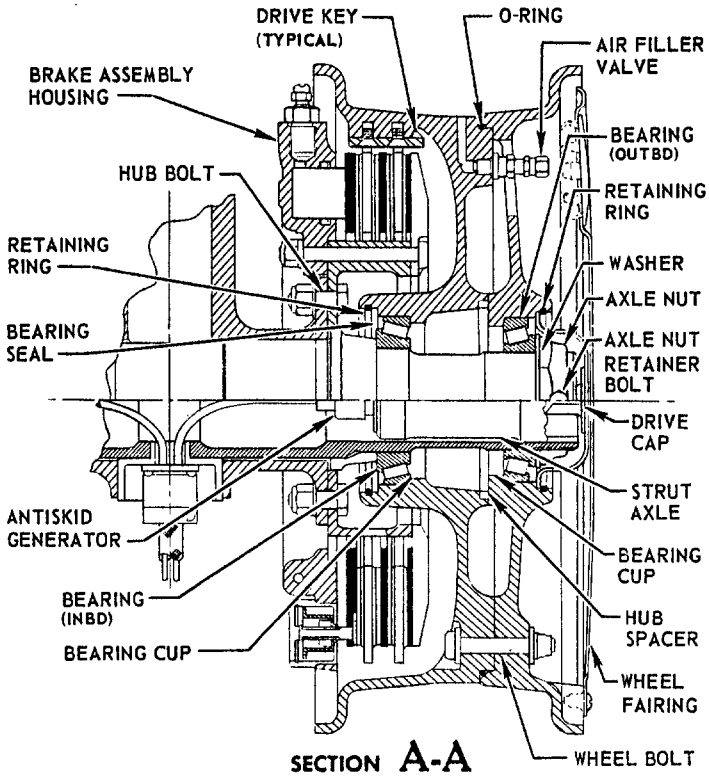
If a new or rebuilt brake assembly is being installed, be sure the plastic protective plug in the forward inlet port has been removed and replaced with an AN814-4DL plug (torque 120 inch-pounds) and safety-wire.

4. Install shuttle valve to brake assembly. Safety shuttle valve swivel bolt.
5. Rotate drive key slots in two rotor disks so that they are in alignment.
6. Install bearing and seal in inner side of wheel assembly.

NOTE

If wheel that was removed is being reinstalled, check drive keys in wheel for damage and security before installing wheel.

7. Slide wheel assembly over strut axle and brake and seat wheel against spacer on inboard end of strut axle.
8. Slide wheel outer bearing over strut axle and push bearing against wheel assembly.
9. Position washer over axle; then install axle nut.
10. Without rotating wheel assembly, torque axle nut to 125 (±10) foot-pounds. Back off axle nut; then retorque to 25 foot-pounds. Align slot in axle nut with nearest hole in axle. Install two axle nut retainer bolts.
11. Install axle nut dust cap and snap ring. Check tire pressure.
12. Bleed wheel brake system. (Refer to Bleeding Wheel Brake System.)
13. Perform operational check-out of wheel brake and anti-skid systems. Check brake assembly for evidence of hydraulic leaks.
14. If outboard wheel was removed, replace wheel fairing assembly.
15. Shut down and disconnect hydraulic test stand. Check fluid level of hydraulic system reservoir. Service reservoir if necessary.
16. Lower airplane and remove jacks.
17. Install static ground wire using AN520-10R10 screw, LD153-0002-1403 washer and MS21042L3 nut. Secure with .032 safetywire. Adjust static ground wire so that wire is in contact with ground.



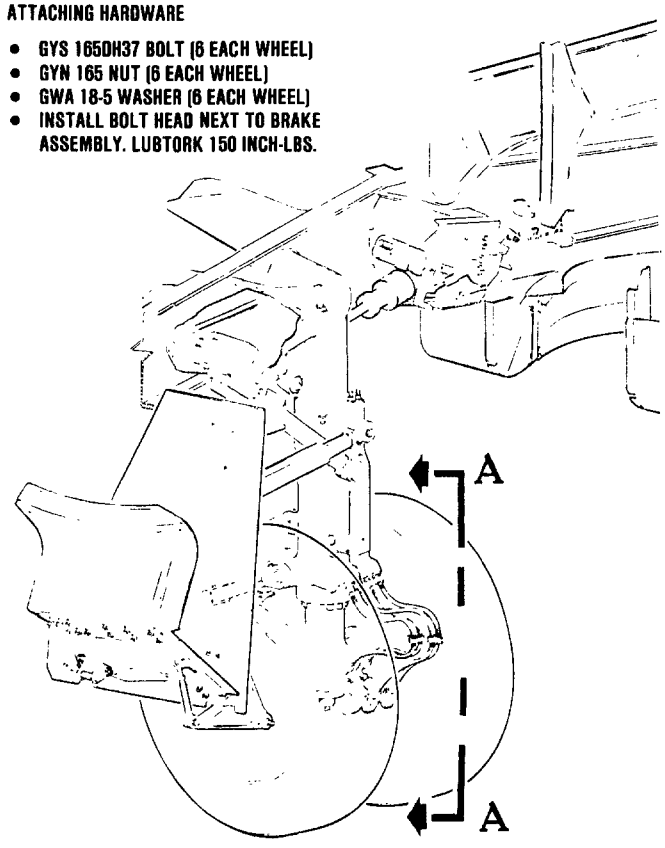

ATTACHING HARDWARE
• GYS 1650H37 BOLT (6 EACH WHEEL)
• GYN 165 NUT (6 EACH WHEEL)
• GWA 18-5 WASHER (6 EACH WHEEL)
• INSTALL BOLT HEAD NEXT TO BRAKE ASSEMBLY. LUBTORK 150 INCH-LBS.
A
A

Figure 32-40-405 — Installing Wheel and Brake

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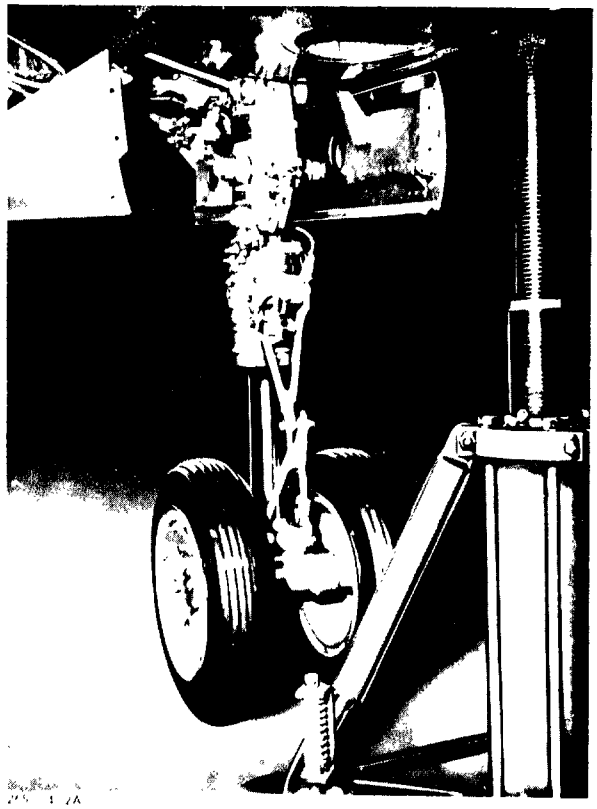


NOTE

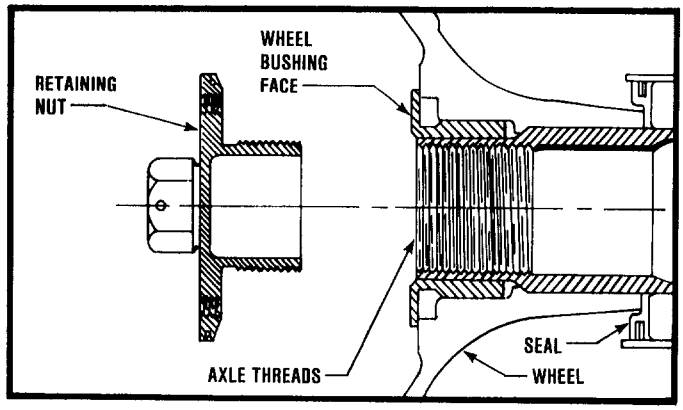
When nose wheel tires are replaced, they must be from the same manufacturer, and the amount of tread on the tires must be about equal. This ensures more even tread wear and prevents excessive nose wheel shimmy or vibration due to mismatched tires.

1. Inspect axle for condition. Check bearing shield for cleanness.
2. Spread a thin film of grease (Specification MIL-G-25760) on axle and axle splines, and slide wheel onto axle.
3. Thoroughly clean and dry retaining nut, axle threads, and wheel bushing face. To obtain an accurate torque, these parts must be free of all lubricants.
4. With wheels off ground, equally tighten both retaining nuts. Check that no clearance exists between the retaining nuts and wheel bushing face (this assures retaining nuts torqued against wheels and not bottoming against both ends of axle). Install one nut retainer and spacer over two wheel bolt heads and washers. Align and install two lockbolts and washers, diametrically opposite each other, into the retaining nut.

NOTE

Make sure lockbolts are NAS1302-1H and washers are LD153-0011-0014, as the lockbolts do not extend into the wheel hub. Shorter or longer bolts will result in thread damage.

5. Torque opposite retaining nut to 63 (±5) foot-pounds. Loosen and retorque to 13-25 foot-pounds while aligning lockbolt holes in retainer nut with locknut holes in retainer.
6. Install lockbolts and safety with lockwire as shown.



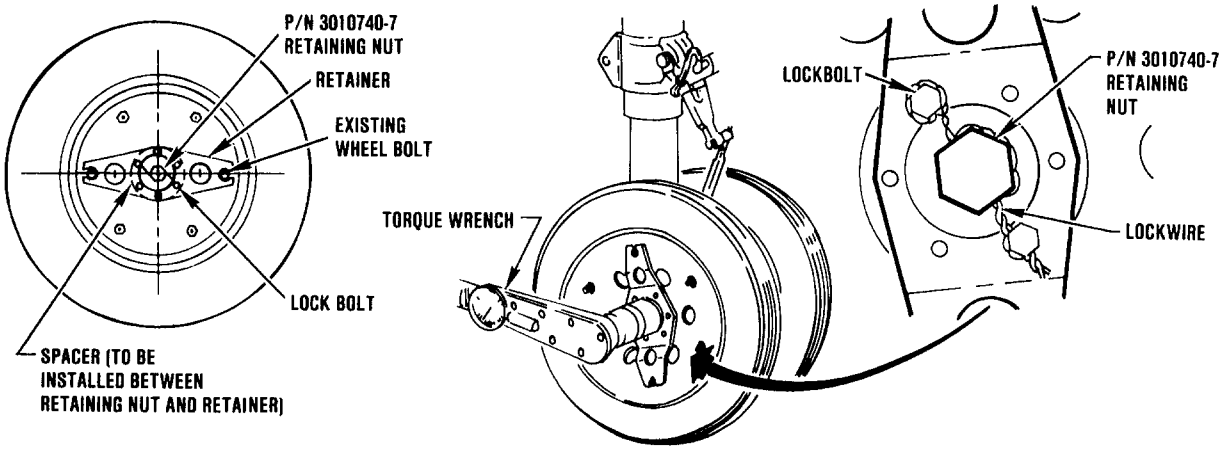


Figure 32-40-406 — Installing Nose Gear Wheels

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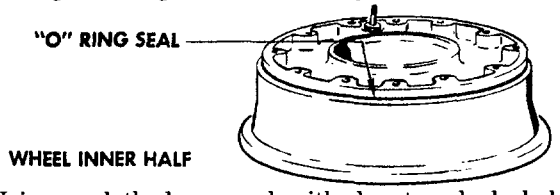
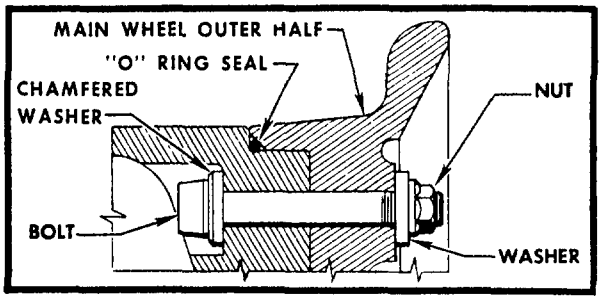
NOTES

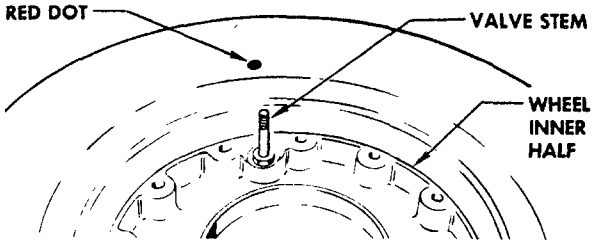
- ***Before mounting nose wheel tires, make sure tires are matched (same manufacturer, tread wear, etc.)***
- ***For equal braking action, the main gear tire tread design (smooth, nonskid, channel tread, etc.) should be similar but not necessarily made by the same manufacturer.***



1. Using a cloth dampened with denatured alcohol, clean wheel flange O-ring seal and its mating surfaces in wheel and flange. Lubricate flange O-ring seal with a light coat of grease (Specification MIL-G-81322A) before installation.
2. Place inner wheel half on a flat surface and install flange O-ring seal in groove provided.



MAIN GEAR WHEEL



3. Position tire on wheel half with balance mark (red dot on casing) aligned with valve stem.

NOTE

If necessary, dry talcum or water-free lubricants may be used on wheel flanges.

CAUTION

Do not assemble wheel halves in such a way that the wheel balance weights hardware are installed opposite each other. This could cause damage to the wheel due to the interference fit caused by the balancing weight attaching hardware.

4. Check that O-ring seal is not twisted and for correct seating, then position outer wheel half, centering valve stem in valve hole.
5. Compress wheel halves enough to allow installation of four wheel-half bolts 90 degrees apart; draw up evenly until wheel halves seat; then install remaining wheel-half bolts.

CAUTION

On main wheels the boltheads must be on inboard side of the wheel. Using antiseize compound (Specification MIL-T-5544), lubricate bolt threads and bearing surfaces of washers and boltheads.

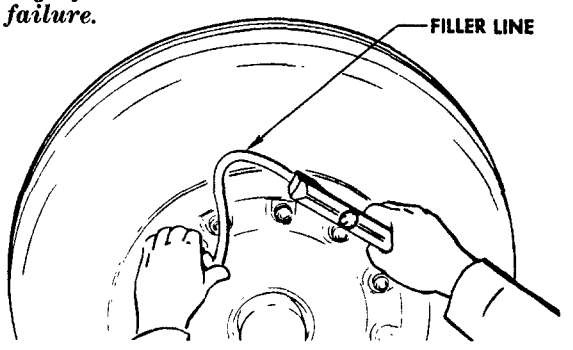
CAUTION

On main gear wheels install chamfered washers with beveled edge facing bolthead. Use plain washers under nuts.

6. Tighten bolts evenly in a crisscross pattern in increments of 5 foot-pounds to a final "Lubtork" value as shown on wheel.

CAUTION

Do not tighen wheel bolts and nuts with impact or power wrenches. Uneven tightening of bolts and nuts can cause wheel failure.



7. Inflate casing with enough air to seat beads properly (100 psi maximum).
8. Inflate tire to proper inflation pressure. (Refer to chapter 12.) Inspect tire and wheel for defects. Check valve for leaks by putting a small amount of water on end of valve stem and watching for air bubbles. If bubbles appear, replace valve core. Install valve cap to extreme finger-tightness.

NOTES

- ***It is recommended that only nitrogen be used for tire inflation to preclude possible condensation build-up with air inflation; nitrogen will not sustain combustion and will reduce degradation of innerliner material due to oxidation.***
- ***It is sometimes desirable to lubricate tires to facilitate mounting and seating of the tire beads against the wheel flanges. When this becomes necessary, it is recommended that approved dry talcum be used during mounting. Care must be used with tubeless tires, however, to ensure that none of the talc gets on the heal and/or seating area of the tire bead.***

WARNING

High pressure air sources and booster pumps not designed for tire inflation must not be used, because of possible damage to tire and/or injury to personnel.

Figure 32-40-407 — Mounting Tubeless Tires

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NOTES

- *To static balance the nose wheel and tire assembly, the Snap-on Tools Corporation wheel balancer, Part No. WK-2C, or equivalent, may be used.*
- *Both nose gear wheels must be from the same manufacturer.*

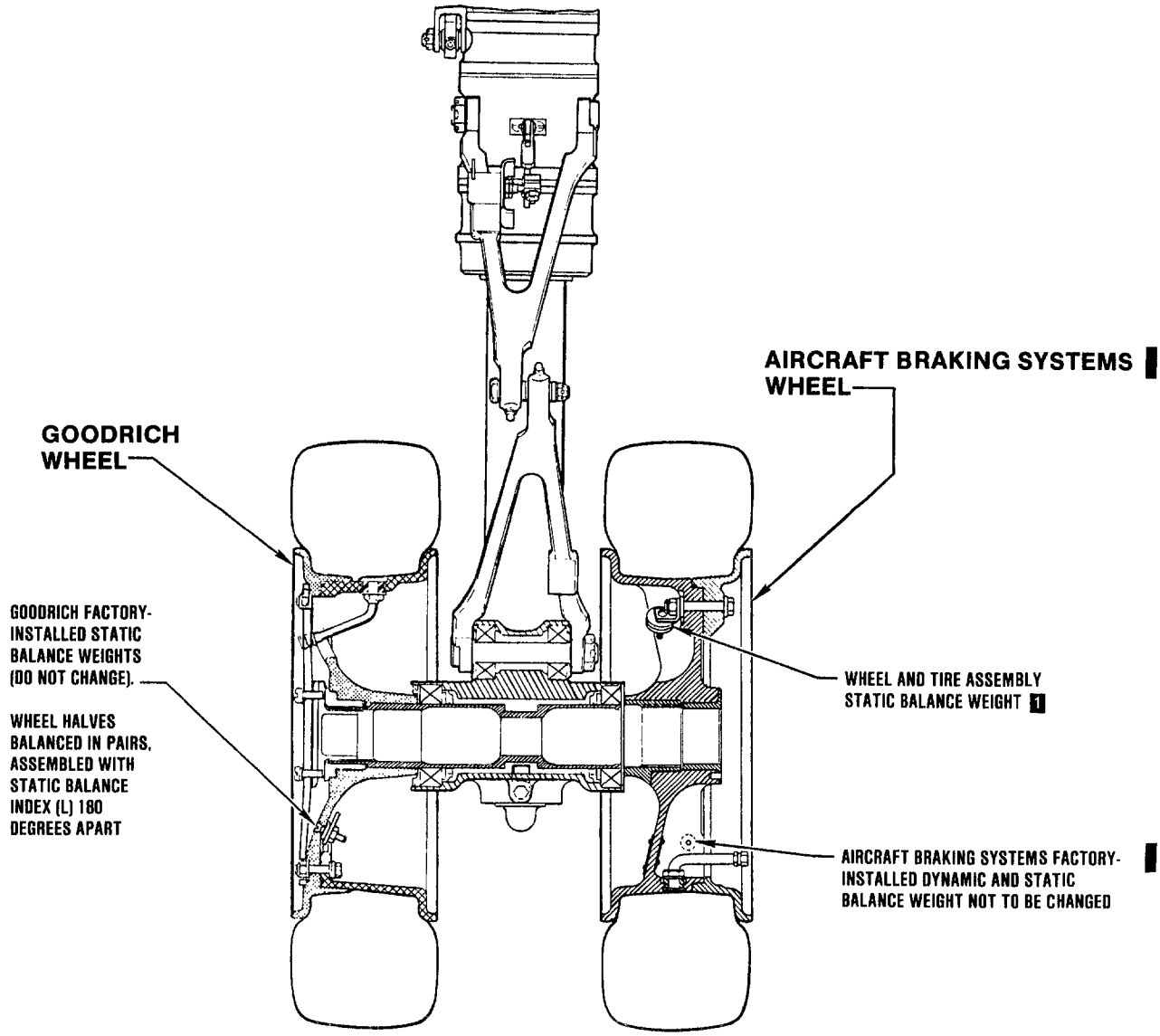
AIRCRAFT BRAKING SYSTEMS WHEEL
GOODRICH WHEEL
GOODRICH FACTORY-INSTALLED STATIC BALANCE WEIGHTS (DO NOT CHANGE).
WHEEL HALVES BALANCED IN PAIRS, ASSEMBLED WITH STATIC BALANCE INDEX (L) 180 DEGREES APART
WHEEL AND TIRE ASSEMBLY STATIC BALANCE WEIGHT 1
AIRCRAFT BRAKING SYSTEMS FACTORY-INSTALLED DYNAMIC AND STATIC BALANCE WEIGHT NOT TO BE CHANGED
1 MAY BE INSTALLED ON AIRCRAFT BRAKING SYSTEMS WHEELS TO ACHIEVE STATIC BALANCE. (REFER TO AIRCRAFT BRAKING SYSTEMS OVERHAUL MANUAL AP 333)
S370-2-33-38B

Figure 32-40-408 — Nose Wheel and Tire Static Balance

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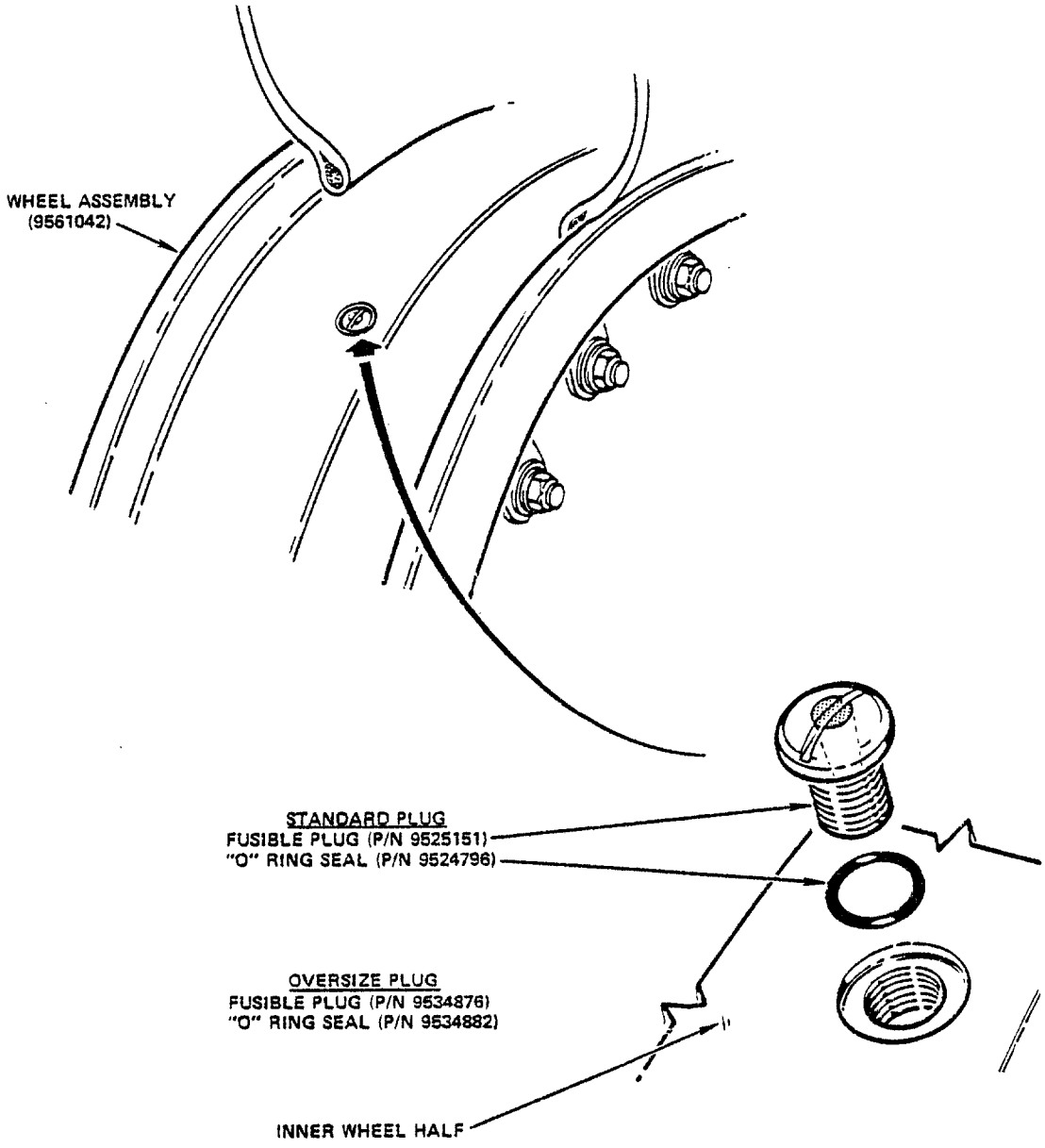
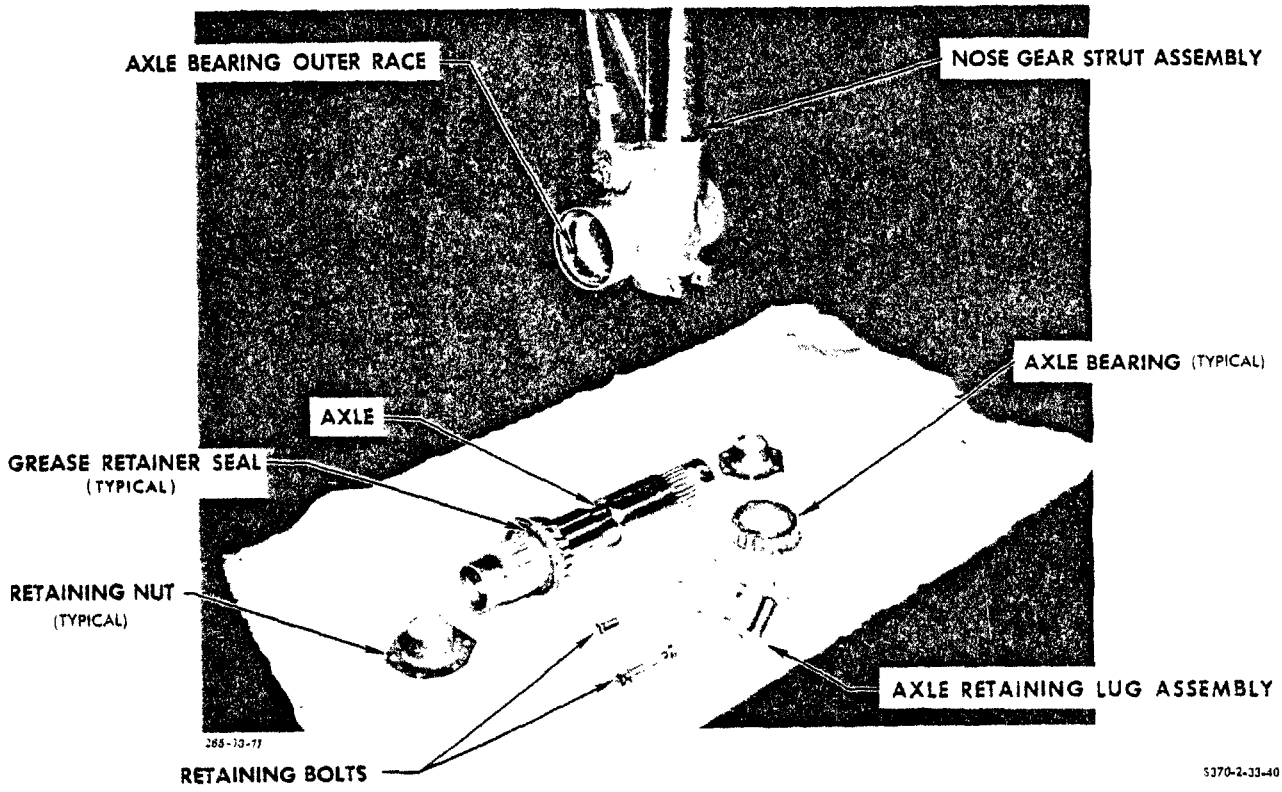


S370-2-33-39A

Replacing Main Landing Gear Wheel Fusible Plugs
Figure 32-40-409



Nose Gear Wheels, Axle, and Bearing Assembly
Figure 32-40-410

OPERATIONAL TEST — WHEEL BRAKE SYSTEM

1. Check operation of emergency brake system as follows:
   A. Pull emergency brake control handle full aft. (The handle is on the face of the center pedestal.)
   B. Pump right rudder pedal five times, then hold pedal depressed. Have an assistant observe that right inboard and outboard wheel brake is applied.
   C. Release right rudder pedal. Have assistant observe that right inboard and outboard wheel brakes are fully released. This can be checked by manually moving the brake rotor disks.
2. Repeat step 1 to test the left brake assembly.
3. Connect external 28 VDC electrical power to airplane.
4. Engage HYD PRESSURE - AUX circuit breaker, on left overhead forward circuit-breaker panel.
5. Position HYDRAULIC - AUX SYSTEM selector switch, on hydraulic control panel, at OFF.
6. Apply hydraulic pressure.
7. Position emergency brake control handle full forward.
8. Momentarily depress rudder pedals. If parking brake is set, check that parking brake handle returns to the full forward (brakes-off) position.
9. Depress and hold right rudder pedal. Have assistant visually check that right inboard and outboard wheel brake is applied.
10. Release rudder pedal. Have assistant visually check that right inboard and outboard wheel brake is released. This can be checked by manually moving the brake rotor disks.
11. Repeat steps 9 and 10 to test the left inboard and outboard wheel brake assembly.
12. Check operation of parking brake as follows:
    A. Depress and hold rudder pedals depressed. White rudder pedals are depressed, pull parking brake handle full aft.
    B. Release rudder pedals. Have assistant check that all brakes remain applied by checking that stop plunger has engaged valve arm on control valve and retained it in park position.
    C. Momentarily depress both rudder pedals. Check that parking brake control handle returns to full forward (brakes-off) position.
    D. Have an assistant check that all brakes are released.
13. Check operation of brakes on auxiliary hydraulic system as follows:
    A. Dump hydraulic pressure from normal hydraulic system accumulator, using dump valve in aft equipment compartment.
    B. Position HYDRAULIC - AUX SYSTEM selector switch, on hydraulic control panel, at ON.
    C. Accomplish steps 9 through 11.

NOTE

If auxiliary pressure fails to operate wheel brakes, refer to trouble shooting procedures for hydraulic power system in chapter 29.

14. Set parking brake as in step 12. Return all switches to off position.
15. Disconnect external 28 VDC electrical power.
16. Remove hydraulic pressure.
17. Secure airplane as necessary.

CHECKING WHEEL BRAKE SYSTEM

NOTES

- ***A push-pull scale capable of measuring over 120 pounds and two 3000 psi hydraulic pressure gauges of known accuracy (to be used to check brake pressure at or near the wheels) are required for these procedures.***
- ***If hydraulic units have been replaced or hydraulic lines have been disconnected, the wheel brake system must be properly serviced and the system must be bled before the system is checked out.***

1. Connect hydraulic pressure gauges into wheel brake pressure line. Do not connect gauges into emergency brake line. The gauge may be attached to the brake bleeder fitting at each brake assembly. If gauge is attached to brake bleeder fitting, open bleeder valve on each brake unit enough to permit hydraulic pressure to be admitted to pressure gauge.
2. Apply hydraulic pressure.
3. Check brake pressure generated at wheel brake assembly versus a specific pedal load at rudder pedal as follows:

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ADJUSTMENT/TEST

CHECKING WHEEL BRAKE SYSTEM (Cont'd)

NOTE

When making checks in step 3, apply pedal load gradually until stated pedal load is reached. Do not exceed pedal load and then drop back to it, as this produces higher brake pressure than specified because of friction of brake valves on pedal linkage.

A. Holding a push-pull scale at a 90-degree angle from face of rudder pedal, apply a 90-pound force to toe of rudder pedal. Indicated hydraulic pressure at the wheel brake should be not less than 550 psi.

B. If pedal loads are high or low, adjust rudder pedal return springs. To increase pedal load, move rudder pedal return spring attach point on plate attached to pressure box assembly to a higher hole. To decrease rudder pedal load, move spring attach point to a lower hole. If this procedure fails, replace brake control valve.

C. Repeat step A, using auxiliary accumulator.

D. Repeat steps A through C for opposite brake. The difference in pressure obtained in step B and brake pressure obtained in this step should not exceed 100 psi. If necessary, readjust rudder pedal return springs.

4. Fully depress brake pedal. Brake pressure should be 1625 psi minimum with a 160-pound force at the toe of the rudder pedal.
5. Apply and release brakes. Check that brake pressure releases at same time as pedal load is released.
6. Check operation of parking brake as follows:
   A. Depress rudder pedals until 675 psi is indicated on gauge at brake assemblies.
   B. Hold rudder pedals depressed and pull parking brake control handle full aft; then release rudder pedals. Hydraulic pressure at the brake assembly should not drop below 530 psi after 3 minutes. Depress rudder pedals to release parking brake.
   C. Depress rudder pedals slowly in order to gradually increase pressure at brake. The gauge at the brake assembly should indicate cracking of the parking brake valve thermal relief valve (the gauge will fluctuate) at 1400 psi maximum. While holding brake pressure at thermal relief valve reseat pressure (1300 to 1400 psi), pull parking brake handle full aft; then release rudder pedals. Hydraulic pressure at the brake should not drop below 600 psi after 3 minutes.
   D. Momentarily depress and release both rudder pedals. The brake handle should return to normal position, and the brakes should release. Have assistant check that all brakes are released. This can be checked by manually moving the brake rotor disks.
   E. Check that parking brake cannot be set with brake pressure less than 450 psi. To do this, depress rudder pedals until 450 psi is indicated on gauge at brake assembly. Pull parking brake handle full aft and hold it; then release rudder pedals. The brakes should release, hydraulic pressure at the brake should drop to the system return pressure reading, and the parking brake handle should return to the normal position when released.
7. Remove normal hydraulic pressure. Using auxiliary accumulator charge, repeat step 6.
8. Apply normal hydraulic pressure; then shut down pump.
9. Pull emergency brake control handle full aft. Pump one rudder pedal until a solid feel is obtained; then apply a force, not to exceed 120 pounds, at toe of rudder pedal.
10. Have an assistant check brake pressure at gauge at wheel brake. There should be a minimum pressure of 300 psi at the brake. Release rudder pedal. Repeat check for opposite pedal and brake. The difference in brake pressure must not exceed 35 psi. If difference is more that 35 psi, rebleed and check brake for leaks.
11. Disconnect hydraulic pressure gauges at brake assemblies.
12. Bleed wheel brake system.
13. Check hydraulic reservoir fluid level. Service reservoir if necessary. Install all access doors.
14. Secure airplane as desired.

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ANTISKID BRAKE SYSTEM - DESCRIPTION/OPERATION NA-370-2 THROUGH -6, -8 AND -9

1. General

The wheel brake antiskid system, manufactured by Goodyear Tire & Rubber Co, Inc, provides controlled braking action to allow minimum landing roll and to reduce tire wear due to skidding. This is accomplished by automatic reduction of brake pressure at either wheel in case of an impending skid, and reapplication of brake pressure as the wheel recovers speed. The system consists of four wheel-driven dc generators, producing a voltage proportional to wheel speed; a control box, which monitors generator output; and two solenoid-operated control valves, which reduce and reapply pressure as directed by the control box signal. Each control valve affects pressure to the dual brakes of one main landing gear shock strut. An impending skid condition of either of the dual wheels mounted to that strut will reduce pressure to both brakes simultaneously. A fail-safe provision is incorporated so that in case of a failure which would result in brake pressure being held off for longer than 2.7 seconds, the complete antiskid system is deactivated and full control reverts to the pilot. A failed condition is indicated to the pilot by the "ANTI SKID BRAKE FAIL" light in the cockpit. Electrical power for the antiskid system is taken from the 28-volt dc essential bus through the "LANDING GEAR - ANTI-SKID - POWER" circuit breaker on the right overhead circuitbreaker panel in the cockpit. The antiskid circuit is controlled by the "ANTI SKID" switch on the center overhead control panel in the cockpit.

An additional circuit is provided through the landing gear load switches to prevent the fail-safe feature from deactivating the antiskid system during landing gear retraction. *During landing gear retraction, the automatic wheel stopper system applies the brakes to stop the wheels from rolling. *This sudden stoppage is interpreted by the antiskid system as a constant skid, since the wheels will not begin to roll again. The fail-safe circuit, after 2.7 seconds, will then deactivate the antiskid system. To avoid this unwanted deactivation and resulting warning light, the circuit which generates the brake release signal, normally applied when a skid is sensed, is bypassed to ground when the landing gear struts are extended. The circuit is wired through relays so that both main gear struts must be extended before the circuit can function. (See figure 32-41-2.)

2. Wheel-driven Generators

The wheel-driven generators are installed in the main landing gear axle and are driven by fittings inside the main wheel hub caps. (See figure 32-41-1.) The generators, when driven by the wheels, deliver a voltage proportional to wheel rpm to the antiskid control box.

3. Antiskid Control Box

The antiskid control box is mounted on the right hand side of the fuselage at station 174. Electrical power for the control box is taken from the 28-volt dc essential bus through the "LANDING GEAR - ANTI-SKID - POWER" circuit breaker. The control box monitors the voltage of the wheel-driven

* Airplanes not changed by Service Bulletin 72-8.

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generator and activates the antiskid control valves in case of an impending skid. As the wheel-driven generators turn, a current is generated which decreases as the main wheels decelerate. A skid signal is caused by any sudden drop in generator current, caused by a sharp decrease in the normal deceleration of the wheels. When the skid signal is received, the control box directs a current to the control valve for the dual wheels affected by the impending skid, causing the valve to shut off brake pressure to the wheels. A fail-safe system is incorporated in the control box so that in case of a failure which would result in brake pressure being held off for more than 2.7 seconds, the complete antiskid system is deactivated and full control reverts to the pilot.

4. Antiskid Control Valves

The antiskid control valves are solenoid-operated two-position units, mounted on the aft bulkhead of each main wheel well. (See figure 32-41-1.) The valves are actuated by a 28-volt current from the antiskid control box during an impending skid condition in any one of the main wheels. When a valve is actuated, hydraulic pressure is blocked to the affected pair of brake assemblies and pressure within the brakes is routed back to the reservoir. This allows the wheel to roll freely and regain speed, thereby canceling the skid signal to the control box. When the skid signal is canceled, the control box de-energizes the valve, allowing braking action to be reapplied. (See figure 32-40-1.)

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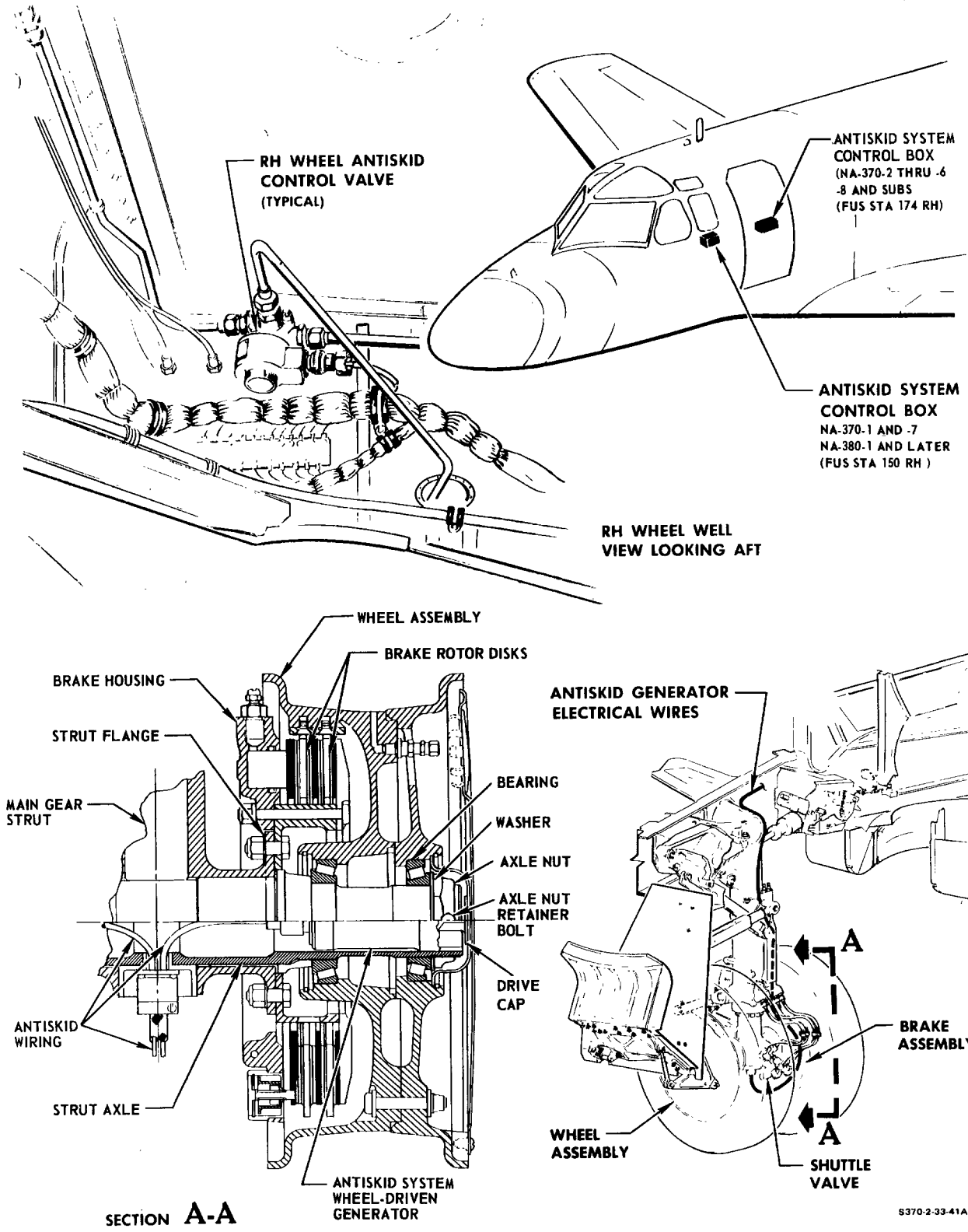
RH WHEEL ANTISKID
CONTROL VALVE
(TYPICAL)
ANTISKID SYSTEM
CONTROL BOX
(NA-370-2 THRU -6
-8 AND SUBS
(FUS STA 174 RH)
ANTISKID SYSTEM
CONTROL BOX
NA-370-1 AND -7
NA-380-1 AND LATER
(FUS STA 150 RH)
RH WHEEL WELL
VIEW LOOKING AFT
WHEEL ASSEMBLY
BRAKE ROTOR DISKS
BRAKE HOUSING
STRUT FLANGE
MAIN GEAR
STRUT
BEARING
WASHER
AXLE NUT
AXLE NUT
RETAINER
BOLT
DRIVE
CAP
ANTISKID
WIRING
STRUT AXLE
ANTISKID SYSTEM
WHEEL-DRIVEN
GENERATOR
SECTION A-A
ANTISKID GENERATOR
ELECTRICAL WIRES
A
A
BRAKE
ASSEMBLY
WHEEL
ASSEMBLY
SHUTTLE
VALVE
S370-2-33-41A

ANTISKID BRAKE SYSTEM - TROUBLE SHOOTING NA-370-2 THROUGH -6, -8 AND -9

1. Trouble Shooting Antiskid Brake System

The following trouble shooting information is applicable only to the antiskid portion of the brake system and is based on the assumption that electrical power is applied to the airplane and that hydraulic power is available to the system. Normal brake system trouble shooting is covered in 32-40-1.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. ANTISKID SYSTEM INOPERATIVE - ALL WHEELS.		
(1) Faulty antiskid control box.	Check voltage at control valve electrical plugs while simulating a skid condition. (Refer to 32-41-2.) Voltage should be at least 18 volts.	Replace control box.
(2) Fail-safe system in operation.	Check electrical plugs at control box and control valves for security.	Connect or tighten electrical plugs.
B. ANTISKID SYSTEM INOPERATIVE - ONE WHEEL.		
(1) Wheel-driven generator faulty.	Check operation of generator in affected wheel. (Refer to 32-41-2.)	Replace faulty generator.
(2) Faulty antiskid control box.	Disconnect electrical plug at affected control valve and connect jumper wires between "A" and "B" of valve receptacle and pins "A" and "B" of electrical plug. Check voltage at control valve electrical plug while simulating a skid condition. (Refer to 32-41-2.) Voltage must be at least 18 volts.	If voltage is not at least 18 volts, replace control box. If ample voltage is present, refer to probable cause (3).
(3) Faulty antiskid control valve.	Refer to isolation procedure for probable cause (2).	If voltage is 18 volts or better, replace control valve.
(4) Hub cap not driving generator.	Check that drive adapter and cap alignment pin are intact and undamaged. NOTE: Be sure that alignment pin engages hole in wheel when cap is installed.	Replace cap.

ANTISKID BRAKE SYSTEM - MAINTENANCE PRACTICES NA-370-2 THROUGH -6, -8 AND -9

1. Replacement of Wheel-driven Generator

Removal and installation procedures for both inboard and outboard wheel generators are the same. The parts referred to are shown in figure 32-41-1.

A. Removing Wheel-driven Generator

(1) Remove main wheel hub cap.

(2) Remove two bolts retaining generator in axle.

(3) Pull generator from axle. Disconnect electrical plug.

B. Installing Wheel-driven Generator

(1) Connect generator to electrical plug in main wheel axle and lockwire electrical plug.

(2) Position generator in axle and install two retaining bolts through wheel nut, axle, and generator.

(3) Perform operational check of antiskid system.

(4) Install hub cap on main wheel.

NOTE: The hub cap is provided with an alignment pin to prevent slippage of the cap. Ensure that pin is intact and that it properly engages hole provided in wheel when cap is installed.

2. Operational Check of Antiskid System

A. Perform check as follows:

(1) Remove hub cap from wheel being checked.

(2) Connect 28-volt dc external electrical power to utility external power receptacle. (Refer to Chapter 6, figure 6-00-3, for location of receptacle.)

(3) Close the following circuit breaker:

HYD PUMP CONT (If hydraulic test stand is not used).
ANTI-SKID-POWER
ANTI-SKID-WARN

NOTE: If a hydraulic test stand is used, connect hydraulic test stand to airplane test stand connections. (Refer to Chapter 6, figure 6-00-3, for location of test connections.) Set test stand to deliver 4 gpm at 3000 psi.

(4) Have an assistant place "HYDRAULIC PUMP" switch at AUTO or ON (If hydraulic test stand is not used) and "ANTI-SKID" switch at ON and apply pressure to appropriate brake pedal.

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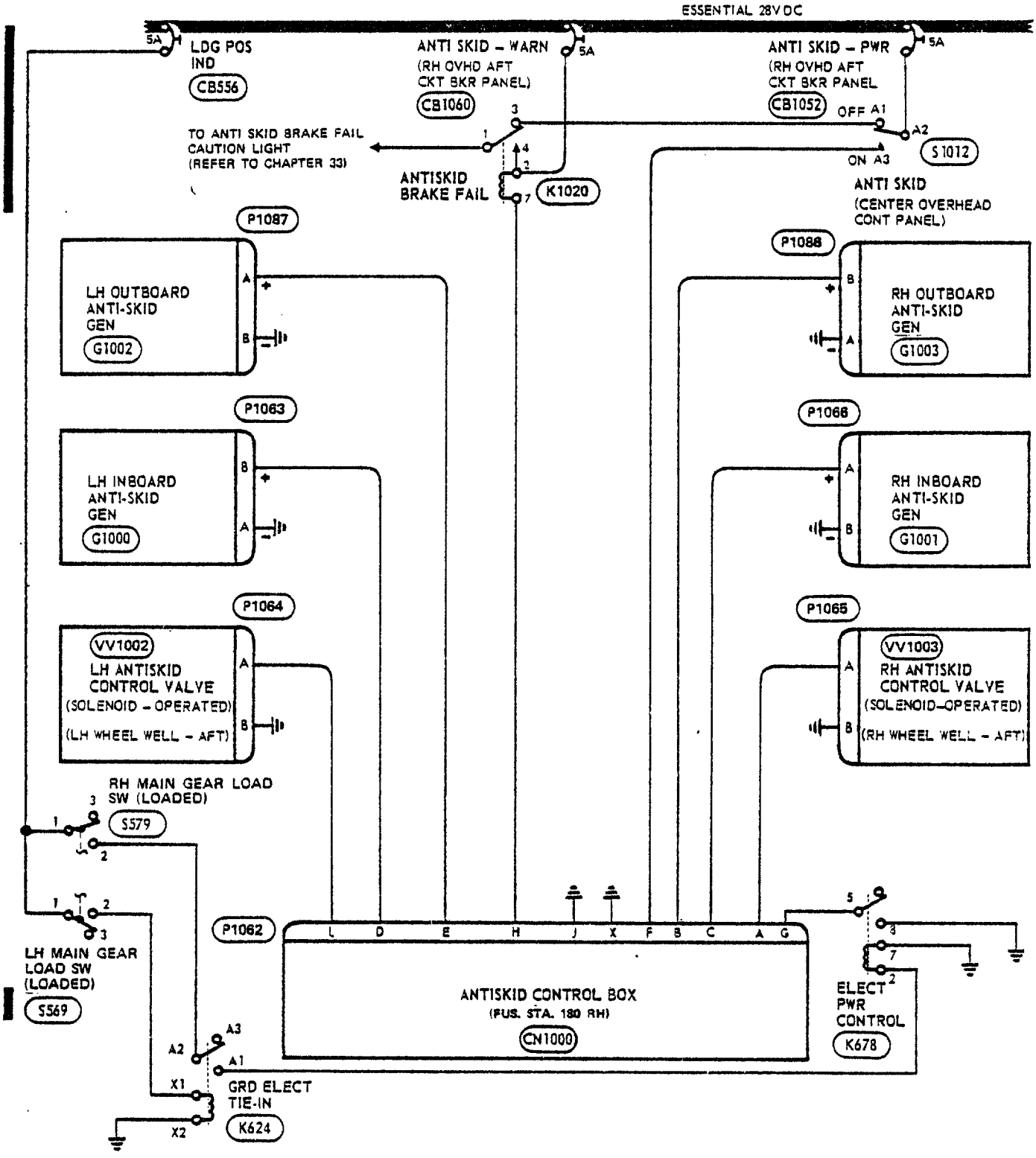

S370-2-33-42A

Antiskid System Electrical Schematic -
NA-370-2 Through -6, -8 and -9
Figure 32-41-201

CAUTION: ROTATE GENERATORS ONLY IN DIRECTION OF NORMAL WHEEL ROTATION. BACKWARD ROTATION WILL DAMAGE ANTISKID CONTROL BOX.

(5) Twist wheel-driven generator shaft, between thumb and forefinger, in direction of normal wheel rotation. This twisting action simulates a skid condition by generating and stopping voltage suddenly. The brake should release as evidenced by movement of the brake lining and audible cycling of anti-skid control valve.

(6) Release brake pedal and repeat test for opposite wheel.

(7) Turn "ANTI SKID" switch to OFF.

(8) Disconnect hydraulic test stand if used and external electrical power.

3. Operational Check of Wheel-driven Generators

A. Obtain a 20,000-ohms-per-volt volt-ohm-milliammeter, and perform the following test:

(1) Remove hub caps from main wheels.

(2) Disconnect electrical plug from antiskid control box.

(3) Connect voltmeter negative lead to applicable pin of electrical plug for generator being checked and positive lead to ground. (See figure 32-41-201).

CAUTION: ROTATE GENERATORS ONLY IN DIRECTION OF NORMAL WHEEL ROTATION. BACKWARD ROTATION WILL DAMAGE ANTISKID CONTROL BOX.

(4) Rotate right-hand inboard or outboard wheel-driven generator shaft in direction of normal wheel rotation, by twisting between thumb and forefinger. A voltage reading should be indicated.

(5) Connect voltmeter negative lead to applicable pin of electrical plug for generator being checked, and repeat step (4) for left generators. (See figure 32-41-201).

(6) When test is completed, connect control box electrical plug, and install hub caps on wheels.

4. Operational Check of Fail-safe System

A. Proceed as follows:

(1) Connect 28-volt dc electrical power to utility external power receptacle. (Refer to Chapter 6, figure 6-00-3).

(2) Place "ANTI SKID" switch at ON. The "ANTI SKID BRAKE FAIL" light should stay off.

NOTE: The "ANTI SKID BRAKE FAIL" light may flash on briefly when the "ANTI SKID" switch is turned on.

(3) Disconnect electrical plug from right-hand antiskid control valve. The caution light should come on in about 3 seconds.

(4) Rotate left then right wheel driven generator shaft briskly in direction of normal wheel rotation. Check that no braking action occurs.

(5) Place "ANTISKID" switch at OFF. The caution light should go on.

(6) Reconnect right-hand valve electrical connector.

(7) Place "ANTISKID" switch at ON. The caution light out.

(8) Repeat procedure for left valve.

(9) On completion of check, turn all switches off, connect control valve electrical plug, and disconnect external power.

MAINTENANCE PRACTICES

ANTI-SKID BRAKE SYSTEM — DESCRIPTION/OPERATION, S/N 370-1 AND -7, AND S/N 380-1 AND SUBS

General

The wheel brake anti-skid system is a fully modulated, electrically controlled, hydraulically operated system that allows a minimum landing roll and reduced tire wear. The fully modulated system provides simultaneous proportional hydraulic pressure to all four main gear wheel brakes. During normal skid control, the left inboard and outboard wheel brakes are controlled simultaneously and independently of the right inboard and outboard wheel brakes. In addition, touchdown-locked-wheel protection is provided simultaneously to all wheels. System test and brake valve dump monitoring is also provided.

The antiskid system consists of four wheel driven DC generators that produce a voltage proportional to wheel rotational speed, an antiskid control box to monitor wheel generator output, and two pressure operated control valves that reduce and apply pressure as directed by the antiskid control box. Each control valve affects pressure to both brakes on one main gear shock strut.

Whenever braking pressure is applied and wheel deceleration exceeds the pre-determined maximum fixed deceleration limits, (indicating a skidding condition), braking pressure for the affective pair of wheels is automatically reduced to allow the wheel to recover before brake pressure is reapplied. Immediately following the initial brake application the antiskid system becomes more sensitive than before, and wheel deceleration of a lesser rate will cause the brake pressure to be reduced and then reapplied. Each time the threshold limits of wheel deceleration is exceeded, the antiskid system will successfully increase in sensitivity. The system will automatically adjust its sensitivity to the amount that is applicable to the braking condition encountered. When braking pressures are reduced below the skid threshold, all tendencies to skid are eliminated and the system reverts to a passive condition, returning braking control to the pilot.

If a malfunction to the antiskid system should occur during braking operations, the ANTISKID swich should be positioned at "OFF," the "ANTISKID OFF" caution light will come on, and complete control of wheel brake system will return to manual braking.

Electrical power for the antiskid system is taken from the 28-volt DC essential bus through the "ANTISKID" circuit breaker on the right overhead circuit breaker panel in the cockpit. The antiskid circuit is activated only when the gear handle is down, and the left and right hand main landing gear load switches are loaded (airplane weight on gear). It is controlled by a three position "ANTISKID" switch on the electrical overhead control panel.

Wheel-Driven Generators

The wheel-driven generators are installed in the main landing gear axles. The armature shaft for each generator is coupled to and driven by a coupling in each main gear wheel hub cap. The hub caps are secured to the wheels by a lock ring and an antirotation pin. (See Figure 32-41-1.) The generators when driven by the rotation of wheels, deliver a voltage proportional to the rotation speed of the wheels to the antiskid control box.

Antiskid Control Box

The antiskid control box is mounted in the electrical equipment rack on the right hand side of the fuselage at FS 150. Electrical power for the control box is taken from the 28-volt DC essential bus through the "ANTISKID" circuit breaker. Each of the following circuits/components are located within the antiskid control box: decerlation detector, variable bleeder, amplifier, and variation detector. Two auxiliary circuits, the touchdown protection and the wheel locked protection circuits are also located within the antiskid control box.

The signals from the wheel driven generators are monitored by the deceleration detector which in turn provides a signal equivalent to the deceleration rate of the wheels to the variable bleeder. In a normal or no skid condition, the signal from the decerleration detector to the variable bleeder does not exceed the threshold of the variable bleeder and is sent to ground. In a skid condition the signal from the deceleration detector will exceed the threshold of the variable bleeder. The excess output of the deceleration detector is sent directly to the amplifier and then to the antiskid valve. The valve in turn reduces pressure to the brake. At the same time the signal is sent to the antiskid valve, the variation detector receives teh same signal and returns it to the amplifier for increased output. Also at the same time the output from the variation detector changes the threshold sensitivity of the variable bleeder so that less of

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MAINTENANCE PRACTICES (Continued)

Antiskid Control Box (Cont'd)

the detection signal is bypassed to ground, and more of the deceleration signal is sent to the amplifier. Whenever pedal force is reduced below the skid threshold, the antiskid control reverts to the passive condition and will not cause any reduction in brake pressure. The antiskid system features a locked wheel protection circuit within the antiskid control box that prevents brake application prior to touchdown and prevents wheel lockup during the landing roll. The locked wheel circuit cannot operate unless the locked wheel arming circuit is activated.

The locked wheel arming circuit is activated whenever the antiskid switch is on, the landing gear handle is down and either both left and right main gear load switches are unloaded (airborne) or any aircraft main landing gear wheel speed is above approximately 20 MPH.

When the arming circuit is activated and any wheel speed is below 12 MPH, both locked wheel detector circuits receiving signals from all four wheel speed generators, sense a locked wheel and send a full dump signal to their respective left and right antiskid amplifiers. All brakes will be released. As the wheels spin up and all exceed 12 MPH, the dump signal will be removed and normal operation will resume.

Antiskid Control Valves

The antiskid valve is a pressure operated two stage flapper nozzle servo-valve mounted on the aft side of the forward bulkhead of the nose gear wheel well (FS 50). The valves are actuated by signals received from the antiskid control box. The antiskid control valve consists of a polarized electric torque motor and two stages of hydraulic power amplification.

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ANTISKID BRAKE SYSTEM - TROUBLE SHOOTING NA-370-1 AND -7 AND NA-380-1 AND LATER

1. Trouble Shooting Antiskid Brake System

The following trouble shooting information is applicable only to the antiskid portion of the brake system and is based on the assumption that electrical power is applied to the airplane and that hydraulic power is available to the system.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. ANTISKID SYSTEM INOPERATIVE - ALL WHEELS.		
(1) Faulty antiskid control box.	Check voltage at control valve electrical plugs while simulating a skid condition. (Refer to 32-41A-2) Voltage should be at least 18 volts.	Replace control box.
(2) Fail-safe system in operation.	Check electrical plugs at control box and control valves for security.	Connect or tighten electrical plugs.
B. ANTISKID SYSTEM INOPERATIVE - ONE WHEEL.		
(1) Wheel-driven generator faulty.	Check operation of generator in affected wheel. (Refer to 32-41A-2.)	Replace faulty generator
(2) Faulty antiskid control box.	Disconnect electrical plug at affected control valve. Check voltage at pin "C" of electrical plug while simulating a skid condition (Refer to 32-41A-2). Voltage must be at least 18 volts.	If voltage is not at least 18 volts, replace control box. If ample voltage is present, refer to probable cause (3).
(3) Faulty antiskid control valve.	Refer to isolation procedure for probable cause (2).	If voltage is 18 volts or better, replace control valve.
(4) Hub cap not driving generator.	Check that drive adapter and cap alignment pin are intact and undamaged. NOTE: Be sure that alignment pin engages hole in wheel when cap is installed.	Replace cap.

ANTISKID BRAKE SYSTEM - MAINTENANCE PRACTICES NA-370-1 AND -7, AND NA-380-1 AND LATER

1. Replacement of Wheel-driven Generator

Removal and installation procedures for both inboard and outboard wheel generators are the same. The parts referred to are in figure 32-41-1.

A. Removing Wheel-driven Generator

(1) Remove lock-ring securing hub cap to main wheel.

(2) Remove hub cap from main wheel.

(3) Remove two bolts retaining generator to axle.

(4) Pull generator from axle and disconnect electrical plug from generator.

B. Installing Wheel-driven Generator

(1) Connect electrical plug in axle to generator.

(2) Position generator in axle and install two retaining bolts through wheel nut, axle and generator.

(3) Perform operational check of antiskid system.

(4) Install hub cap and lockring on main wheel.

CAUTION: MAKE CERTAIN ANTIROTATION PIN BETWEEN HUB CAP AND WHEEL IS CORRECTLY ALIGNED.

2. Operational Check of Antiskid System

A. To perform operational check of antiskid system proceed as follows:

(1) Remove left and right main landing gear wheel hub caps and open the RH and LH STALL WARNING VANE HEATER and PITOT HTR PWR circuit breakers.

(2) Connect external 28 VDC electrical power to airplane.

(3) Close the HYD PUMP CONT and LANDING GEAR - ANTISKID CONT and POS circuit breakers.

(4) Disconnect main landing gear load switches. Position load switches in ground (loaded) position.

(5) Set HYDRAULIC PUMP switch to AUTO and ANTISKID switch to ON.

(6) Depress both left and right brake pedals.

(7) Position and hold ANTISKID switch at TEST for about 5 seconds and release. Check for the following:

(a) Pilot's and copilot's LH and RH ANTISKID lights should come on for 2 to 3 seconds, then go out.

(b) Both RH and LH brakes should release and then reapply.

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(8) Position main gear load switching to unloaded (airborne) position. Check for the following:

(a) Both pilot's and copilot's LH and RH ANTISKID lights should be out and stay out as long as load switches are in the unloaded position.

(b) Both RH and LH brakes should be released.

(9) Position ANTISKID switch to TEST position. Check for the following:

(a) Both pilot's and copilot's LH and RH ANTISKID lights should illuminate and stay illuminated as long as the switch is in the TEST position.

(b) Both LH and RH brakes should be applied.

(10) While monitoring ANTISKID lights in cockpit, position main gear load switches to loaded (ground) position. Check for the following:

(a) Both pilot's and copilot's LH and RH ANTISKID lights remain out.

(b) Both LH and RH brakes should apply 2 to 4 seconds after load switches are placed in loaded (ground) position.

(11) Position ANTISKID switch to TEST position and hold.

CAUTION: ROTATE GENERATORS ONLY IN DIRECTION OF NORMAL ROTATION. BACKWARD ROTATION WILL DAMAGE ANTI-SKID CONTROL BOX.

(12) Turn left outboard main wheel-driven generator shaft briskly in the direction of normal wheel rotation. Check for the following:

(a) Momentary releasing of both left main wheel brakes.

(13) Discontinue turning left outboard main wheel-driven generator shaft, then repeat step (12) for left inboard main wheel-driven generator.

NOTE: When performing steps (12) and (13), only the left main wheel brakes should release and reapply.

(14) Release ANTISKID TEST switch.

(15) Turn left outboard main wheel-driven generator shaft briskly in the direction of normal wheel rotation. Check for the following:

(a) Both LH and RH main wheel brakes should release and reapply.

(16) Discontinue turning left outboard main wheel-driven generator shaft, then repeat step 15 for the left inboard main wheel-driven generator.

NOTE: When performing steps (15) and (16) all brakes should release and reapply.

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(17) Position ANTISKID switch to TEST position and hold.

(18) Turn right outboard main wheel-driven generator shaft briskly in the direction of normal wheel rotation. Check for the following:

(a) Momentary releasing of both right main wheel brakes.

(19) Discontinue turning right outboard main wheel-driven generator shaft, then repeat step (18) for the right inboard main wheel-driven generator.

NOTE: When performing steps (18) and (19), only the right main wheel brakes should release and reapply.

(20) Release ANTISKID TEST switch.

(21) Turn right outboard main wheel-driven generator shaft briskly in the direction of normal wheel rotation. Check for the following:

(a) Both RH and LH main wheel brakes should release and reapply.

(22) Discontinue turning right outboard main wheel-driven generator shaft, then repeat step (21) for the right inboard main wheel generator.

NOTE: When performing steps (21) and (22) all brakes should release and reapply.

(23) Set ANTISKID switch to OFF. Set HYDRAULIC-PUMP switch to OFF and RESET.

(24) Remove external electrical power from airplane.

(25) Install left and right main landing wheel hub caps and outboard wheel fairings.

(26) Connect and safety main landing gear load switches.

(27) Close RH and LH STALL WARNING VANE HEATER and PITOT HTR PWR circuit breakers.

3. Operational Check of Wheel-driven Generators

A. Obtain a 20,000-ohms-per-volt, volt-ohm-millimeter and perform check as follows:

(1) Remove lock-ring securing hub cap to main wheel.

(2) Disconnect plugs P964 and P965 from antiskid control box in electrical equipment rack at fuselage station 150 RH.

(3) To check inboard generators connect positive lead of meter to pin "G" of applicable plug disconnected from antiskid control box and negative lead to pin "X". For outboard generators connect positive lead of meter to pin "A" of applicable plug disconnected from antiskid control box and negative lead to pin "B".

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(4) With leads of meter connected to applicable pins of electrical plug of generator being checked, have assistant rotate generator shaft in direction of normal wheel rotation by twisting shaft between thumb and forefinger.

(5) Observe meter to make certain a voltage reading is indicated as generator shaft is rotated.

(6) Repeat steps (4) and (5) for each generator to be checked.

(7) When checks are completed, connect electrical plugs to antiskid control box and install main wheel hub caps and lock rings. Make certain antirotation pins are correctly engaged.

4. Operational Check of Wheel Driven Generator Output Voltages Versus RPM

A. Obtain a variable and reversible speed drive (RPM accuracy 0.05%),Voltmeter (John Flukes Model 8038 or equivalent). Refer to figure 32-41A-203 and perform check as follows:

(1) Remove lock-ring securing hub cap to main wheel.

(2) Disconnect plugs P964 and P965 from anti-skid control box in electrical equipment rack at fuselage station 150 RH.

(3) Connect voltmeter to the correct pins and polarity in plug (P964 or P965). See figure 32-41A-203 for set-up to each generator.

NOTE: Generator to voltmeter set-up must be positive (+) to positive and minus (-) to minus for correct polarity.

(4) With leads of voltmeter connected to applicable pins of electrical plug of generator being checked, have assistant rotate generator (normal wheel rotation) with variable speed drive to a known rpm.

(5) Observe voltmeter voltage at known rpm and check chart (figure 32-41A-20), and see if voltage output is within limits. Make two different known rpm checks (preferably 1800 and 3600 rpm).

(6) Repeat steps (3) to (5) for each generator to be checked.

(7) When checks are completed, connect electrical plugs to anti-skid control box and lockwire plugs.

(8) Install hub caps and lock rings.

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5. Operational Check of Anti-Skid Wheel Generator, Using an Anti-Skid Inertial Drive Tool

A. Obtain a anti-skid inertial drive tool from Goodyear Aerospace Corp., (P/N 5E6-2373) or may be locally manufactured (See figure 32-41A-204). This tool attaches to a wheel generator to simulate a wheel spin-up.

(1) Remove lock-ring securing hub cap, and remove hub cap.

(2) Carefully install anti-skid inertial drive tool to anti-skid generator and install hub cap lock-ring to lock tool on wheel.

NOTE: Anti-skid inertial drive tool may be held in place by hand (not using lock-ring), while spining flywheel by hand.

(3) Connect external 28 VDC electrical power to airplane.

(4) Close the HYD PUMP CONT and LANDING GEAR - ANTISKID CONT and POS circuit breakers.

(5) Set HYDRAULIC PUMP switch to AUTO and ANTISKID switch to ON.

(6) Depress both left and right brake pedals.

(7) Briskly spin (direction of normal wheel rotation) the anti-skid inertial drive tool fly-wheel (using finger hole in fly-wheel) until the pilot's and copilot's ANTISKID lights come on. Let fly-wheel coast down thru the skid threshold (below 20 mph), the ANTISKID lights should go out. Check the following:

(a) Serviceable generator - The ANTISKID lights will go out without flickering.

(b) Faulty generator - The ANTISKID lights will flicker on an off (maybe many times) and then off.

(8) Perform these check steps (6) and (7) on each wheel generator.

(9) Set ANTISKID switch to OFF. Set HYDRAULIC-PUMP switch to OFF and RESET.

(10) Remove external electrical power from airplane.

CAUTION: MAKE CERTAIN ANTIROTATION PIN BETWEEN HUB CAP AND WHEEL IS CORRECTLY ALIGNED.

(11) Install left and right main landing wheel hub caps and lockrings.

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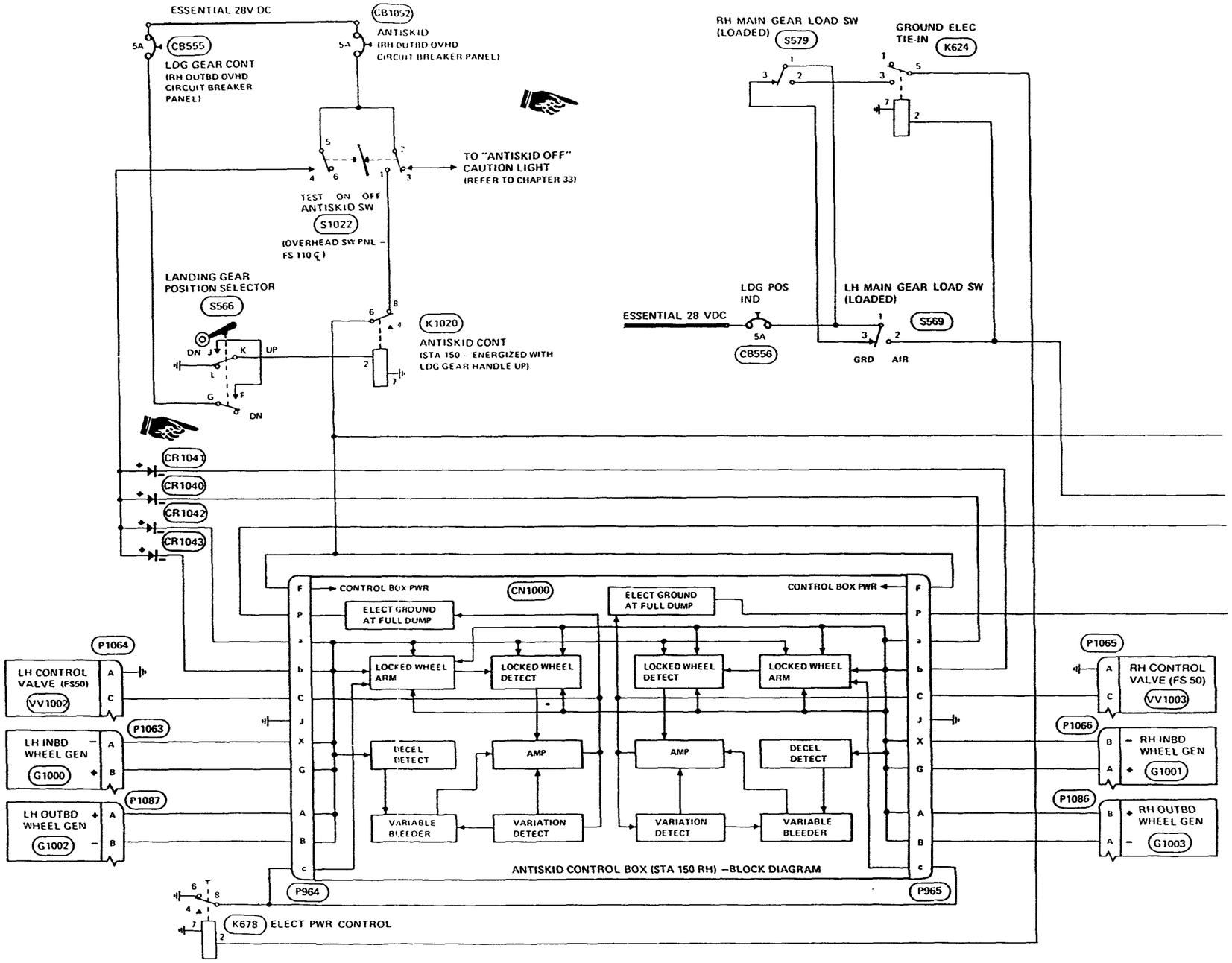

Figure 32-41A-201 — Antiskid System Electrical Schematic — S/N 370-1 and -7, 380-1 Thru -4, -6 Thru -9 and -11 (Sheet 1 of 2)

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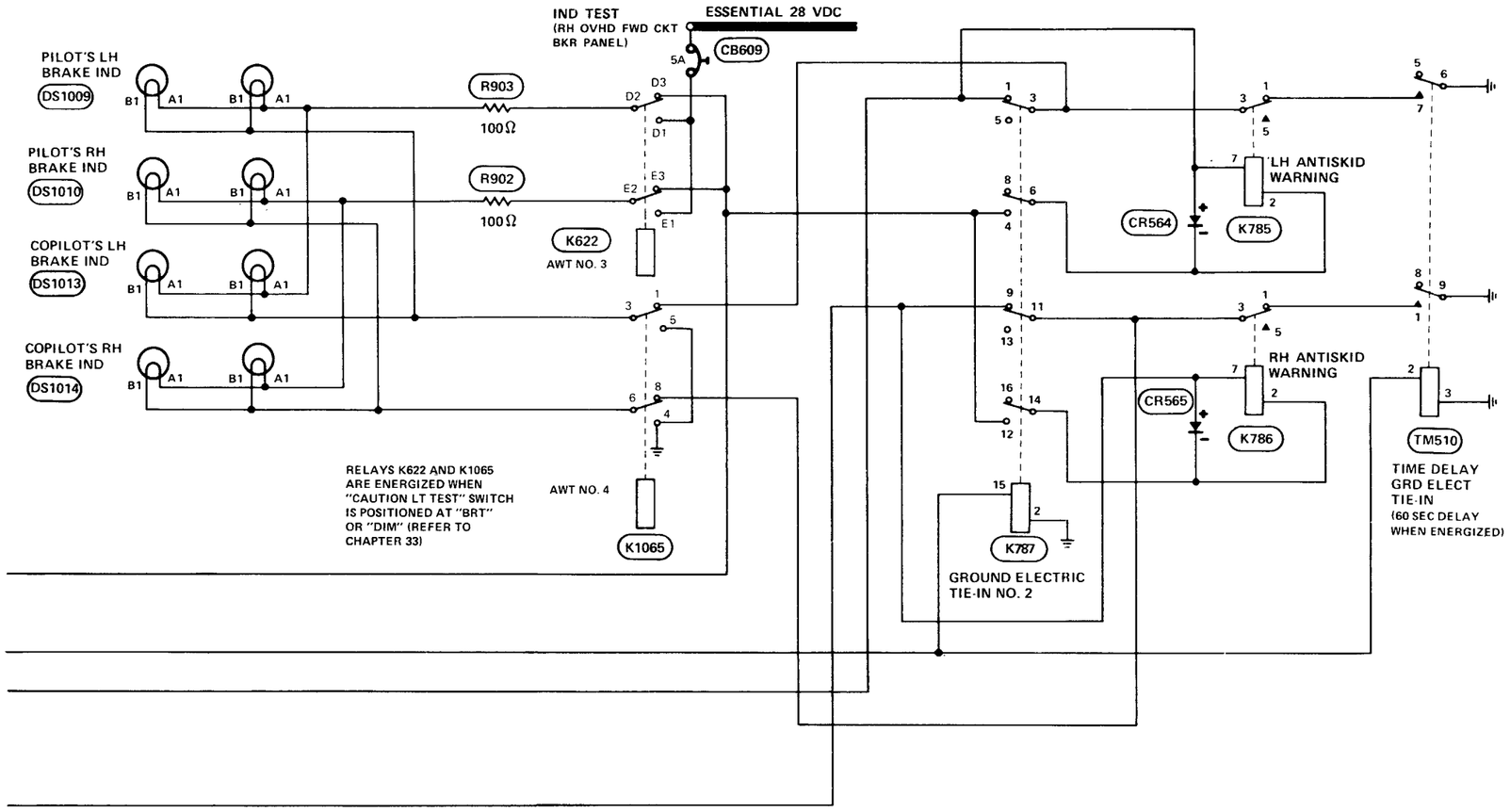

S380-2-41-4B

Figure 32-41A-201 — Anitskid System Electrical Schematic — S/N 370-1 and -7, 380-1 Thru -4, -6 Thru -9 and -11 (Sheet 2 of 2)

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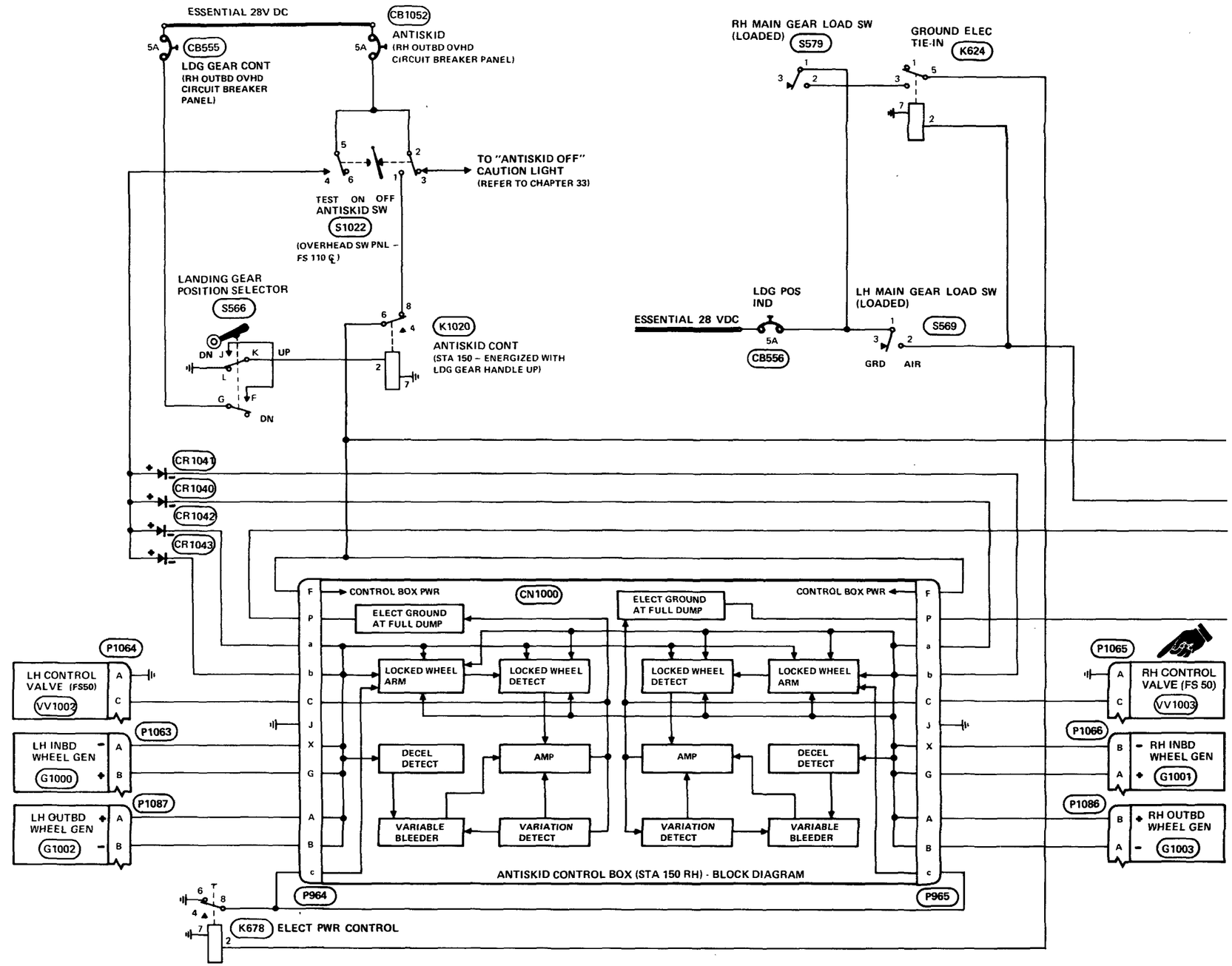

Figure 32-41A-202 — Antiskid System Electrical Schematic — S/N 380-5, -10, -12 and Subs (Sheet 1 of 2)

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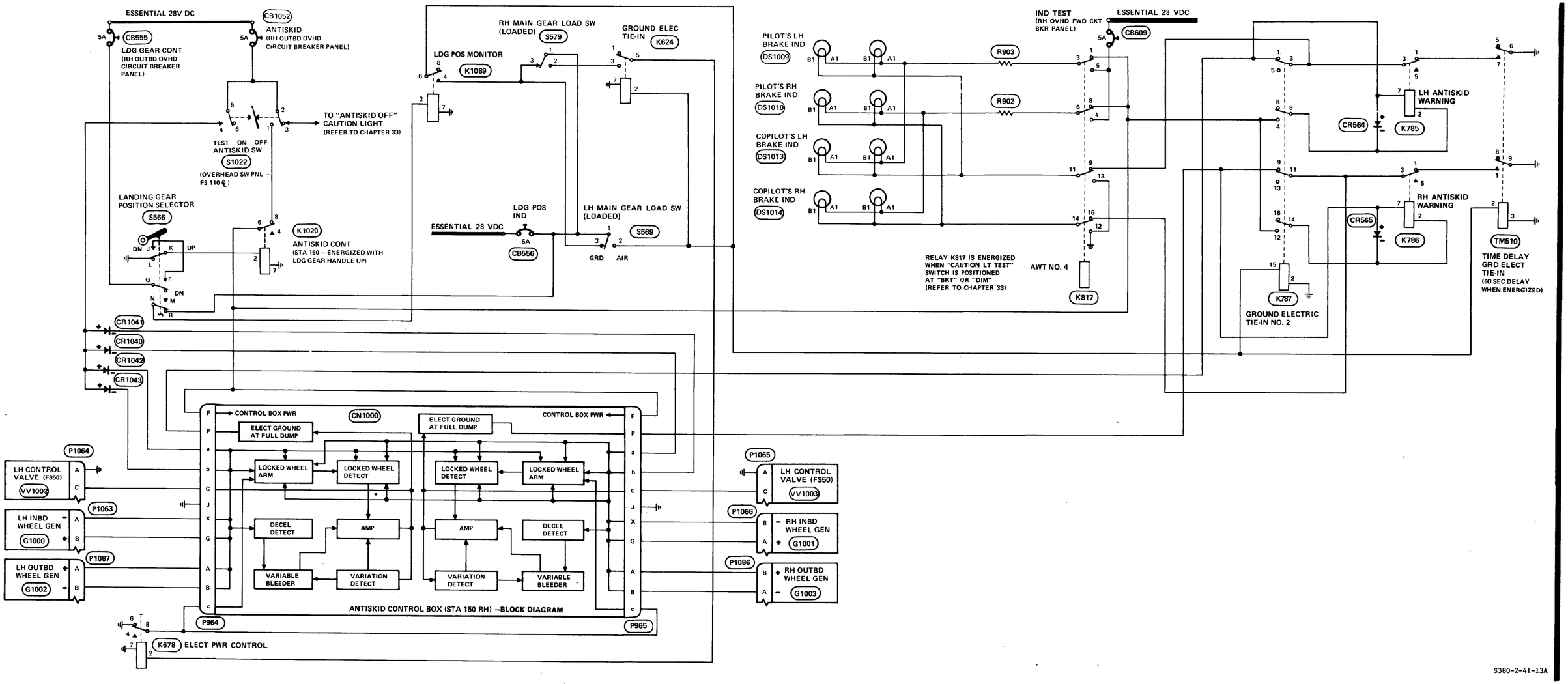
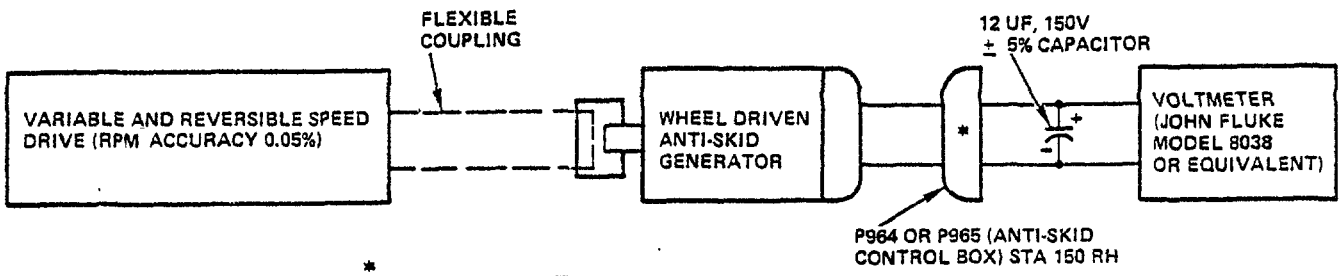

ESSENTIAL 28V DC
CB1052
ANTISKID
(RH OUTBD OVHD CIRCUIT BREAKER PANEL)
5A
CB555
LDG GEAR CONT
(RH OUTBD OVHD CIRCUIT BREAKER PANEL)
TO "ANTISKID OFF" CAUTION LIGHT (REFER TO CHAPTER 33)
TEST ON OFF
ANTISKID SW
S1022
(OVERHEAD SW PNL – FS 110 ℄)
LANDING GEAR POSITION SELECTOR
S566
DN
UP
K1020
ANTISKID CONT
(STA 150 – ENERGIZED WITH LDG GEAR HANDLE UP)
RH MAIN GEAR LOAD SW (LOADED)
S579
GROUND ELEC TIE-IN
K624
LDG POS MONITOR
K1089
LDG POS IND
ESSENTIAL 28 VDC
CB556
LH MAIN GEAR LOAD SW (LOADED)
S569
GRD
AIR
PILOT'S LH BRAKE IND
DS1009
PILOT'S RH BRAKE IND
DS1010
COPILOT'S LH BRAKE IND
DS1013
COPILOT'S RH BRAKE IND
DS1014
R903
R902
IND TEST
(RH OVHD FWD CKT BKR PANEL)
ESSENTIAL 28 VDC
CB609
RELAY K817 IS ENERGIZED WHEN "CAUTION LT TEST" SWITCH IS POSITIONED AT "BRT" OR "DIM" (REFER TO CHAPTER 33)
AWT NO. 4
K817
LH ANTISKID WARNING
CR564
K785
RH ANTISKID WARNING
CR565
K786
K787
GROUND ELECTRIC TIE-IN NO. 2
TM510
TIME DELAY GRD ELECT TIE-IN
(60 SEC DELAY WHEN ENERGIZED)
CR1041
CR1040
CR1042
CR1043
CN1000
CONTROL BOX PWR
ELECT GROUND AT FULL DUMP
LOCKED WHEEL ARM
LOCKED WHEEL DETECT
DECEL DETECT
AMP
VARIABLE BLEEDER
VARIATION DETECT
ANTISKID CONTROL BOX (STA 150 RH) –BLOCK DIAGRAM
P964
P965
K678
ELECT PWR CONTROL
P1064
LH CONTROL VALVE (FS50)
VV1002
P1063
LH INBD WHEEL GEN
G1000
P1087
LH OUTBD WHEEL GEN
G1002
P1065
LH CONTROL VALVE (FS50)
VV1003
P1066
RH INBD WHEEL GEN
G1001
P1086
RH OUTBD WHEEL GEN
G1003

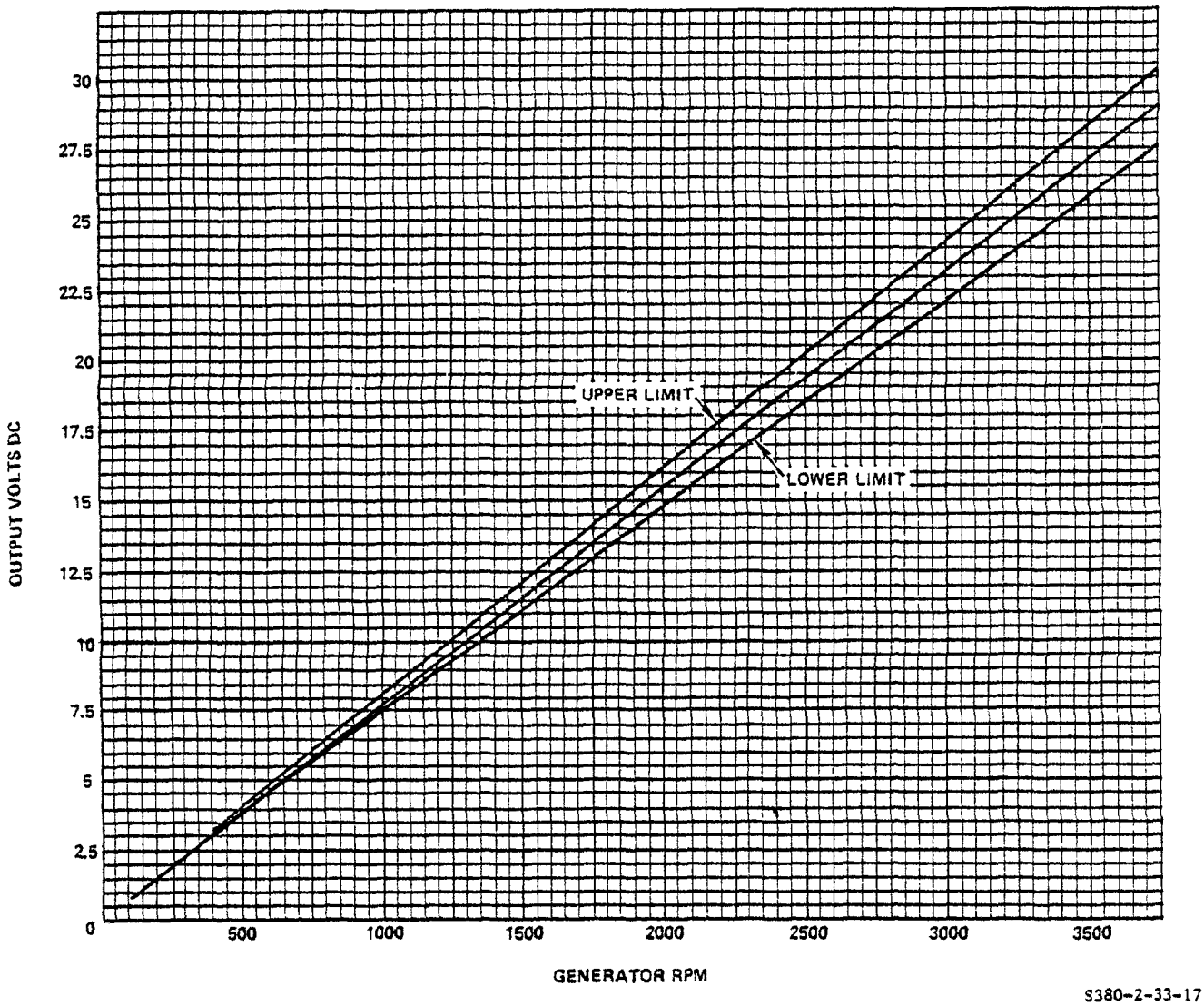
S380-2-41-13A


FLEXIBLE COUPLING
12 UF, 150V ± 5% CAPACITOR
VARIABLE AND REVERSIBLE SPEED DRIVE (RPM ACCURACY 0.05%)
WHEEL DRIVEN ANTI-SKID GENERATOR
*
+
-
VOLTMETER (JOHN FLUKE MODEL 8038 OR EQUIVALENT)
P964 OR P965 (ANTI-SKID CONTROL BOX) STA 150 RH

*

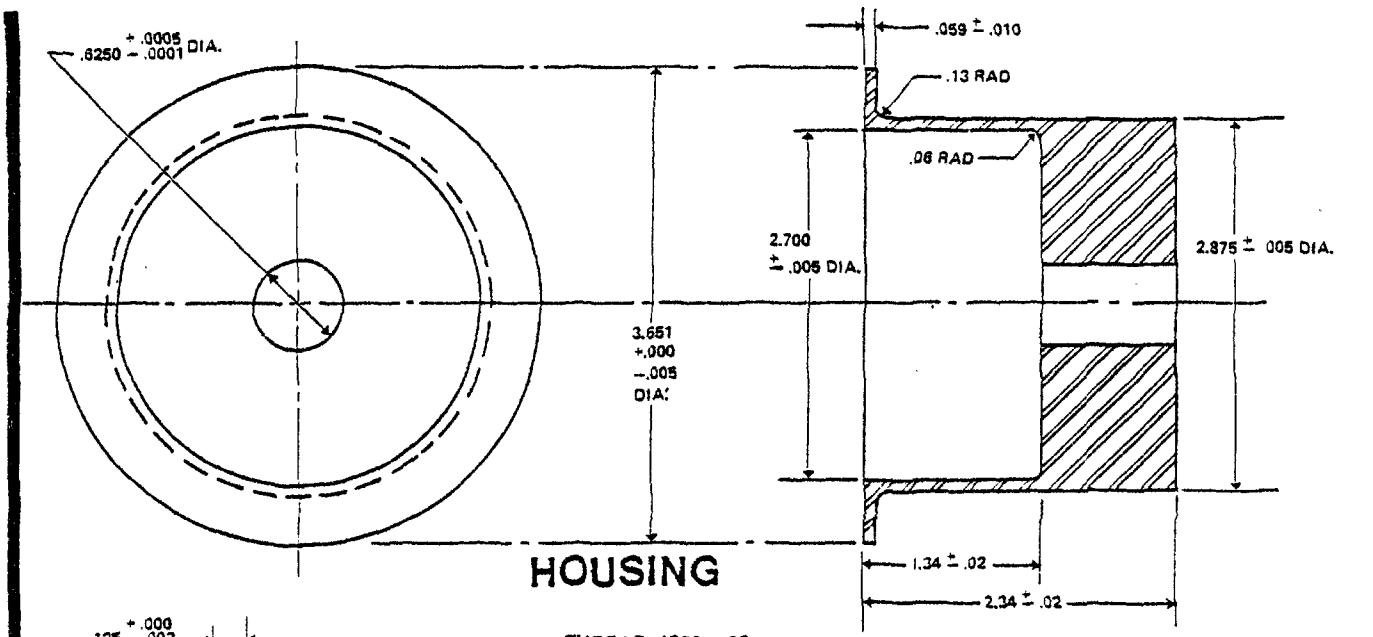
POLARITY SET-UP			
GEN LOCATION	PLUG	PIN	ROTATION
RH OUTBD	P965	A(+), B(-)	CW
LH INBD	P964	X(-), G(+)	CW
LH OUTBD	P964	A(+), B(-)	CCW
RH INBD	P965	X(-), G(+)	CCW

NOTE:
- GENERATOR TO VOLTMETER SET-UP MUST BE POSITIVE (+) TO POSITIVE AND MINUS (-) TO MINUS
- THIS CHECK IS FOR AIRPLANES NA-370-1 AND -7 AND NA-380-1 AND LATER

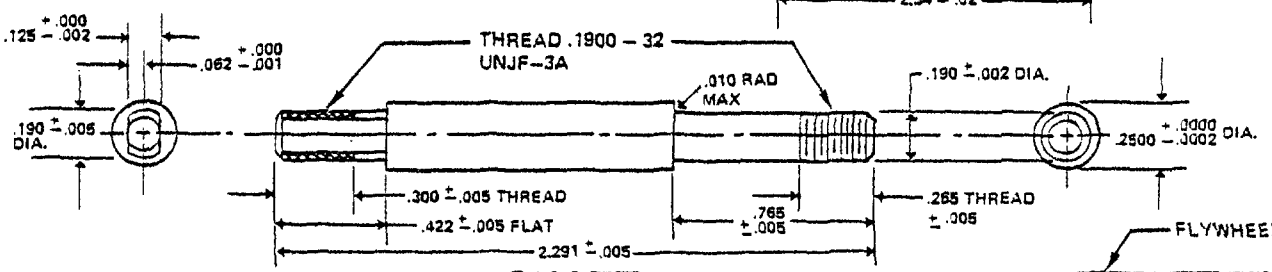


S380-2-33-17

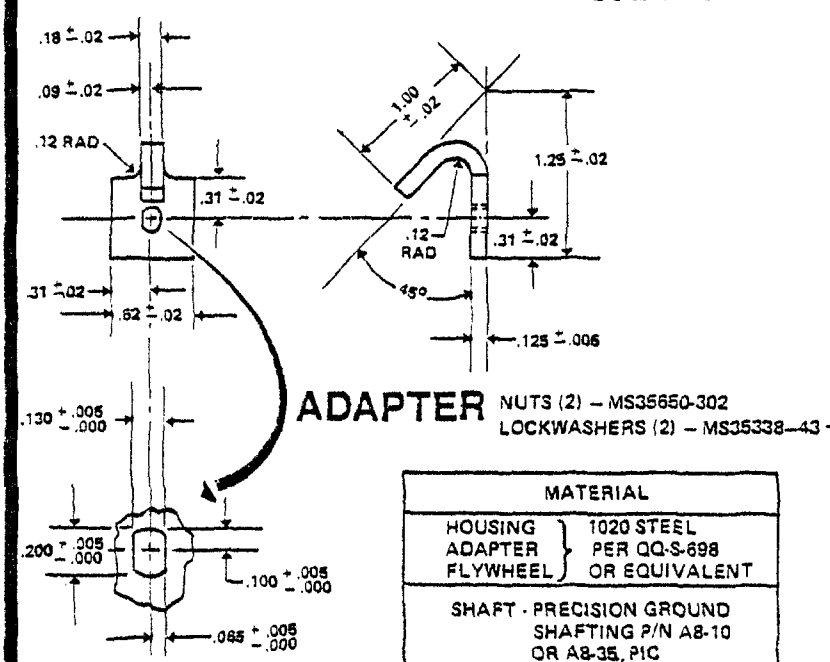
Wheel Driven Anti Skid Generator
Output Voltages Versus RPM
Figure 32-41A-203



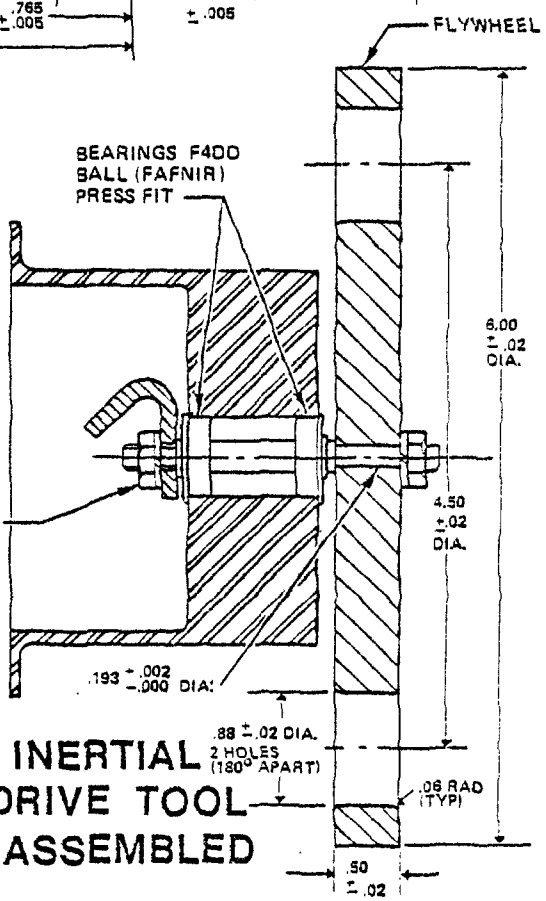
HOUSING



SHAFT



ADAPTER NUTS (2) – MS35650-302
LOCKWASHERS (2) – MS35338-43



INERTIAL DRIVE TOOL - ASSEMBLED

MATERIAL	
HOUSING, ADAPTER, FLYWHEEL	1020 STEEL PER QQ-S-698 OR EQUIVALENT
SHAFT - PRECISION GROUND SHAFTING P/N A8-10 OR A8-35, PIC DESIGN CORP; VAN NUYS, CALIF.	
TREATMENT	
HOUSING, ADAPTER, FLYWHEEL	CADMIUM PLATE PER QQ-P-416, CLASS I, TYPE III

NOTE
All dimensions are in inches

S380-2-33-17

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Anti-Skid Inertial Drive Tool
Figure 32-41A-204

STEERING - DESCRIPTION AND OPERATION

1. Description of Nose Wheel Steering System

The nose wheel steering system controls the direction of the airplane during taxiing, take-off, and landing. The system is electrically actuated, hydraulically operated, and mechanically controlled by movement of the rudder pedals. Steering is engaged and controlled from either the pilot's or copilot's side of the cockpit by actuation of the steering control push-button switch on the control wheel and movement of the rudder pedals. The steering system is alternately engaged and disengaged by actuation and release of the steering control push-button switches.

The steering system receives electrical power from the 28-volt dc essential bus and hydraulic power from the airplane normal or auxiliary hydraulic system. In case of normal hydraulic system failure, the auxiliary hydraulic system supplies 3000 psi of hydraulic pressure to the nose wheel steering system for a limited number of operations. The system is electrically actuated by operation of the relays, steering control box, command and feed-back potentiometers, nose and main gear load switches, and solenoid-operated steering control valve. When the weight of the airplane is on the nose gear and main gear shock struts, the relays (through operation of the nose gear load switch and cabin pressure dump relay) complete the electrical circuit to the solenoid operated hydraulic steering control valve. One relay also acts as a holding relay which keeps the steering system energized, allowing the steering control switch to be released, thus enabling the pilot to perform other activities while taxiing the airplane. The system is mechanically controlled by actuation of a command potentiometer which is connected to the rudder pedal linkage. An amber caution light is illuminated on the ground when either the main or the stand-by system is de-energized. Hydraulic pressure from the hydraulic system is routed directly to the steering control valve. The steering control valve is a dual unit, consisting of three solenoid-operated shutoff valves. One solenoid valve admits hydraulic pressure to the steering control solenoids. The two identical steering control solenoids admit hydraulic pressure to the steering actuating cylinder. One control solenoid admits hydraulic pressure to one side of the steering cylinder piston; the other solenoid admits hydraulic pressure to the opposite side of the steering cylinder piston. The de-energized solenoid always positions its respective cylinder port to hydraulic system return. Mechanical linkage connects the steering cylinder piston with the nose wheels. Hydraulic pressure applied to the steering cylinder piston through the mechanical linkage causes the nose wheels to move in the desired direction.

When the rudder pedals and rudder are moved out of the neutral position, the command potentiometer produces a current flow within the steering control box to overcome the ac current flow produced by the oscillator circuit. This causes one of the two steering control valve control solenoids to remain energized. Hydraulic pressure is then steadily applied to one side of the steering cylinder piston, moving the nose wheels to the desired position. Movement of the nose wheels causes the feed-back potentiometer to gradually nullify the current from the command potentiometer.

When the nose wheels have reached the desired position, the feed-back potentiometer will be nulled (cancelled), stopping the current flow from the command potentiometer. When the command potentiometer and the feed-back potentiometer are in balance (nose wheels turned to rudder pedal position), the steering control valve opens both cylinder ports to return. As forces acting on the nose wheels tend to return the wheels to a trail position, the feed-back potentiometer is thrown out of balance with the command potentiometer, and a dc signal again energizes the proper solenoid to return the wheels to the position selected by the rudder pedals. Movement of the rudder pedals and rudder toward neutral will again cause the command potentiometer to control operation of the steering control valve control solenoids. When the nose wheel steering system is de-energized, both sides of the steering cylinder are ported to the hydraulic system return through the steering control valve.

The nose gear steering system consists of a dual electrical system: the main system and the stand-by system. A main steer fail (amber) caution light, operating in conjunction with the master caution light, is illuminated on the ground at any time the main steering system is inoperative. Two identical green indicator lights are illuminated on the ground, operating independently of the master caution light, to indicate when either the main or the stand-by system is energized. When the main system fails, electrical power is automatically transferred to the stand-by system. (See figure 32-50-2.) Either system can be selected manually by positioning the nose gear steering system select switch to the desired position. The select switch is on the air conditioning system control panel.

5. Operation of Nose Wheel Steering System

A dual electrical system is provided, comprising a main system and a standby system. Electrical actuation of the normal steering system is controlled by a nose gear steering system selector switch (figure 32-50-2 or 32-50-3), steering control switches on the control wheels, the main circuit breakers, nose and main gear load switches, and relays. The relays complete the electrical circuit to actuate the nose wheel steering system and to change over to the standby system if the main system fails. Electrical actuation of the standby system is controlled by a "NOSE GR STEER" selector switch, the standby circuit breaker, and relays. (See figure 32-50-2 or 32-50-3.)

When either main gear shock strut is compressed, the electrical circuit is completed from the "LANDING GEAR - POS" circuit breaker through the closed contacts of relay K1089 to the coil of the cabin pressure dump relay. With the "NOSE GR STEER" selector switch in the MAIN position, the circuit is then completed from the "NOSE GEAR - MAIN STEER" circuit breaker through the normally open contacts of the cabin pressure dump relay to the nose gear steering control relay. At the same time, the electrical circuit is completed through the actuated nose gear load switch to the steering control relay. On NA-370 airplanes, actuation of either steering switch will complete the circuit through the control relay in the nose gear steering control relay. On NA-380 airplanes, actuation of either steering switch will complete a circuit through the nose steer ground control relay (K1010) to the nose gear steer control relay (K607). When the control relay (K607) is initially energized, an electrical circuit is completed through contacts of the control

relay through a normally closed set of contacts of the nose steering transfer relay to the solenoid in the steering control valve. The solenoid, when energized, allows hydraulic pressure to enter the control valve. At the same time that the circuit is completed to the solenoid of the steering control valve, it is also completed to the nose gear steering main control box. Electrical power is then supplied to a pair of transistors within the control box. The transistors form an oscillator that supplies an ac signal at a frequency of about 30 cps to one half of a control coil in the micropositioner relay. The micropositioner relay consists of a primary coil, two momentary contacts, and a swinging contact arm which is pivoted at the center and electrically connected to ground. Alternating current flow through the primary coil causes a magnetic force to exist between the pivoted center arm and the relay coil. This attracts the pivoted arm against one of the two contacts and pulls it away from the other contact. When the alternating current changes polarity, the arm is repelled by magnetic force away from the contact and is pushed against the opposite contact. The repelling and attracting force occurs at the transistor oscillator frequency rate. When the swinging contact arm is in contact with one or the other of the two contacts, an electrical circuit is completed from ground through the micropositioner contact, through a set of normally closed contacts of the nose steering transfer relay to one of two identical steering solenoids in the steering control valve, through another set of normally closed contacts of the nose steering transfer relay and through the nose gear steering control to the 28-volt dc essential bus. When the swinging arm is in contact with the opposite contact of the relay, the opposite control solenoid in the steering control valve is energized in the same manner as the other solenoid.

Under this condition, there is no dc flow through the lower half of the micropositioner relay coil because the command and feed-back potentiometers are electrically balanced, causing the same voltage potential to exist at ends of the lower half of the coil. Movement of the rudder from neutral position changes the position of the center tap on the command potentiometer and upsets the electrical balance between the command and feed-back potentiometer. One end of the lower half of the micropositioner relay coil will become more negative or positive with relation to the opposite end. This causes a dc flow through the lower half from negative to positive and produces a magnetic force between the micropositioner center swinging arm and the coil. The polarity of the magnetic force is determined by the direction of dc flow through the lower half of the coil, which in turn is determined by the direction that the center tap on the command potentiometer was moved by the rudder pedals. The resulting magnetic force through the lower half of the coil is enough to overcome the magnetic force produced by the ac flow through the upper half of the micropositioner relay coil. The magnetic force causes the pivoted center arm in the micropositioner relay to be held in contact with one of the two relay contacts. Which of the two relay contacts that is held in contact is determined by the polarity of the magnetic force. With the center arm held in contact, the electrical circuit is completed to one of the two identical steering control solenoids in the steering control valve. The solenoid is energized which admits hydraulic pressure through the solenoid shufoff valve to one side of the steering cylinder piston. Movement of the

steering cylinder piston is transmitted through mechanical linkage to a collar on the nose gear shock strut. Rotation of the collar turns the nose wheels in the desired direction. Movement of the steering cylinder piston also repositions the center tap on the feed-back potentiometer. When the steering cylinder piston has moved far enough to position the nose wheels in a position that corresponds with the position of the rudder pedals, the feed-back potentiometer center tap will have moved to a position to match the position of the command potentiometer center tap. The command and feed-back potentiometers are again electrically balanced. In this condition, the voltage potential at both ends of the lower half of the micropositioner relay coil are again equal. With equal voltages at both ends of the coil, there is no dc flow and no magnetic force is produced. This allows the magnetic force produced by the current flow in the upper half of the coil to control the movement of the center arm on the relay. The solenoid in the steering control valve is no longer constantly held in the energized position, which shuts off a constant source of hydraulic pressure to the one side of the steering cylinder piston. Movement of the rudder pedal toward neutral will again upset the electrical balance between the command and feed-back potentiometers. This causes the process to be repeated, and will position the nose wheels to the new position corresponding to the new position of the rudder pedals. The nose wheels will continue to travel in the desired direction as long as the position of the center tap of the command potentiometer keeps ahead of the repositioning center tap of the feed-back potentiometer. Steering travel throw is determined by rudder pedal movement and is limited by external stops on the nose gear shock strut.

Either of two failures in the main system will automatically transfer control to the stand-by system: the main right steering monitor and main left steering monitor relays being energized simultaneously, or when the command potentiometer is positioned in one direction and the feed-back potentiometer indicates the nose wheel is in the extreme opposite direction. When both relays are energized simultaneously, the electrical circuit is completed to the nose steering transfer hold relay through sets of normally open contacts of the main right steering monitor and main left steering monitor relays which are connected in series in the nose steering transfer hold relay circuit. When the command potentiometer is positioned in one direction and the feed-back potentiometer indicates the nose wheel is in the extreme opposite direction, a circuit is made through the command potentiometer monitor switch, the feed-back potentiometer monitor switch, and a set of normally open contacts of the main right steering monitor or main left steering monitor relays. Once the nose steering transfer hold relay has been energized, a holding circuit is provided by a set of its own normally open set of contacts. To de-energize the nose steering transfer hold relay, the "NOSE GR STEER" selector switch is momentarily placed at STANDBY and returned to MAIN.

When control is automatically transferred to the stand-by system, the nose steering transfer relay is energized. With the nose steering transfer relay energized, the following will take place simultaneously:

- The solenoid of the steering control valve (which allows hydraulic pressure to enter the steering control valve) which was receiving electrical power through a set of normally closed contacts of the nose steering transfer relay will now receive its electrical power through a set of normally open contacts of the nose steering transfer relay.
- The electrical circuit, from the nose gear steering main control box to either one of the solenoids in the steering control valve which had continuity through a set of normally closed set of contacts of the nose steering transfer relay, is now de-energized.
- The nose gear steering stand-by control box will be energized through a set of normally open contacts of the nose steering transfer hold relay.

Once electrical power has reached the nose gear steering stand-by control box, the same internal functions take place electrically in the stand-by control box as in the main control box. The signal out of the stand-by control box going to either solenoid of the steering control valve, passes through normally closed sets of contacts of the stand-by steering lockout relay and normally open sets of contacts of the nose steering transfer relay. The steering control switches (on the pilot's and copilot's control wheels) are not used in the stand-by system.

3. Steering Control Box

The nose wheel steering control box is mounted in the lower right equipment bay, forward of the windshield. (See figure 32-50-1.) Externally, the box is rectangular in shape with the only protrusions being a fuse holder and an electrical disconnect plug receptacle. Internally, the box consists of resistors, capacitors, crystal diodes, two transistors, and a polarized micropositioner relay. The resistors establish the operating voltages for the transistors, while the diodes act as check valves to control the routing of electrical current through the circuit. The transistors form an oscillating circuit that operates at a 30 cps frequency that is determined by the rate of change (RC) time constant of capacitor C1 and resistor R3. The micropositioner relay consists of a primary coil which is equally divided into halves, two momentary contacts, and a swinging contact arm, pivoted at the center and electrically grounded. The ends of the swinging arm have tiny magnets. When the swinging arm is in contact with either one of the relay contacts, a circuit is completed from ground through the momentary contact, crystal diode, and the coil of one of the two identical steering control solenoids in the steering control valve. If the swinging arm is in contact with the opposite momentary contact, the circuit will be completed to the opposite steering control solenoid valve.

4. Nose Gear Load Switch

The nose gear load switch is attached to the nose gear upper torque link. (See figure 32-50-1.) The switch is a double-pole, double-throw switch actuated by rotation of a cammed shaft in the mechanism. The shaft is connected by linkage to a fixed point on the nose gear shock strut. Extension and compression of the nose gear shock strut causes the torque links to close and spread apart. Movement of the upper torque link

changes the position of the switch in relation to the switch linkage fixed attach point. This causes rotation of the switch shaft, resulting in actuation or deactuation of the switch.

5. Steering Feed-Back Potentiometers

The nose wheel steering feed-back potentiometers are mounted on the left side of the steering actuating cylinder which is attached to the aft side of the nose gear shock strut. (See figure 32-50-1.) The two feed-back potentiometers are used: one for the main system and one for the stand-by system. Their function is identical in each system. The feed-back potentiometer is cylindrical in shape, mounted in a vertical position, parallel with the steering actuating cylinder piston. The potentiometer consists of a variable resistor having a center tap. The electrical leads are emitted through a three-pin electrical plug at the top of the potentiometer. The lower end of the potentiometer is connected to a bracket which is free to move with actuation of the steering cylinder piston. This provides the means of varying the resistance between the ends of the resistor and the center tap. In operation, movement of the steering actuating cylinder piston changes the value of the feed-back voltage. If, at any time, the value of the command and feed-back potentiometer voltages are unequal, hydraulic pressure is constantly supplied to one side of the steering cylinder piston, which turns the nose wheels in the desired direction. The wheels will continue to turn in that direction until the feed-back and command potentiometer voltages are equal. This causes the steering control valve to shut off a steady supply of hydraulic pressure to the steering actuating cylinder.

6. Steering Command Potentiometer

The steering system command potentiometers are installed on the right side of the aisle below the floor, just aft of the cockpit. (See figure 32-50-1.) The two command potentiometers are used: one for the main system and one for the stand-by system. Their function is identical in each system. The command potentiometer is a variable resistor enclosed in a rectangular case. The potentiometer is electrically connected to the steering control system by means of two three-pin electrical disconnect plugs. An arm attached to the shaft protruding from the case is connected to the rudder control linkage by a push rod. Rotation of the shaft by movement of the rudder pedals provides a means of varying the resistance between the ends of the resistor and the potentiometer center tap. In operation, movement of the rudder pedals changes the value of the resistance between the center tap and the ground end of the resistor. The change in resistance causes a change in the command voltage to the steering system control box. An electrical circuit is then completed to energize the control portion of the steering control valve. The correct solenoid in the steering control valve is held energized, which directs a steady supply of hydraulic pressure to the correct side of the nose wheel steering actuating cylinder. This causes the nose wheels to turn in the desired direction. The wheels will continue to turn until the changes in the feed-back voltage from the feed-back potentiometer equals the command voltage. When this occurs, the electrical circuit to the control portion of the steering control valve is intermittently opened, causing the control valve to shut off a steady flow

of hydraulic pressure to the steering actuating cylinder. This produces a hydraulic fluid lock in both sides of the cylinder piston holding the nose wheel in the new position.

7. Steering Control Relay

The nose wheel steering system control relay is mounted in the wall between the cockpit and the entrance compartment, behind the pilot's seat. (See figure 32-50-1.) The relay holds the steering system electrical circuit energized after the steering switch has been initially depressed, eliminating the need of holding the steering switch depressed while the airplane is being steered. The control relay consists of two individual relays, a control and a hold relay. When the steering switch is initially depressed, an electrical circuit is completed through the coil of the control relay, causing this relay to become energized. Contacts in the relay are actuated, completing a circuit to energize the shutoff solenoid in the steering control valve. At the same time, another set of control relay contacts route electrical power to energize the holding relay coil. The holding relay coil, when energized, acts as a holding circuit which keeps the entire relay and the steering control valve shutoff solenoid energized, permitting the steering control switch to be released. (See figure 32-50-2 or 32-50-3.) The circuit will remain energized until the hold-in circuit is broken. This is done when the nose wheels lift off the runway during the take-off run. At this time, the nose gear load switch is deactuated, breaking the hold-in circuit. The control relay and the shutoff solenoid in the steering control valve are now de-energized, closing off hydraulic pressure to the nose wheel steering actuating cylinder. The hold-in circuit may also be broken by momentarily depressing the steering control switch a second time. This causes an equal voltage to be applied to both ends of the control relay coil, causing the relay to be de-energized and to break the hold-in circuit as the control switch is released.

8. Steering Hold Relay

The nose wheel steering hold relay is mounted in the wall between the cockpit and the entrance compartment, behind the pilot's seat. (See figure 32-50-1.) The relay is placed in the circuit so that it is energized when either steering control switch is held depressed. This completes an electrical circuit through the steering control relay, the steering control switches, and the coil of the steering hold relay. With the steering hold relay energized, an electrical circuit is completed through contacts of the steering hold relay to the shutoff solenoid in the steering control valve. This bypasses the steering control relay, permitting the steering control valve to be energized when either steering control switch is held depressed and the nose gear and either main gear load switch are actuated (strut compressed).

9. Steering Control Valve

The nose wheel steering system control valve is mounted on a bracket at the aft end of the nose wheel well. (See figure 32-50-1.) The steering control valve is a double valve, consisting of a solenoid shutoff valve and two solenoid-operated control valves. Both the shutoff and the control valves are enclosed in a single unit, but are electrically connected into the

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steering control system by individual electrical disconnect plugs. When energized, the solenoid shutoff valve allows hydraulic pressure to be admitted to the directional control valves. When de-energized, the shutoff valve is spring-loaded to the closed position. The directional control valves control the admittance of hydraulic pressure to a specific side of the nose wheel steering actuating cylinder piston. When de-energized, the directional control valves are spring-loaded to the closed position, porting the nose wheel steering actuating cylinder to the hydraulic system return through the directional control valve.

10. Main Left Steering Monitor Relay

The main left steering monitor relay (K712) is in the top left electronic equipment bay at fuselage station 70. (See figure 32-50-1). When the left rudder pedal is positioned to steer the airplane to the left, (left rudder pedal forward), the main command potentiometer wiper (R543) is repositioned, causing an electrically unbalanced condition between the main command and feed-back potentiometers (R542). This unbalanced condition causes a signal to go to the main nose gear steering control box, which provides an electrical ground for the left steer solenoid. This results in the nose wheels turning to the left. At the same time a ground is provided for a left steer, the main left steering monitor relay (K712) will also be energized, thus monitoring that an actual left steer signal is in progress. Function of the K712 relay is to provide part of an electrical circuit which will effect an auto transfer to stand-by in the event a malfunction occurrs on the main system.

11. Main Right Steering Monitor Relay

The main right steering monitor relay (K711) is in the top left electronic equipment bay at fuselage station 70. (See figure 32-50-1). When the right rudder pedal is positioned to steer the airplane to the right, (left rudder pedal forward), the main command potentiometer wiper (R543) is repositioned, causing an electrically unbalanced condition between the main command and feed-back potentiometers (R542). This unbalanced condition causes a signal to go to the main nose gear steering control box, which provides an electrical ground for the right steer solenoid. This results in the nose wheels turning to the right. At the same time a ground is provided for a right steer, the main right steering monitor relay (K711) will also be energized, thus monitoring that an actual right steer signal is in progress. Function of the K711 relay is to provide part of an electrical circuit which will effect an auto transfer to stand-by in the event a malfunction occurrs on the main system.

12. Stand-by Left Steering Monitor Relay

The stand-by left steering monitor relay (K708) is in the top left electronic equipment bay at fuselage station 70. (See figure 32-50-1). When the "NOSE GR STEER" selector switch is at "Stand-by" or the main gear steering system has malfunctioned, causing the electrical control to automatically change to the stand-by system, the stand-by left steering monitor relay will function as part of the stand-by steering "Lock-out" or disconnect circuit. When the left rudder pedal is positioned to steer the airplane to the left, (left rudder pedal forward) the stand-by command potentiometer wiper (R543) is repositioned, causing an unbalanced condition electrically between the stand-by command potentiometer and stand-by feed-back potentiometer (R542). This unbalanced con-

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dition causes a signal to go to the stand-by nose gear steering control box, which provides an electrical ground for the left steer solenoid. This results in the nose wheels turning to the left. At the same time a ground is provided for a left steer, the stand-by left steer monitor (K708) relay will be energized, thus monitoring that an actual left steer signal is in progress. Function of the K708 relay is to provide part of an electrical circuit which will effect an auto "Lock-out", or disconnect of the steering in the event a malfunction occurs in the stand-by system.

13. Stand-by Right Steering Monitor Relay

The stand-by right steering monitor relay (K710) is in the top left electronic equipment bay at fuselage station 70. (See figure 32-50-1). When the "NOSE GR STEER" selector switch is at "Stand-by" or the main gear steering system has malfunctioned, causing the electrical control to automatically change to the stand-by system, the stand-by right steering monitor relay will function as part of the stand-by steering "Lock-out" or disconnect circuit. When the right rudder pedal is positioned to steer the airplane to the right, (right rudder pedal forward) the stand-by command potentiometer wiper is repositioned, causing an unbalanced condition electrically between the stand-by command potentiometer and stand-by feed-back potentiometer. This unbalanced condition causes a signal to go to the stand-by nose gear steering control box, which provides an electrical ground for the right steer solenoid. This results in the nose wheels turning to the right. At the same time a ground is provided for a right steer, the stand-by right steer monitor (K710) relay will be energized, thus monitoring that an actual right steer signal is in progress. Function of the K710 relay is to provide part of an electrical circuit which will effect an auto "Lock-out", or disconnect of the steering in the event a malfunction occurs in the stand-by system.

14. Feed-back Potentiometer Monitor Switch

The feed-back potentiometer monitor switch is mounted on the right side of the nose gear strut, near the steering actuating cylinder. The feed-back potentiometer monitor switch is electrically connected in series with the command potentiometer monitor switch. (See figure 32-50-2 or 32-50-3). The feedback potentiometer monitor switch is actuated by the movement of the nose gear steering cylinder, and the commard potentiometer monitor switch is actuated by rudder pedal movement. If abnormal combinations of rudder pedal position, nose wheel position, and steering valve input signals occur, the switches will automatically cut off the main system and transfer steering to the stand-by system before a hard-over steering position of the nose wheel is reached.

15. Command Potentiometer Monitor Switch

The command potentiometer monitor switch is mounted on the right side of the aisle at fuselage station 130. (See figure 32-50-1). The command potentiometer switch is electrically connected in series with the feed-back potentiometer switch. (See figure 32-50-2 or 32-50-3). The command monitor switch is actuated by rudder pedal movement, and the feed-back potentiometer monitor switch is actuated by the steering actuating cylinder. If abnormal combinations of rudder pedal position, nose wheel position, and steering valve input signals occur, the switches will automatically cut off the main system and transfer steering to the stand-by system before a hard-over steering position of the nose wheel is reached.

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16. Nose Steering Transfer Relay

The nose steering transfer relay is at fuselage station 114. (See figure 32-50-1.) When a malfunction of the main nose gear steering system occurs, the nose steering transfer hold relay is energized. The nose steering transfer hold relay, through a set of its normally open contacts, energizes the nose steering transfer relay. When the nose steering transfer relay is energized, the control of the nose gear steering system is transferred electrically from the main system to the stand-by system. The nose steering transfer relay may also be energized by positioning the "NOSE GR STEER" selector switch at STANDBY.

17. Nose Steering Transfer Hold Relay

The nose steering transfer hold relay is in the LH top electronic equipment bay at fuselage station 70. (See figure 32-50-1.) When a malfunction of the main nose gear system occurs, the nose steering transfer hold relay is energized. Electrical power is routed across a normally open set of contacts of the nose steering transfer hold relay to energize the nose steering transfer relay. When the nose steering transfer relay is energized, the control of the nose gear steering system is transferred electrically from the main system to the stand-by system. The nose steering transfer hold relay has its own holding circuit which prevents the control of the steering system reverting back to the main system automatically once it has transferred to the stand-by system.

18. Stand-by Steering Lockout Relay

The stand-by steering lockout relay is at fuselage station 114. (See figure 32-50-1.) When the nose gear steering is electrically controlled by the stand-by system and a malfunction occurs permitting the stand-by left steering monitor and the stand-by right steering monitor relays to be energized simultaneously, the stand-by steering lockout hold relay is energized. With the stand-by steering lockout hold relay energized, an electrical circuit is completed to the stand-by steering lockout relay. The stand-by steering lockout relay will then remove the electrical power to the steering control valve, permitting the nose gear to go to a trail condition.

19. Stand-by Steering Lockout Hold Relay

The stand-by steering lockout hold relay is in the LH top electronic equipment bay at fuselage station 70. (See figure 32-50-1.) When the nose gear steering system is electrically controlled by the stand-by system and a malfunction occurs permitting the stand-by left steering monitor and the stand-by right steering monitor relays to be energized simultaneously, the stand-by steering lockout hold relay will be energized. A normally open set of its own contacts forms a holding circuit which will keep the stand-by steering lockout hold relay energized. The stand-by steering lockout relay is energized through a normally open set of contacts of the stand-by steering lockout hold relay.

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20. Steering Swivel Valve

The nose wheel steering swivel valve is mounted at the aft end of the nose wheel well, aft of the nose gear shock strut. The swivel valve allows the hydraulic lines from the nose steering actuating cylinder to pivot (swivel) when the nose gear shock strut is extended and retracted. The valve consists essentially of two fixed fittings which connect the steering control valve lines to the swivel valve and two swivel fittings that connect the steering actuating cylinder hydraulic lines to the swivel valve.

21. Nose Steer Ground Control Relay - Airplanes NA-380-1 and Later

The nose steer ground control relay (K1010) is in the relay box at fuselage station 150 right-hand aft. The relay is de-energized when the main landing gear struts are loaded (weight of airplane on gear). In this condition, when either of the nose gear steering buttons are depressed, the relay completes an electrical circuit to the nose gear steer control relay (K607). With the weight of the airplane off of the main landing gear, electrical power is then routed to the relay, thus the relay is energized. When energized, and when either of the nose gear steering buttons are depressed, the relay provides a circuit to the optional auto-pilot system.

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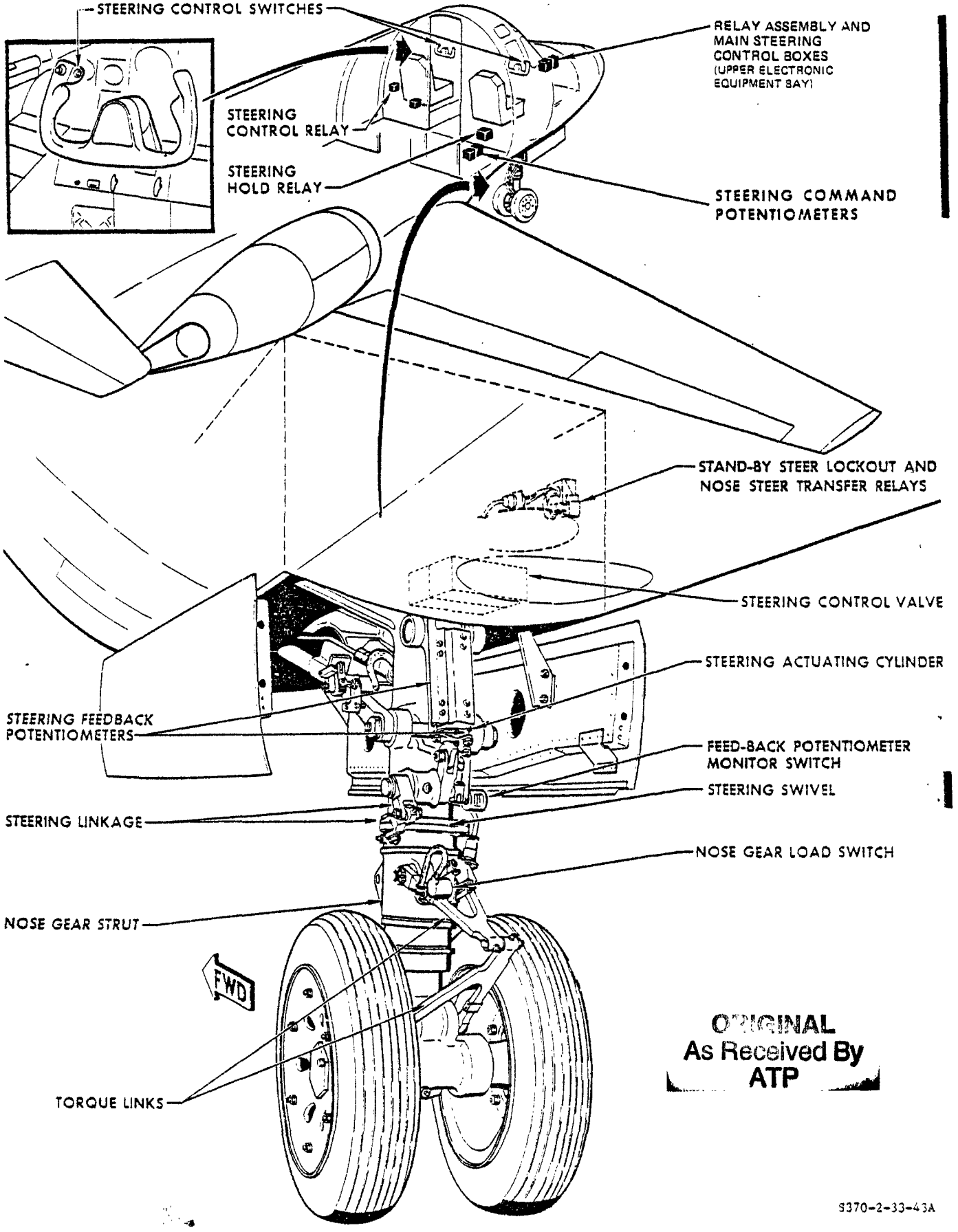

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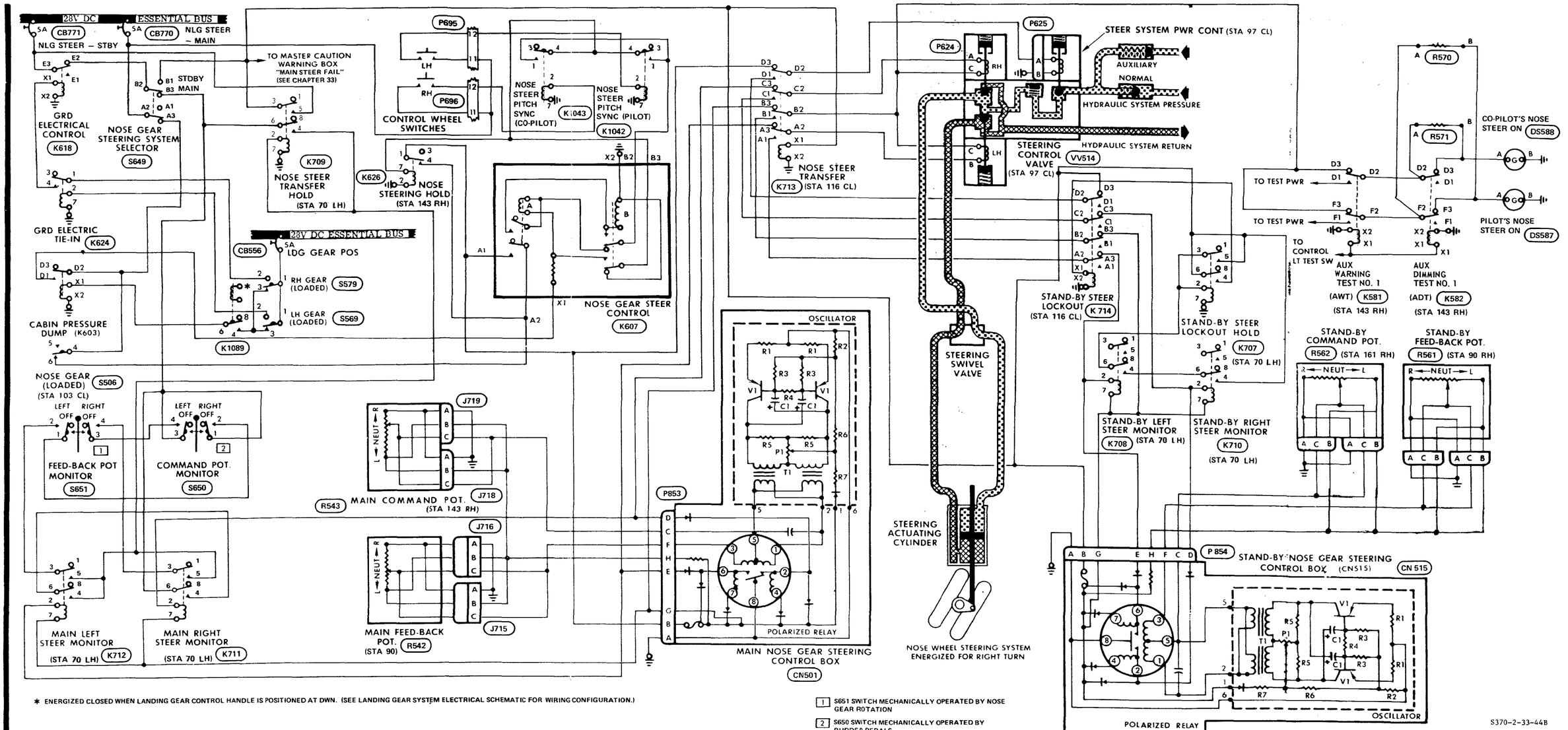
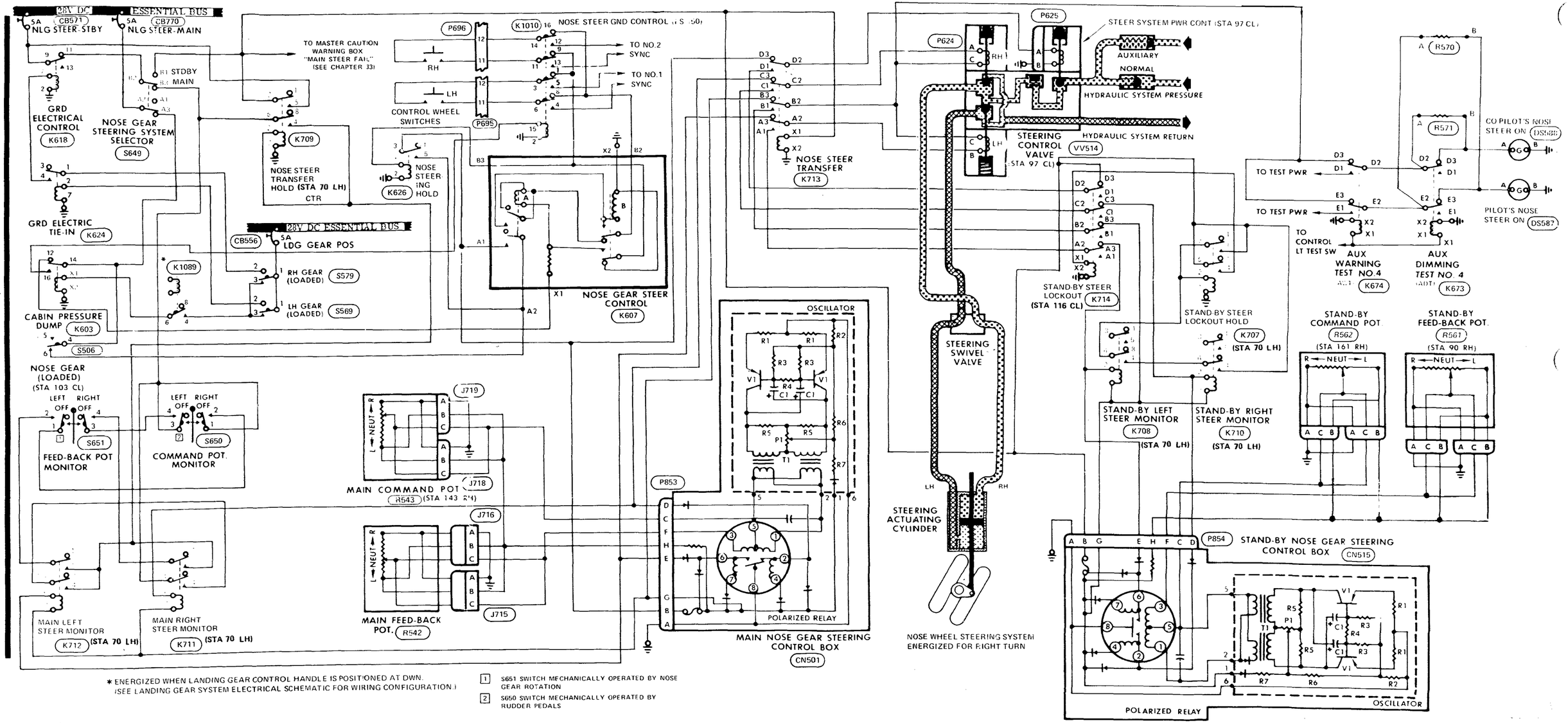
Nose Wheel Steering System
Figure 32-50-1



* ENERGIZED CLOSED WHEN LANDING GEAR CONTROL HANDLE IS POSITIONED AT DWN. (SEE LANDING GEAR SYSTEM ELECTRICAL SCHEMATIC FOR WIRING CONFIGURATION.)

[1] S651 SWITCH MECHANICALLY OPERATED BY NOSE GEAR ROTATION

[2] S650 SWITCH MECHANICALLY OPERATED BY RUDDER PEDALS



* ENERGIZED WHEN LANDING GEAR CONTROL HANDLE IS POSITIONED AT DWN. (SEE LANDING GEAR SYSTEM ELECTRICAL SCHEMATIC FOR WIRING CONFIGURATION.)

[1] S651 SWITCH MECHANICALLY OPERATED BY NOSE GEAR ROTATION

[2] S650 SWITCH MECHANICALLY OPERATED BY RUDDER PEDALS

Nose Wheel Steering Schematic - NA-380-1 and Later Airplanes
Figure 32-50-3

STEERING - TROUBLE SHOOTING

1. Trouble Shooting Nose Wheel Steering System

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. STEERING SYSTEM INOPERATIVE.		
(1) "NOSE GEAR - MAIN STEER.," "NOSE GEAR - STBY STEER.," and/or "LANDING GEAR - POS" circuit breakers out.	Check cockpit for correct position of circuit breakers.	Push in circuit breakers.
(1A) Defective relay K1089 or K1010*.	Check relay coil for open circuit.	Replace relay.
(2) Blown fuse in steering control box.	Check fuse in steering control box.	Replace fuse and check circuit for defective wiring. (Refer to Wiring Diagram Manual.)
(3) Defective steering control valve.	Disconnect steering control valve electrical disconnect plug to control valve shutoff valve. Apply electrical power to airplane and hold steering switch depressed. Check for voltage at pin "A" of electrical plug. If voltage is available, the circuit is complete to control valve. Check continuity between pin "B" of electrical plug and ground (structure). If continuity is available, the circuit is complete to and from the electrical plug. Therefore, the steering control valve is defective.	Replace steering control valve.
B. STEERING SYSTEM DOES NOT ENGAGE WHEN STEERING SWITCH IS DEPRESSED AND NOSE GEAR IS COMPRESSED.		
(1) Nose gear load switch improperly adjusted or defective.	Disconnect electrical plug at right side of nose gear strut. Check continuity between pins "6" and "4" of electrical plug. If there is no continuity, disconnect link at switch arm. Rotate switch arm.	Adjust or replace switch.

*NA-380-1 and Later Airplanes

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
	If there is continuity when switch arm is rotated, the nose gear load switch is improperly adjusted. If there is no continuity, the switch is faulty.	
(2) Defective steering control box.	Disconnect steering control box electrical plug. Check for voltage at pin "B" of electrical plug receptacle. If there is voltage at pin "B," check continuity between pins "B" and "H" of plug receptacle. If there is continuity, check continuity between pins "C" and "F." If there is no continuity, the control box is defective. If there is continuity, connect electrical plug and apply electrical power; then depress steering switch. Check by sound or feel the rhythmic actuation of the steering control valve control solenoids. If there is no actuation of the solenoids, the oscillator in the steering control box is not operating.	Replace steering control box.
C. AIRPLANE TENDS TO STEER TO ONE SIDE.		
(1) Feed-back potentiometer not electrically balanced with command potentiometer.	Position rudder in neutral and install rig pin through bell crank. Engage nose wheel steering system. Check position of zero mark on steering protractor with index mark on strut. If zero mark and index mark do not align, the feed-back potentiometer must be electrically balanced with the command potentiometer.	Adjust feed-back potentiometer.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
D. NOSE WHEEL CANNOT BE TURNED FAR ENOUGH ONE WAY.		
(1) Incorrect rudder travel.	Check rudder travel. If rudder travel is short, steering throws will be short.	Rig rudder control system.
E. AIRPLANE TURNS IN ONE DIRECTION ONLY.		
(1) Defective steering control valve.	Disconnect electrical plug to steering control solenoids. Check continuity between pins "B" and "C," and then between "A" and "C" of same electrical plug receptacle. If there is no continuity in either case, the control valve is defective.	Replace steering control valve.
F. NOSE WHEELS GO TO FULL STEERING THROW WHEN RUDDER IS MOVED OUT OF NEUTRAL POSITION.		
(1) Feed-back potentiometer inoperative.	Disconnect feed-back potentiometer electrical plug. Shut down hydraulic test stand. Disconnect steering actuating cylinder rod end. Connect an ohmmeter to pins "A" and "C" of electrical plug receptacle. Push nose gear piston in and out. The resistance in ohms should change when the piston is moved. If the resistance does not change, the potentiometer is defective.	Replace feed-back potentiometer.
G. WITH NOSE GEAR STEERING SYSTEM IN STAND-BY AND RIGHT-HAND RUDDER PEDAL PUSHED TO ITS FORWARD POSITION, NOSE GEAR STEERING SYSTEM GOES TO TRAIL POSITION.		
(1) Continuity between pins "A" and "E" of nose gear steering stand-by control box receptacle.	Remove electrical power from airplane and check for continuity between pins "A" and "E" of nose gear steering stand-by control box receptacle.	Replace or repair nose gear steering stand-by control box.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
(2) Electrical wiring in stand-by left steering monitor relay circuit is grounded.	Remove electrical power from airplane and check circuit for a grounded condition.	Repair as necessary. (Refer to Wiring Diagram Manual.)
H. NOSE STEERING SYSTEM AUTOMATICALLY CHANGES FROM MAIN NOSE GEAR STEERING SYSTEM TO STAND-BY NOSE GEAR STEERING SYSTEM WHEN RIGHT-HAND RUDDER PEDAL IS PUSHED TO ITS FORWARD POSITION.		
(1) Command potentiometer monitor switch out of adjustment or inoperative.	Check switch for proper operation and adjustment.	Adjust or replace command potentiometer monitor switch.
(2) Feed-back potentiometer monitor switch out of adjustment or inoperative.	Check switch for proper operation and adjustment.	Adjust or replace feed-back potentiometer monitor switch.
(3) Continuity between pins "E" and "A" of nose gear steering main control box receptacle.	Remove electrical power from airplane and check for continuity between pins "E" and "A" of nose gear steering main control box receptacle.	Replace or repair nose gear steering main control box.
(4) Electrical wiring in main left steering monitor relay circuit is grounded.	Remove electrical power from airplane and check circuit for a grounded condition.	Repair as necessary. (Refer to Wiring Diagram Manual.)
I. NOSE STEERING SYSTEM AUTOMATICALLY CHANGES FROM MAIN NOSE GEAR STEERING SYSTEM TO STAND-BY NOSE GEAR STEERING SYSTEM WHEN LEFT RUDDER PEDAL IS PUSHED TO ITS FORWARD POSITION.		
(1) Command potentiometer monitor switch out of adjustment or inoperative.	Refer to probable cause (1) under trouble H.	
(2) Feed-back potentiometer monitor switch out of adjustment or inoperative.	Refer to probable cause (2) under trouble H.	
(3) Continuity between pins "D" and "A" of nose gear steering main control box receptacle.	Remove electrical power from airplane and check for continuity between pins "D" and "A" of nose gear steering main control box receptacle.	Replace or repair nose gear steering main control box.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
(4) Electrical wiring in main right-hand steering monitor relay circuit is grounded.	Remove electrical power from airplane and check circuit for a grounded condition.	Repair as necessary. (Refer to Wiring Diagram Manual.)

STEERING - MAINTENANCE PRACTICES

1. General

The hydraulic system furnishes the fluid supply for the steering system. To bleed steering system of air, operate system to full throw several times. For lubrication requirements, refer to Chapter 12.

Adjustment of the steering system is confined to an adjustment of the command and feed-back potentiometers, nose wheel steering actuating cylinder, and rudder travel stops. If no malfunction of system units within the steering system has occurred, the only adjustments necessary should be electrically balancing the feed-back potentiometer to match the command potentiometer, and/or adjusting the steering actuating cylinder and rudder travel stops to obtain full steering travel. If the rudder travel stops are adjusted, the rudder travel must remain within the specified tolerances.

2. Replacement of Nose Gear Load Switch

A. Removing Nose Gear Load Switch

(1) Disconnect switch electrical plug (quick-disconnect) at side of strut.

(2) Remove electrical clamps, switch link, jam nut, and bracket attach screws.

(3) Remove switch and switch bracket as a unit.

B. Installing and Adjusting Nose Gear Load Switch

See figure 32-50-401.

3. Replacement of Steering Actuating Cylinder

A. Removing Steering Actuating Cylinder

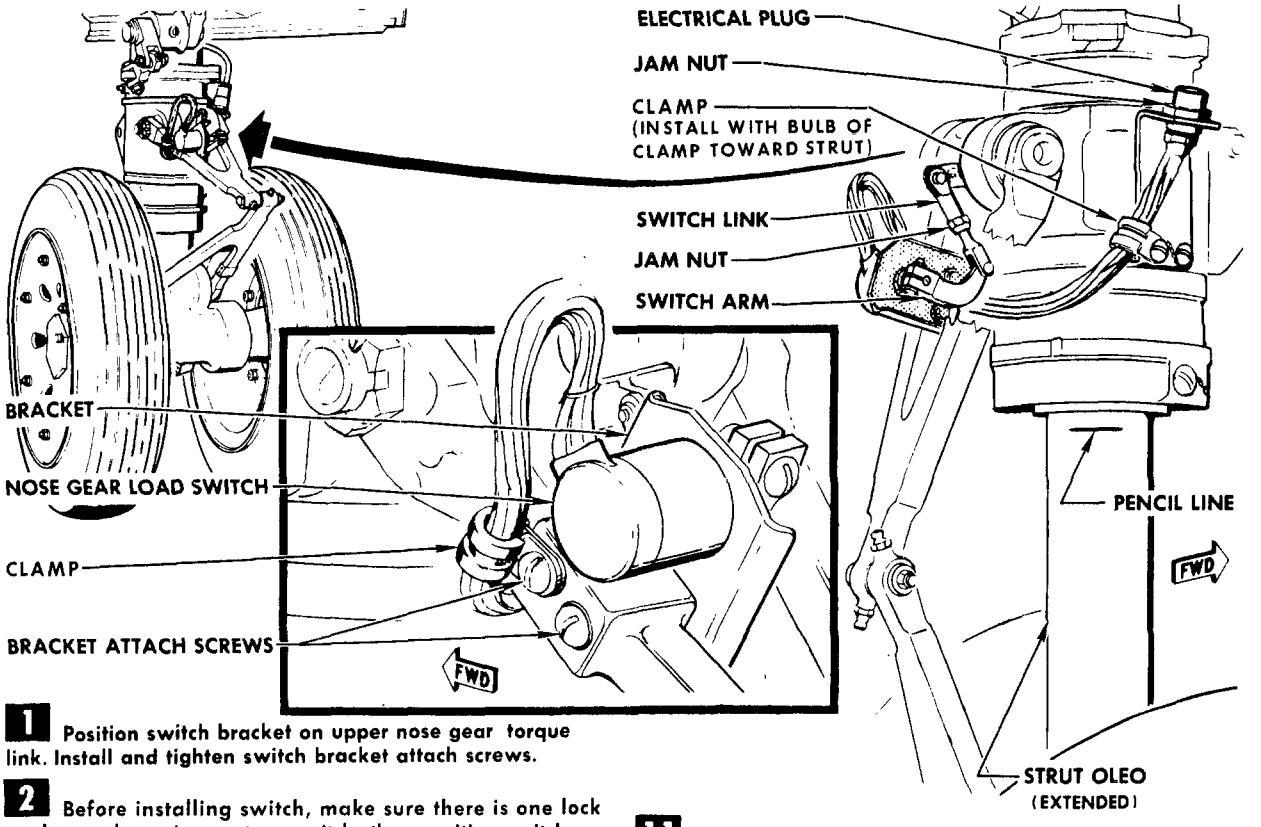
(1) Disconnect and cap hydraulic lines.

(2) Remove lockwire from feed-back potentiometers, attaching screws, and jam nuts. Count and record number of threads on potentiometer rod extending above top jam nut. Remove lower jam nuts and attaching screws, and remove potentiometers.

(3) Disconnect actuating cylinder rod end and bracket attach bolts from either side of actuating cylinder. Remove bracket; then remove actuating cylinder.

B. Installing and Adjusting Steering Actuating Cylinder

See figure 32-50-402.



1 Position switch bracket on upper nose gear torque link. Install and tighten switch bracket attach screws.

2 Before installing switch, make sure there is one lock washer and one jam nut on switch; then position switch on switch mounting bracket with leads up. Check that index pins are seated in index holes in bracket and tighten and safety jam nut with lockwire.

3 Insert switch electrical plug through hole in bracket at right side of strut. Install and tighten electrical plug jam nut.

4 Install clamp that secures electrical leads to strut. (To prevent electrical leads from striking nose wheel, position clamp with bulb facing strut.)

5 Place jack under airplane forward fuselage. Raise airplane until nose gear strut is fully extended.

6 Draw a pencil line on strut oleo 1-3/4 inches below bottom of strut gland nut.

7 Place jack under strut and compress strut to pencil line on strut oleo.

8 Connect a continuity light to pins D and F of electrical plug at right side of strut.

9 Adjust length of switch link to 2-5/16 inches, eye to eye. Place switch arm on switch shaft with crook on arm forward and up. Do not tighten locking bolt. Attach switch link to arm.

10 Looking at shaft end of switch, rotate shaft within switch arm clockwise until continuity light just comes on; then tighten locking bolt, securing switch arm to switch shaft.

11 Slowly extend strut. Again compress strut. Continuity light should come on within ±1/4 inch of pencil line.

12 Disconnect continuity light and connect light to pins A and C of electrical plug. Repeat step 11

13 Tighten switch link jam nut and safety switch link clevis pin.

14 Connect electrical plug (quick-disconnect) at side of strut.

Caution When the quick-disconnect is connected correctly, an orange band will show at the center. If the quick-disconnect is not connected correctly, loss of steering can result.

15 Remove jack under strut.

16 Lower airplane and remove jack under forward fuselage.

17 Check operation of landing and taxi lights.

18 Perform operational check-out of nose wheel steering system.

19 Disconnect external dc electrical power. Shut down and disconnect hydraulic test stand. Install all access doors.

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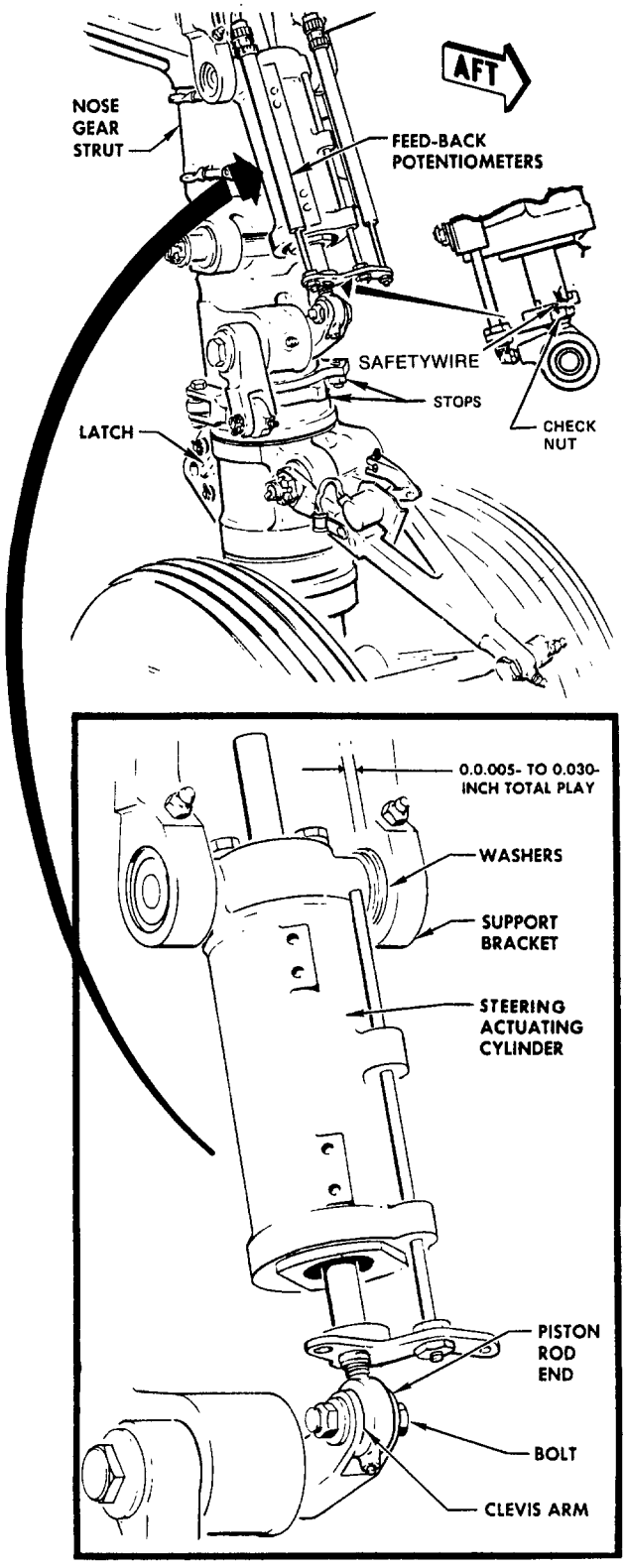
Installing and Adjusting Nose Gear Load Switch
Figure 32-50-401

1. Position nose wheel steering actuating cylinder in tip mounting bracket. Install 2W1AL-72-44-25 washers between actuating cylinder and support bracket as necessary to center actuating cylinder rod end in clevis arm and to maintain a total end play of 0.005 to 0.030 inch at top of actuating cylinder.
2. Install and tighten support bracket attach bolts. Torque bolts 70-90 inch lbs.
3. Disengage latch at forward side of shock strut.
4. Rotate nose wheels to full left turn position (stops on strut in contact). Remove safetywire and loosen rod end check nut.
5. Fully retract actuating cylinder piston until piston is bottomed. Adjust piston rod end to obtain a "slip fit" of attach bolt when piston is aligned with clevis arm.
6. Shorten rod end three turns; then temporarily connect rod end to clevis arm.
7. Manually rotate steering collar to full right turn position (stops on strut in contact).
8. Remove rod end attach bolt and disengage steering cylinder from clevis arm; check for 0.08 inch minimum overstroke by measuring stroke required to fully bottom actuating cylinder in extended position.
9. Install and tighten rod end attach bolt to remove all end play between bolt head and nut. Do not apply load to bellcrank clevis. Lubetork [1] check nut to 150 ± 5 inch lbs. and secure with 0.032 in. safetywire.
10. Uncap, connect, and tighten hydraulic lines.
11. Install feed-back potentiometer to actuating cylinder.
12. Lubricate actuating cylinder rod end.

CAUTION

If the airplane system hydraulic pump is to be used to perform this checkout procedure, restrict hydraulic pump operations to a minimum to prevent pump motor over heating.

13. Connect external 28 VDC electrical power. Connect hydraulic test stand (if available) to airplane hydraulic system ground test panel. Regulate test stand to 4 g.p.m. minimum at 3000 psi. If test stand is not available, use aircraft system hydraulic pump.
14. Check operation of nose wheel steering system. Check actuating cylinder for evidence of hydraulic leaks.
15. Remove 28 VDC external electrical power. Shut down hydraulic test stand. Check hydraulic system reservoir fluid level. Service reservoir if needed. Disconnect hydraulic test stand.
16. Engage latch at forward side of strut and install all access doors.



[1] Lubricate threads with anti-seize (MIL-T-5544) or equivalent.

Figure 32-50-02 — Installing and Adjusting Acutating Cylinder

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MAINTENANCE PRACTICES (Continued)

REPLACEMNT OF STEERING FEED-BACK POTENTIOMETER

Removing Steering Feed-back Potentiometer

NOTE

There are two feed-back potentiometers: main and stand-by. All parts in the following procedure are identified in Figure 32-50-403.

1. Disconnect four electrical plugs if both potentiometers are being removed.
2. If main feed-back potentiometer is being removed, proceed as follows:
   A. Break safetywire from lower end of potentiometer. Count and record number of threads visible above upper jam nut. This will aid when replacement potentiometer is installed. Remove lower jam nut.
   B. Break safetywire from screws attaching potentiometer to nose wheel steering actuating cylinder. Note position of potentiometer on actuating cylinder so that replacement potentiometer may be installed in same position. Remove attach screws and remove potentiometer.
3. If stand-by feed-back potentiometer is being removed, proceed as follows:
   A. Break safetywire from lower end of potentiometer. Count and record number of threads visible above upper jam nut. This will aid when replacement potentiometer is installed. Remove lower jam nut.

NOTE

There is a bushing installed on the end of the potentiometer shaft in the yoke assembly which may fall out when the lower jam nut is removed.

   B. Break safetywire from screws attaching potentiometer to steering actuating cylinder. Note position of potentiometer on actuating cylinder so that replacement may be installed in the same position. Remove attach screws and potentiometer.

Installing and Adjusting Steering Feed-back Potentiometer

See Figure 32-50-403.

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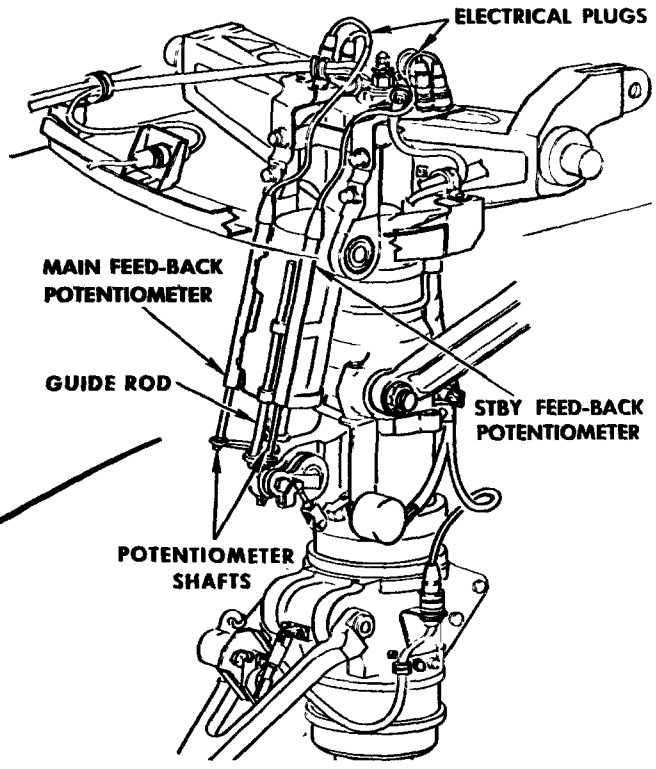
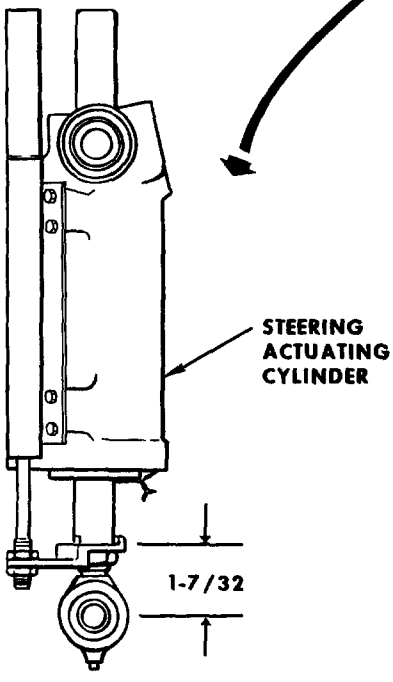
1 Measure distance from center of rod end to bottom of actuating cylinder piston rod. If distance is 1-7/32 inches or less, no adjustment is necessary. If distance exceeds 1-7/32 inches, loosen yoke locknut and adjust rod end to obtain proper distance. Shorten rod end as necessary to properly position yoke when installed. Tighten yoke locknut to 145-155 inch-pounds and safety locknut.

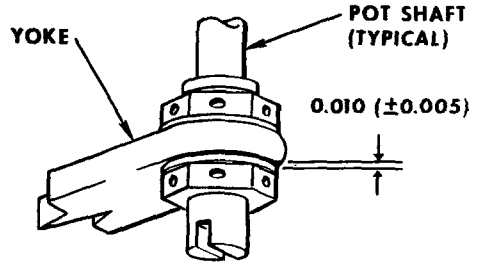




2 Install feed-back pots as follows:

a. Install pots as noted during removal. Tighten bolts enough to hold pots securely and still allow position adjustment.

b. Install both pot shafts to yoke. Install bushings and washers on pot shafts, ensuring that a clearance of 0.010 (±0.005) inch exists between lower face of yoke and lower washers. If clearance is excessive, trim bushings.



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Installing and Adjusting Feed-back Potentiometer
Figure 32-50-403 (Sheet 1 of 2)

c. Slowly extend and retract actuating cylinder. Position main feed-back pot for proper alignment.

Caution The pot shaft bushings must have clearance in the yoke through the full range of actuator travel. Do not apply any bending or side forces to the pot shafts. Misalignment of pot shafts could result in damage to the pot.

d. Tighten feed-back pot attach screws and safety with lockwire. Make sure that alignment of pot is maintained when tightening attach screws.

e. Disengage latch on forward side of shock strut.

f. Rotate steering column to obtain a slip fit of attach bolt when steering actuator rod end is connected to clevis fitting. Tighten attach bolt.

3 Connect potentiometer electrical plugs and safety plugs with lockwire.

4 Position rudder pedals to neutral position. The gust lock may be used to lock the rudder in neutral.

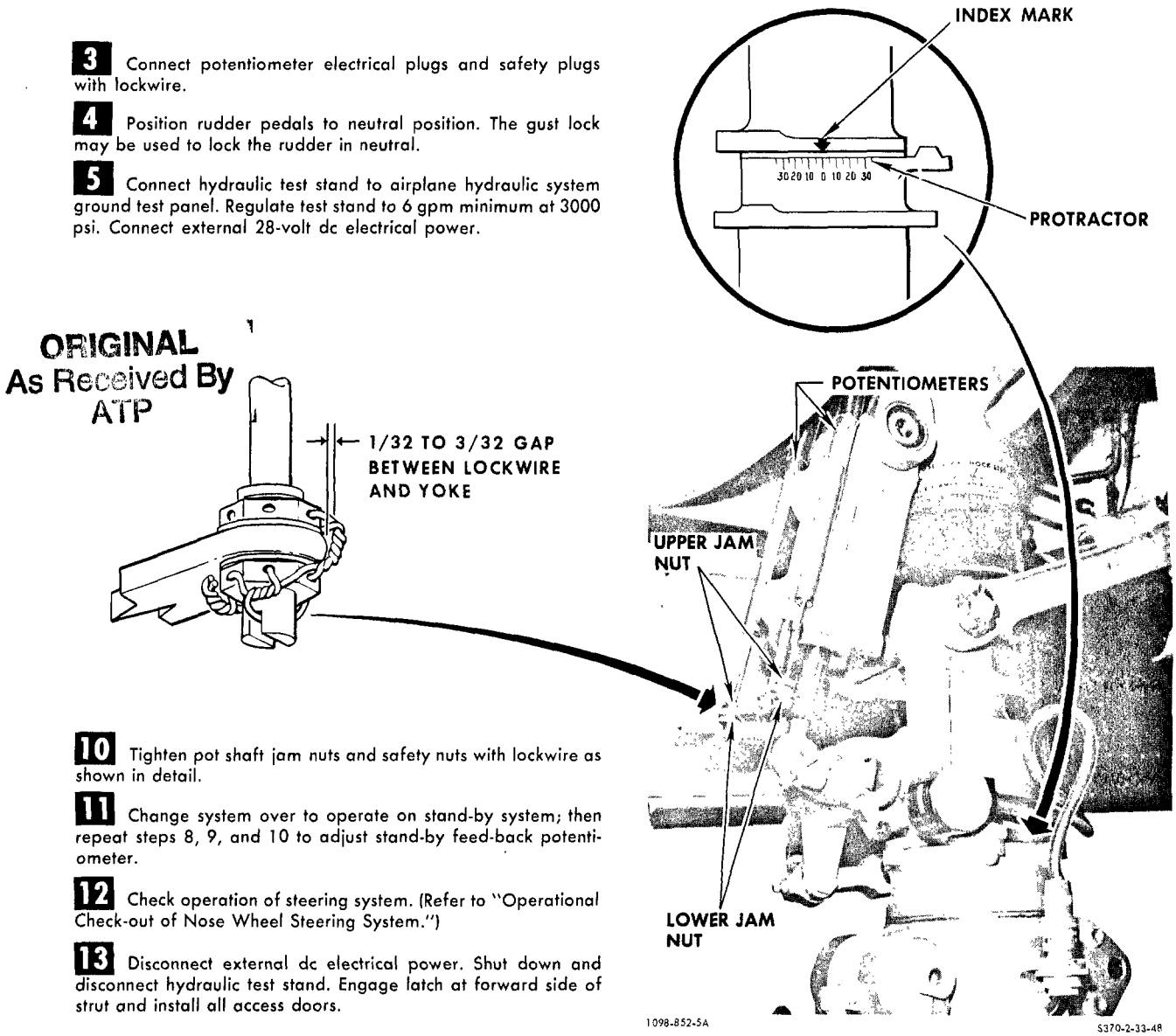
5 Connect hydraulic test stand to airplane hydraulic system ground test panel. Regulate test stand to 6 gpm minimum at 3000 psi. Connect external 28-volt dc electrical power.

6 Push in "NOSE GEAR - STBY STEER.", "NOSE GEAR-MAIN STEER.", and "LGD GEAR POS" circuit breakers.

7 Disengage nose gear steering lock lever at forward side of nose gear strut. This may be done by connecting a tow bar to the strut.

8 Depress and hold steering control switch.

9 Check protractor on forward side of nose gear strut. If zero mark on protractor does not align with index mark on strut, loosen feed-back potentiometer rod jam nuts and adjust length of rod to position zero mark on protractor in alignment with index mark on strut. Adjustment of the potentiometer rod electrically balances the feed-back potentiometer with the command potentiometer.



10 Tighten pot shaft jam nuts and safety nuts with lockwire as shown in detail.

11 Change system over to operate on stand-by system; then repeat steps 8, 9, and 10 to adjust stand-by feed-back potentiometer.

12 Check operation of steering system. (Refer to "Operational Check-out of Nose Wheel Steering System.")

13 Disconnect external dc electrical power. Shut down and disconnect hydraulic test stand. Engage latch at forward side of strut and install all access doors.

Installing and Adjusting Feed-back Potentiometer
Figure 32-50-403 (Sheet 2 of 2)

MAINTENANCE PRACTICES (Continued)

REPLACEMENT OF STEERING COMMAND POTENTIOMETER

Removing Steering Command Potentiometer

There are two nose wheel command potentiometers. Removal procedure for either potentiometer is the same. The command potentiometers are installed on the right side of the aisle at station 146, aft of the cockpit. All parts in the following procedure are identified in Figure 32-50-405.

1. Remove necessary access panels.
2. Pull NOSE GEAR — MAIN STEER and NOSE GEAR — STBY STEER circuit breakers.
3. Disconnect potentiometer electrical plug, and push rod at potentiometer arm.
4. Remove attach and remove potentiometer.

Installing and Adjusting Steering Command Potentiometer

See Figure 32-50-404.

REPLACEMENT OF FEED-BACK POTENTIOMETER MONITOR SWITCH

Removing Feed-back Potentiometer Monitor Switch

1. Disconnect electrical leads and electrical clamps.
2. Remove steering indicator arm clevis pin and bracket attach nut.
3. Remove feed-back potentiometer monitor switch and bracket.

Installing and Adjusting Feed-back Potentiometer Monitor Switch

See Figure 32-50-405.

REPLACEMENT OF STEERING CONTROL VALVE FILTER ELEMENT

NOTE

The Menasco (P/N 1176L001) and Sterer (P/N 57640) nose steering control valve are interchangeable. Before ordering filter replacement parts, make sure what valve is on airplane. Refer to Sabreliner Service Letter No. 30.

The steering control valve filter is on the side of the control valve assembly, next to the pitot valve electrical connector. (See Figure 32-50-1).

Removing Filter Element in Menasco Nose Steering Control Valve (P/N 1176L001)

CAUTION

Too rapid dumping of the accumulators may exceed the reservoir pressure relief valve rate and cause possible damage.

1. In access doors (see Figure 32-50-404A), hold normal and auxiliary accumulator dump valves open until all hydraulic pressure is discharged.
2. Open access panel (see Figure 32-50-404A) to gain access to control valve.
3. Cut lockwire and unscrew filter case.
4. Remove and discard filter element, O-ring and retainer.

Installing Filter Element in Menasco Nose Steering Control Valve (P/N 1176L001)

1. Install new filter element (P/N AN6235-1A), O-ring (P/N 6227B19) and retainer (P/N MS28782-19). Clean filter case with dry-cleaning solvent (P-D-680 or equivalent).

NOTE

Retainer and O-ring must be lubricated with hydraulic fluid (MIL-H-5606()) before assembly.

2. Install filter element and case. Screw filter case down tight and safety to valve body with lockwire.
3. Clean area of spilled hydraulic fluid.
4. Check fluid level in hydraulic fluid reservoir and refill if necessary.
5. Connect hydraulic test stand to hydraulic test panel.
6. Operate nose wheel steering system.
7. Check filter for hydraulic leaks.
8. After leak check, position all switches to off position or aircraft parked position.
9. Disconnect external electrical power and hydraulic test stand.
10. Install all access doors.

Removing Filter Element in Sterer Nose Steering Control Valve (P/N 5764C)

1. Open access panel and access doors (see Figure 32-50-404A).
2. Set PUMP switch to OFF. Open HYD PUMP CONT circuit breaker.

CAUTION

Too rapid dumping of the accumulators may exceed the reservoir pressure relief valve rate and cause possible damage.

3. Discharge normal and auxiliary hydraulic accumulator pressure and reservoir air pressure.
4. Remove lockwire and cap from steering control valve.
5. Remove spring, filter element, packings (O-rings) and retainers from filter cavity.

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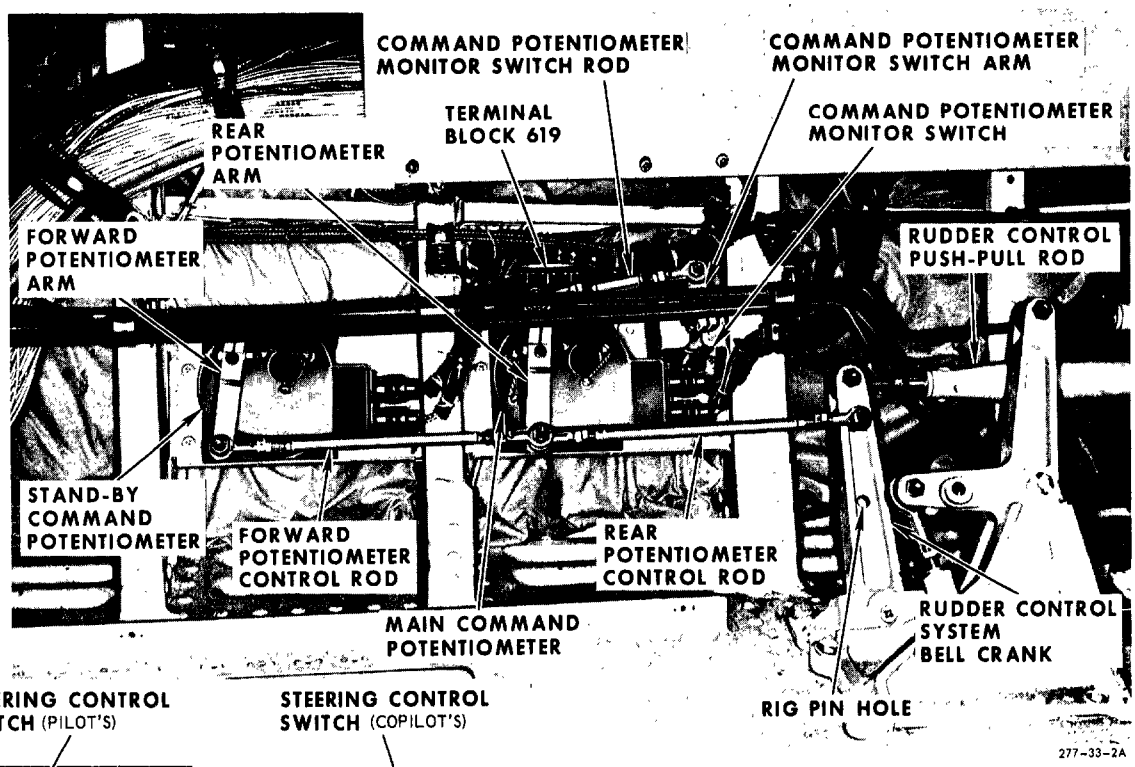
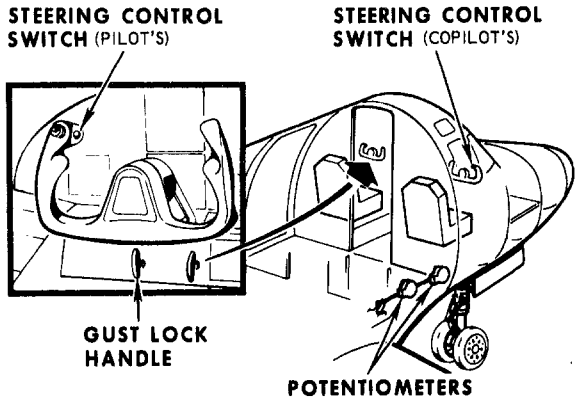
MAINTENANCE PRACTICES (Continued)





NOTE

Two command potentiometers are used. The installation procedures are the same for both potentiometers.

1. Pull the NOSE GEAR-MAIN STEER, NOSE GEAR-STBY STEER, and LANDING GEAR-POS circuit breakers.
2. Remove upholstered panels on lower right side of aisle at FS 146 to 176.5. Install potentiometers on mounting bracket in airplane with receptacles facing aft.

NOTE

When installing potentiometers, make sure the two potentiometer part numbers indicate a matching set.

3. With flat spot of potentiometer shaft facing up, install arm on potentiometers so that arm hangs downward at a 6 o'clock position.
4. Check resistance in ohms between pins "B" and "C" on each potentiometer electrical plug receptacle. The resistance should be 267 (±6) ohms. If resistance is not obtainable with the arm at 6 o'clock, loosen the three adjusting screws and adjust potentiometer. After adjustment, tighten the three adjusting screws and safety with lockwire.
5. Connect potentiometer electrical plugs, and safety with lockwire. (Two plugs connect to each potentiometer.)
6. Position rudder pedals in neutral position and engage gust lock. Install a 5/16-inch diameter rig pin through aligned holes in rudder control system bell crank and bracket aft of potentiometers.
7. Secure one end of rear potentiometer control rod and one end of forward potentiometer control rod to lower end of rear potentiometer arm with attach bolt.

NOTE

At the rear potentiometer arm, the rear potentiometer control rod end is mounted under the forward potentiometer control rod end.

Figure 32-50-404 — Installing and Adjusting Steering Command Potentiometer (Sheet 1 of 3)

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MAINTENANCE PRACTICES (Continued)

8. Release jam nuts on both ends of rear potentiometer control rod. Position rear potentiometer arm parallel to forward face of rear potentiometer. Adjust length of control rod so that when rod is attached to rudder bellcrank, a slip fit is maintained on attach bolt. Secure rear potentiometer control rod to bellcrank with attach bolt. Tighten jam nuts.
9. Release jam nuts on both ends of forward potentiometer control rod. Position forward potentiometer arm parallel to forward face of forward potentiometer. Adjust length of control rod so that when rod is attached to potentiometer arm, a slip fit is maintained on attach bolt. Secure forward potentiometer control rod to forward potentiometer arm with attach bolt. Tighten jam nuts.
10. Install command potentiometer monitor switch on its mounting bracket, making sure dowel pins engage holes. Secure with attach nut, and safety nut to bracket with lockwire. Install switch mounting bracket aft of rear (main) potentiometer.

NOTE

Command potentiometer monitor switches must be installed with wires up. Connect terminals 1 and 3 from monitor switch to terminal 3 of Terminal Board TB 619. Connect terminal 2 from monitor switch to terminal 4 of Terminal Board TB619. Connect terminal 4 from monitor switch to terminal 2 of Terminal Board TB619.

11. Rotate monitor shaft to its mid-point position (half total travel). Connect a continuity light to switch wires 1 and 2, or terminals 3 and 4 of terminal board TB619 located directly above rear command potentiometer. There must be no electrical continuity between test points (continuity light must not come on) with the switch shaft at mid-point position. Slowly rotate switch shaft counterclockwise until continuity light comes on. At this point, mark alignment of shaft slot for reference.
12. Disconnect continuity light from switch wires 1 and 2 (or terminals 3 and 4 of TB619). Connect continuity light to wires 3 and 4 of monitor switch, or to terminals 2 and 3 of TB619. There must be no electrical continuity between test points. Slowly rotate switch shaft clockwise until continuity is obtained. At this point, mark alignment of shaft slot for reference.
13. Rotate monitor switch shaft counterclockwise to its mid-point position. Check that shaft mid-point position lies equidistant between reference marks made in steps 11 and 12.
14. Install monitor switch arm on switch shaft.

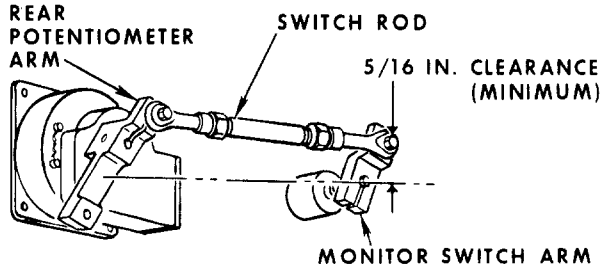
NOTE

It is essential that the switch arm be installed in the 11 o'clock position when switch shaft is held at mid-point position.

15. Connect command potentiometer monitor switch rod to switch arm at 11 o'clock position and to upper end of rear potentiometer arm. Adjust monitor switch rod so that a slip fit is maintained on attach bolts.
16. Remove rig pin from rudder bellcrank and disengage gust lock.
17. Remove bolt that attaches rudder control push-pull rod assembly to upper end of rudder bellcrank. Secure rudder control rod out of the way of bellcrank travel.

NOTE

With the rudder control rod detached from the rudder bellcrank, the rudder pedals will contact the pedal stops without requiring much force. Otherwise, 180 pounds of force would be required.



18. Slowly advance right rudder pedal and check that monitor switch arm does not travel over center. Push right rudder pedal against its stop. With the right rudder pedal against its stop, there must be a clearance of not less than 5/16 inch between the center of the bolt attaching the monitor switch arm to the switch rod and a line passing through the center of the switch shaft and the center of the rear potentiometer arm.
19. If less than a 5/16 inch clearance is observed in step 18, return rudder pedal to neutral position. Install rig pin as instructed in step 6. Detach monitor switch rod from switch arm. Remove monitor switch arm and reinstall at enough degrees under 11 o'clock position to obtain required clearance. Adjust switch rod for slip fit and repeat step 18 after attaching switch rod to switch arm and removing rig pin.

Figure 32-50-404 — Installing and Adjusting Steering Command Potentiometer (Sheet 2 of 3)

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MAINTENANCE PRACTICES (Continued)

20. Repeat procedure in step 18 for full left rudder pedal against its stop. If necessary, correct the required clearance for switch rod and arm as instructed in step 19, except reinstall switch arm at enough degrees over 11 o'clock position for clearance correction.

NOTE

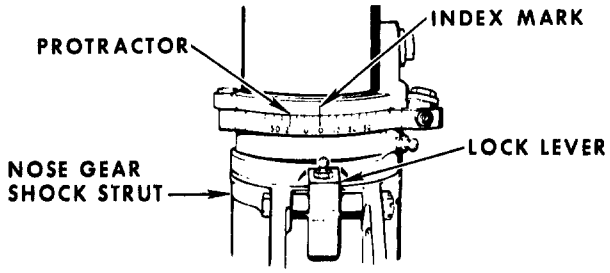
If the 5/16 inch clearance cannot be obtained in both directions of the monitor switch arm travel, the switch arm position, switch shaft mid-travel point, and switch rod length must be simultaneously adjusted for equal clearance in each direction.

21. Install rig pin in rudder bellcrank as instructed in step 6.
22. Connect a tow bar to nose gear shock strut and disengage lock lever on forward side of strut. If other than the Sabreliner Corporation designed tow bar is used, check that lock lever is disconnected from nose gear strut collar.
23. Apply 28 VDC power to airplane. Apply 2,600 to 3,000 psi hydraulic pressure to hydraulic power system.

CAUTION

If airplane hydraulic system pump is used to provide hydraulic power, restrict pump operation to a minimum to prevent overheating.

24. Engage NLG STEER—MAIN, NLG STEER—STBY, and LDG GEAR—POS circuit breakers.



25. Position NOSE GR STEER selector switch to MAIN. Depress and release steering control switch. Check protractor on forward side of nose gear shock strut. If protractor zero mark does not align with index mark, adjust main feed-back potentiometer to align zero mark with index mark (see Figure 32-50-403).

NOTE

Adjustment of the feed-back potentiometer rod electrically balances the feed-back potentiometer with the command potentiometer.

26. Position NOSE GR STEER selector switch to STANDBY. Check that protractor zero mark aligns with index mark. If alignment is not observed, adjust stand-by feed-back potentiometer to align zero mark with index mark.
27. Install bolt attaching rudder control push-pull rod assembly to upper end of rudder bellcrank. Remove rig pin from rudder bellcrank.
28. Remove electrical and hydraulic power.
29. Adjust feed-back potentiometer monitor switch as required. (See Figure 32-50-405.)
30. Perform operational check-out of nose wheel steering system.

Figure 32-50-404 — Installing and Adjusting Steering Command Potentiometer (Sheet 3 of 3)

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MAINTENANCE PRACTICES (Continued)

Installing Filter Element in Sterer Nose Steering Control Valve P/N 57640

1. Thoroughly clean cap, spring, and filter element with cleaning solvent (P-D-680 or equivalent). Be sure all clean parts are thoroughly dry before assembly.

NOTE

Packings (O-rings) and retainers must be lubricated with hydraulic fluid MIL-H-5606 before assembly.

2. Install new O-ring P/N MS28774-011 on filter element.
3. Install filter element in filter cavity.
4. Install new retainer P/N MS28774-019, O-ring P/N MS28775-019 and retainer P/N MS28774-019 on end of cap.
5. Install spring and then tighten cap until it bottoms out in valve housing. Safetywire cap to housing.
6. Clean area of spilled hydraulic fluid.
7. Service hydraulic system.
8. Pressurize system and operate nose steering system. Check filter bowl for leaks.
9. Install access panel and close and secure access doors.

REPLACEMENT OF STAND-BY LOCKOUT OR NOSE STEER TRANSFER RELAYS

Removing stand-by lockout (K714) and Nose Steer Transfer (K713) Relays (See Figure 32-50-1).

1. Remove electrical power from airplane.
2. Remove round access cover.
3. Disconnect electrical plug.
4. Remove relay mount support assembly, by removing two flush head screws from the lower fuselage skin, located just aft of the outer periphery of the access door.
5. Remove relay mount (four bolts) from its support assembly and remove from airplane.

Installing stand-by lockout (K714) and Nose Steer Transfer (K713) Relays (See Figure 32-50-1).

1. Remove electrical power from airplane.
2. After relay has been replaced in its mount, check electrical wiring hook-up. Refer to Wiring Diagram Manual.
3. Install relay mount on support assembly (four bolts) and install support assembly to fuselage skin (two flush head screws).
4. Install electrical plug and lockwire.
5. Install round access cover.
6. Perform operational check-out of nose wheel steering system.

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NOTE

The Menasco (P/N 1176L001) and Sterer (P/N 57640) nose steering control valve are interchangeable. Before ordering replacement parts, make sure what valve is on airplane. Refer to S/L 30 for additional information.

REPLACEMENT PARTS

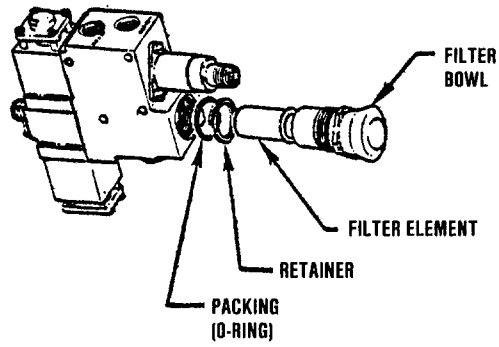
Airplanes with a Menasco nose steering control valve (P/N 1176L001).

- Filter element, P/N AN6235-1A
- Preformed packing (O-ring), P/N AN6227B19
- Retainer, P/N MS28782-19

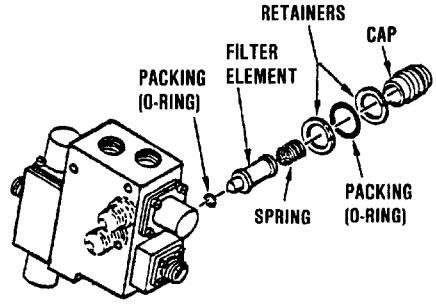
Airplanes with a Sterer nose steering control valve (P/N 57640).

- Preformed packing (O-ring), P/N MS28775-011
- Preformed packing (O-ring), P/N MS28775-019
- Retainers (2), P/N MS28774-019
- Filter Element (1), P/N 35582

MENASCO NOSE STEERING CONTROL VALVE
(P/N 1176L001)



STERER NOSE STEERING CONTROL VALVE
(P/N 57640)



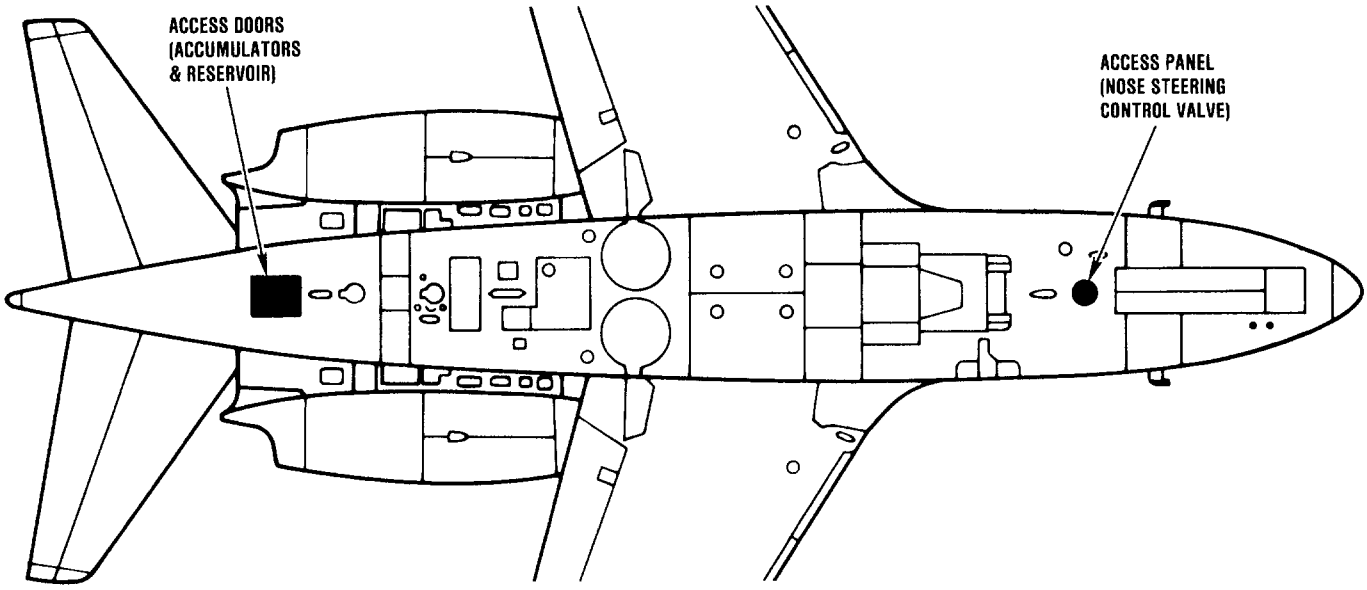


Figure 32-50-404A — Removing and Installing Nose Steering Control Valve Filter Element

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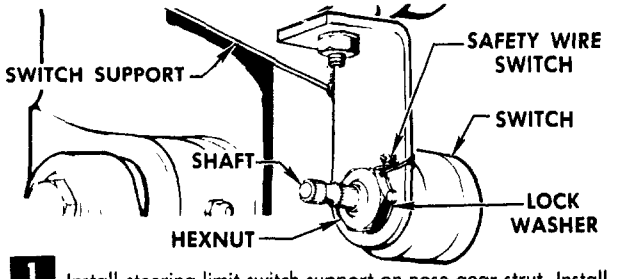
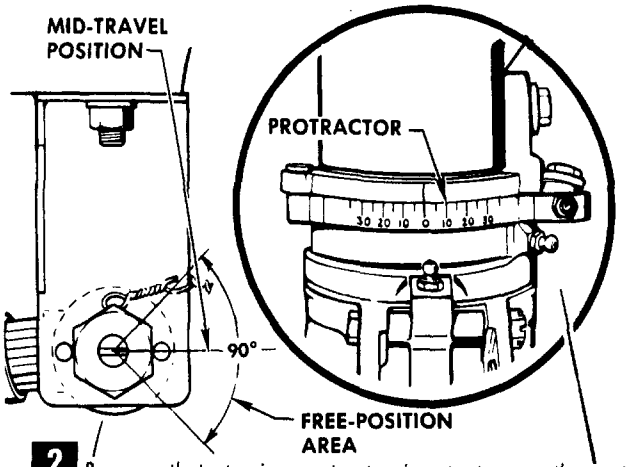
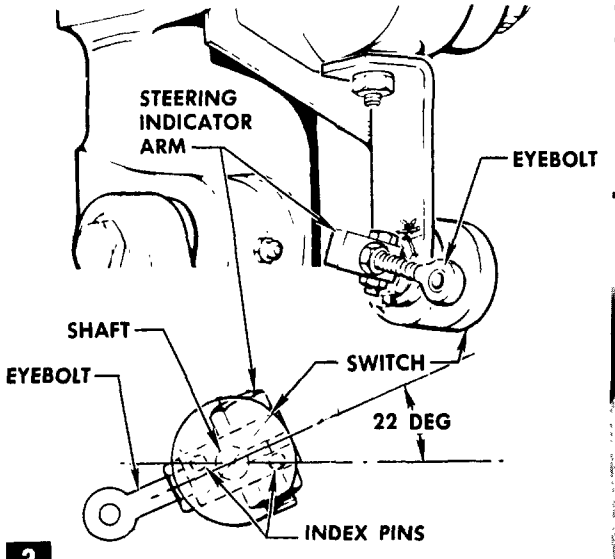


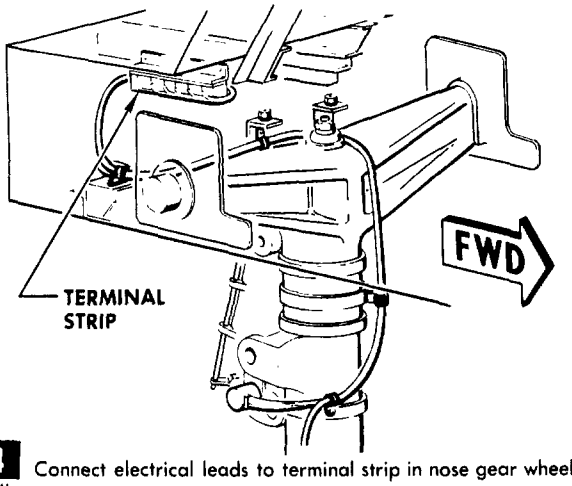
1 Install steering limit switch support on nose gear strut. Install steering limit switch on support. Safety hexnut to switch support, using MS2099F32 lock wire.



2 Be sure that steering protractor is set at zero; then set switch at its mid-travel position.

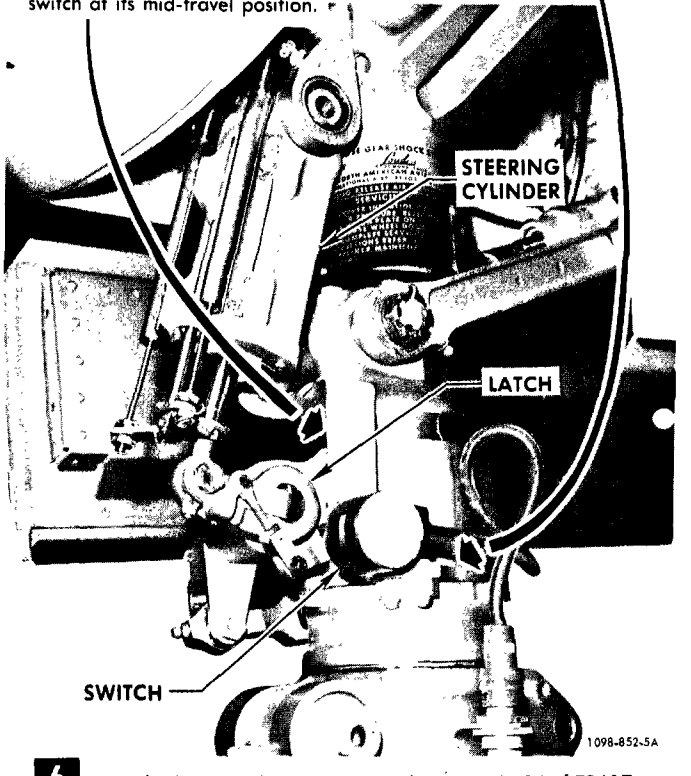


3 Taking care not to move switch shaft, install steering indicator arm on switch shaft 22 (±1/2) degrees from line through index pins on switch, and adjust eyebolt (265-346263).



4 Connect electrical leads to terminal strip in nose gear wheel well.

5 Attach ohmmeter between terminals "3" and "4" of TB617, and manually rotate steering sector for right-hand steering. Ohmmeter must indicate continuity at or before 5 degrees on steering sector.



6 Attach ohmmeter between terminals "1" and "2" of TB617, and manually rotate steering sector for left-hand steering. Ohmmeter must indicate continuity at or before 5 degrees rotation on steering sector.

7 The angle of rotation required to make continuity in these checks must be equal to ±1 degree for both left and right rotation above the 0-degree position on the steering sector. Readjust eyebolt assembly or rotate arm about switch shaft to obtain this symmetry if required.

8 Manually rotate gear full left and right against stops and observe movement of arm to determine that there is no over-centering or binding action of linkage.

S370-2-33-51

Installing and Adjusting Feed-back Potentiometer Monitor Switch
Figure 32-50-405

ADJUSTMENT/TEST

FUNCTIONAL TEST — NOSE WHEEL STEERING SYSTEM (AIRPLANES NOT CHANGED BY S/B 85-2)

EQUIPMENT/TOOLS

- 28 VDC Electrical Power Supply
- Nose Wheel Steering Test Box
- Airplane Tow Bar

NOTES

- ***This functional test will require an outside observer monitoring the nose wheel steering collar.***
- ***These procedures may be used with any type control box installation (P/N's 1176L025-3 or CM533 or 57730); however, when control box Sterer P/N 57730 is installed, the test box must be modified per Figure 32-50-501. Following modification the test box may be used with any control box installation.***

1. Attach airplane tow bar to disengage and lock-out steering latch on NLG steering collar.
2. Remove upper access panel on LH side of nose compartment to gain access to main and standby steering control boxes.
3. Disconnect plug P853 from main steering control box (CN501). Connect test box in circuit between plug P853 and main steering control box (see Figure 32-50-501).
4. Position test box switches M2 and M3 to NORM.
5. Position landing gear handle DOWN.
6. Disengage gust lock (if engaged).
7. Position the following circuit breakers in: LDG GEAR-POS, NLG STEER-MAIN, NLG STEER-STBY.
8. Connect 28 VDC electrical power to airplane and turn on.

NOTE

A hydraulic test stand or the airplane system hydraulic pump may be used to perform this check. Because a hydraulic test stand may not always be available, this check out is based on the use of the airplane system hydraulic pump.

CAUTION

Restrict hydraulic pump operations to minimum to prevent pump overheating.

9. Position HYD PUMP switch to AUTO.
10. Position test box switches M2 and M3 to NORM.
11. Actuate rudder pedals full left and right. Observe full travel of nose steering collar.
12. Position test box switches M2 and M3 to OPEN.
13. Position the NOSE GR STEER switch to MAIN.
14. Press and release the NOSE WHEEL STEER button. Observe that NOSE STEER ON light comes on, and MAIN STEER FAIL light is off.
15. Momentarily move test box switch M2 to GND, then to OPEN. Nose steering collar should move left, then stop.
16. Momentarily move test box switch M3 to GND, then to OPEN. Nose steering collar should move right, then stop.
17. Position test box switches M2 and M3 to GND. NOSE STEER ON and MAIN STEER FAIL lights should come on.
18. Position test box switches M2 and M3 to OPEN. Observe that NOSE STEER ON and MAIN STEER FAIL lights come on. Steering collar should turn when the rudder pedals are depressed.
19. With rudder pedals in neutral, set test box switches M2 and M3 to NORM. Reset system to MAIN by momentarily placing NOSE GR STEER selector switch to STANDBY, hesitate, then back to MAIN. Depress and release steering button on control wheel. Observe that NOSE STEER ON light comes on.

NOTES

- ***If system does not reset to main, open and close NLG STEER-STBY circuit breaker.***
- ***Steps 20 and 21 applicable only to aircraft changed by S/L 30.***

20. Position test box switch M2 to OPEN, then back to NORM. NOSE STEER ON light should come on, MAIN STEER FAIL light should be off, and the steering collar should not move when rudder pedals are depressed.
21. Reset system to MAIN, as per step 12.

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ADJUSTMENT/TEST (Continued)

FUNCTIONAL TEST — NOSE WHEEL STEERING SYSTEM (AIRPLANES NOT CHANGED BY S/B 85-2) (Cont'd)

22. Depress right rudder pedal. While steering collar is turning, place test box switch M2 to GRD. The MAIN STEER FAIL and NOSE STEER ON lights should come on, and the steering collar should continue to turn right.
23. Reset system to MAIN, as per step 12.
24. Depress left rudder pedal. While steering collar is turning, place test box switch M3 to GRD. The MAIN STEER FAIL and NOSE STEER ON lights should come on, and the steering collar should continue to turn left.
25. Position test box switch M3 to NORM. Reset system to MAIN, as per step 12.
26. To verify the rigging of both command monitor switches and the operation of related monitor relay contacts, proceed as follows:
    A. With rudder pedals at neutral and test box switches M2 and M3 at OPEN, engage system.
    B. Test box switch M2 to GRD. Observe that steering collar moves left.
    C. Slowly move right rudder pedal forward. Stop movement as soon as transfer occurs (MAIN STEER FAIL light comes on). Pedal should not move more than 1/2 inch forward of neutral.
    D. Reset system to MAIN, as per step 12. Move M2 and M3 to OPEN.
    E. Test box switch M3 to GRD. Observe that steering collar moves to full right.
    F. Slowly move left rudder pedal forward. Stop movement as soon as transfer occurs (MAIN STEER FAIL light comes on). Pedal should not move more than 1/2 inch forward of neutral.
    G. Reset system to MAIN, as per step 12.
27. To verify rigging of the follow-up monitor switches and operation of related monitor and transfer relay contacts, proceed as follows:
    A. With rudder pedals at neutral and test box switches M2 and M3 at OPEN, engage MAIN system.
    B. Depress left rudder pedal fully and hold. The steering collar should not move from neutral.
    C. Test box switch M3 to GRD. Observe that steering collar moves to right about 3.5 to 5 degrees, then goes to full left as transfer occurs (MAIN STEER FAIL light comes on).
    D. Reset system to MAIN, as per step 12. Move M2 and M3 to Open.
    E. Depress right rudder pedal fully and hold. Steering collar should not move from neutral.
    F. Test box switch M2 to GRD. Observe that steering collar moves to left about 3.5 to 5 degrees, then goes to full right as transfer occurs (MAIN STEER FAIL light comes on).
28. Reset system to MAIN, as per step 12.
29. Turn off electrical power.
30. Disconnect test box from main steering control box and reconnect plug P853.
31. Disconnect electrical plug P854 from standby steering control box (CN515). Connect steering test box between electrical plug P854 and standby steering control box.
32. Turn on electrical power.
33. Position NOSE GR STEER switch (center console) to STANDBY. The NOSE STEER ON and MAIN STEER FAIL lights should come on.
34. Position test box switches M2 and M3 to NORM.
35. Actuate rudder pedals full left and right. Observe full travel of the nose steering collar.
36. Position test box switches M2 and M3 to OPEN, and back to NORM. NOSE STEER ON and MAIN STEER FAIL lights should come on, and the steering collar should move when the rudder pedals are depressed.
37. Position NOSE GR STEER switch to MAIN, then back to STANDBY. The NOSE STEER ON and MAIN STEER FAIL lights should come on, and the steering collar should move when the rudder pedals are depressed.
38. Depress right rudder pedal. While steering collar is turning, place test box switch M2 to GRD. The steering collar should stop turning, the NOSE STEER ON light should be off, and the MAIN STEER FAIL light should come on.
39. Position test box switch M2 to NORM.
40. Position the NOSE GR STEER switch to STANDBY, then to MAIN, then back to STANDBY. The NOSE STEER ON and MAIN STEER FAIL lights should come on.
41. Depress the left rudder pedal. While steering collar is turning, place test box switch M3 to GRD. Steering collar should stop turning, the NOSE STEER ON light should be off, and the MAIN STEER FAIL light should come on.
42. Position test box switch M3 to NORM.

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ADJUSTMENT TEST (Continued)

FUNCTIONAL TEST — NOSE WHEEL STEERING SYSTEM (AIRPLANES NOT CHANGED BY S/B 85-2) (Cont'd)

43. Reset system to MAIN, as per step 12.
44. Turn off electrical power.
45. Disconnect test box from standby steering control box and reconnect electrical plug P854 to standby steering control box (CN515). See Figure 32-50-501.
46. Turn on electrical power.
47. Check nose wheel steering for proper operation.
48. Depress left rudder pedal until rudder is 24° left, as indicated on rudder vertical torque tube protractor in the tail cone.
49. Nose wheel steering protractor should indicate wheels are 45° +1-2 degrees to left.
50. Repeat steps 48 and 49 for opposite rudder pedal.
51. Check standby nose wheel steering system for full travel as follows:
    A. Position NOSE GR STEER switch to STANDBY. The MAIN STEER FAIL and MASTER CAUTION lights should come on.

NOTE

The standby system does not require actuation of the pilot's or co-pilot's switch for operation.

52. Depress left rudder pedal until rudder is 24° left, as indicated on rudder vertical torque tube protractor in tail cone. Observe that the steering collar moves smoothly full left against stop.
53. Repeat steps 51 and 52 for opposite rudder pedal.
54. With rudder pedals in neutral, reset system to MAIN by momentarily placing NOSE GR STEER selector switch to STANDBY, hesitate, then back to MAIN. Depress and release steering button on control wheel. The NOSE STEER ON light should come on.
55. Actuate the rudder pedals full left and right. Observe full travel of the nose steering collar.
56. Position the HYD PUMP switch to OFF.
57. Disconnect electrical power from airplane.
58. Install the upper access panel on the left side of the fuselage.
59. Engage the nose gear steering latch lock-out.
60. Secure airplane as necessary.

FUNCTIONAL TEST — NOSE WHEEL STEERING SYSTEM — PART 1 (AIRPLANES MODIFIED BY S/B 85-2)

EQUIPMENT/TOOLS

- 28 VDC Electrical Power
- Airplane Tow Bar or Dummy Tow Bar

NOTES

- ***This functional test will require an outside observer monitoring the nose wheel steering collar.***
- ***A functional test of the nose wheel steering system must be performed if a unit has been replaced, hydraulic lines have been disconnected, or an adjustment has been made on any unit. The test or testing procedure must be thorough enough to ensure that the system if free of hydraulic leaks and that no malfunction could occur because of improper maintenance.***

1. Disengage and lockout the NLG steering collar. (Use a dummy tow bar or standard tow bar.)
2. Push in the following circuit breakers: LDG GEAR-POS, NLG STEER-MAIN, and NLG STEER-STBY.
3. Connect a 28 VDC electrical power source to the airplane. Turn power ON.
4. Position HYD PUMP switch to ON or AUTO.
5. Disengage gust lock, if engaged.
6. Ensure that the MAIN nose steering system is engaged.
7. Press and hold the NOSE STEER STBY switch/light.
8. Depress right rudder pedal and observe the following conditions:
    A. NOSE STEER ON light ..... on
    B. NOSE STEER STBY light ..... on (release switch)
    C. NOSE STEER TRAIL light ..... off
    D. MAIN STEER FAIL light ..... on
    E. MASTER CAUTION lights ..... on
9. Position rudder pedals to neutral.
10. Cycle the NOSE GR STEER switch from MAIN to STANDBY to MAIN.

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ADJUSTMENT/TEST (Continued)

FUNCTIONAL TEST — NOSE WHEEL STEERING SYSTEM — PART 1 (AIRPLANES MODIFIED BY S/B 85-2) (Cont'd)

11. Press and release the NOSE WHEEL STEER button and observe the following conditions:
    A. NOSE STEER ON light .....on
    B. NOSE STEER STBY light..... off
    C. NOSE STEER TRAIL light..... off
    D. MAIN STEER FAIL light ..... off
    E. MASTER CAUTION lights ..... off

NOTE

While steering collar is turning, press and release the nose gear steer standby switch/light.

12. Position rudder pedals full right and hold.
13. Press and release the NOSE STEER STBY switch/light and observe the following:
    A. NOSE STEER ON light .....on
    B. NOSE STEER STBY light.....on
    C. NOSE STEER TRAIL light..... off
    D. MAIN STEER FAIL light .....on
    E. MASTER CAUTION lights .....on
14. Position rudder pedals to neutral.
15. Cycle the NOSE GR STEER switch from MAIN to STANDBY to MAIN.
16. Press and release the NOSE WHEEL STEER button and observe the following:
    A. NOSE STEER ON light .....on
    B. NOSE STEER STBY light..... off
    C. NOSE STEER TRAIL light..... off
    D. MAIN STEER FAIL light ..... off
    E. MASTER CAUTION lights ..... off
17. Press and hold the NOSE STEER STBY switch/light.
18. Depress rudder pedal full left and observe the following:
    A. NOSE STEER ON light .....on
    B. NOSE STEER STBY light.....on (release switch)
    C. NOSE STEER TRAIL light..... off
    D. MAIN STEER FAIL light .....on
    E. MASTER CAUTION lights .....on
19. Position rudder pedals to neutral.
20. Cycle the NOSE GR STEER switch from MAIN to STANDBY to MAIN.
21. Press and release the NOSE WHEEL STEER button and observe the following:
    A. NOSE STEER ON light .....on
    B. NOSE STEER STBY light..... off
    C. NOSE STEER TRAIL light..... off
    D. MAIN STEER FAIL light ..... off
    E. MASTER CAUTION lights ..... off

NOTE

While steering collar is turning, press and release the nose gear standby switch/light.

22. Position rudder pedals to full left and hold.
23. Press and release the NOSE STEER STBY switch/light and observe the following:
    A. NOSE STEER ON light .....on
    B. NOSE STEER STBY light.....on
    C. NOSE STEER TRAIL light..... off
    D. MAIN STEER FAIL light .....on
    E. MASTER CAUTION lights .....on
24. Reset the MASTER CAUTION lights.
25. Position the rudder pedals to neutral.
26. Press and hold the NOSE STEER TRAIL light/switch.
27. Position rudder pedals to left or right and observe the following:
    A. NOSE STEER ON light ..... off
    B. NOSE STEER STBY light..... off
    C. NOSE STEER TRAIL light.....on (release switch)
    D. MAIN STEER FAIL light .....on
    E. MASTER CAUTION lights .....on
28. Reset the MASTER CAUTION lights.
29. Push rudder pedals left and right. Observe that steering collar does not move.
30. Cycle the NOSE GR STEER switch from MAIN to STANDBY to MAIN.

NOTE

Correct any discrepancies noted in Part 1 before proceeding to Part 2.

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ADJUSTMENT/TEST (Continued)

FUNCTIONAL TEST — NOSE WHEEL STEERING SYSTEM (AIRPLANES MODIFIED BY S/B 85-2) PART 2 — RIGGING AND PROPER OPERATION OF THE FOLLOW-UP MONITOR SWITCHES

NOTE

An outside observer, monitoring the steering collar protractor should verify that transfer occurs 3.5 to 5 degrees (±0 degrees) of steering collar movement from neutral.

1. Position rudder pedals to neutral.
2. Press and release the NOSE WHEEL STEER button and observe the following:
   A. NOSE STEER ON light ..... on
   B. NOSE STEER STBY light ..... off
   C. NOSE STEER TRAIL light ..... off
   D. MAIN STEER FAIL light ..... off
   E. MASTER CAUTION lights ..... off
3. Press and hold the NOSE STEER STBY switch/light.
4. Slowly push rudder pedals right. When NOSE STEER STBY light comes on, signal outside observer that transfer has occurred and release switch.

NOTE

Record steering collar protractor reading as degrees transfer occurs:
_____ Degrees.

5. Position rudder pedals to neutral.
6. Cycle the NOSE GR STEER switch from MAIN to STANDBY to MAIN.
7. Press and release the NOSE WHEEL STEER button and observe the following:
   A. NOSE STEER ON light ..... on
   B. NOSE STEER STBY light ..... off
   C. NOSE STEER TRAIL light ..... off
   D. MAIN STEER FAIL light ..... off
   E. MASTER CAUTION lights ..... off
8. Press and hold the NOSE STEER STBY switch/light.
9. Slowly push rudder pedals left. When NOSE STEER STBY light comes ON, signal outside observer that transfer has occurred and release switch.

NOTE

Record steering collar protractor reading as degrees transfer occurs:
_____ Degrees.

10. Position rudder pedals to neutral.
11. Cycle the NOSE GR STEER switch from MAIN to STANDBY to MAIN.
12. Turn off and disconnect electrical power.
13. Secure airplane as necessary.

NOTES

- ***Add the total number of degrees from steps 4 and 9 to determine rigging error check.***
- ***If transfer occurs at less than 7 degrees or greater than 10 degrees of steering collar movement from neutral, check rigging and operation of monitor switches per the Maintenance Manual.***

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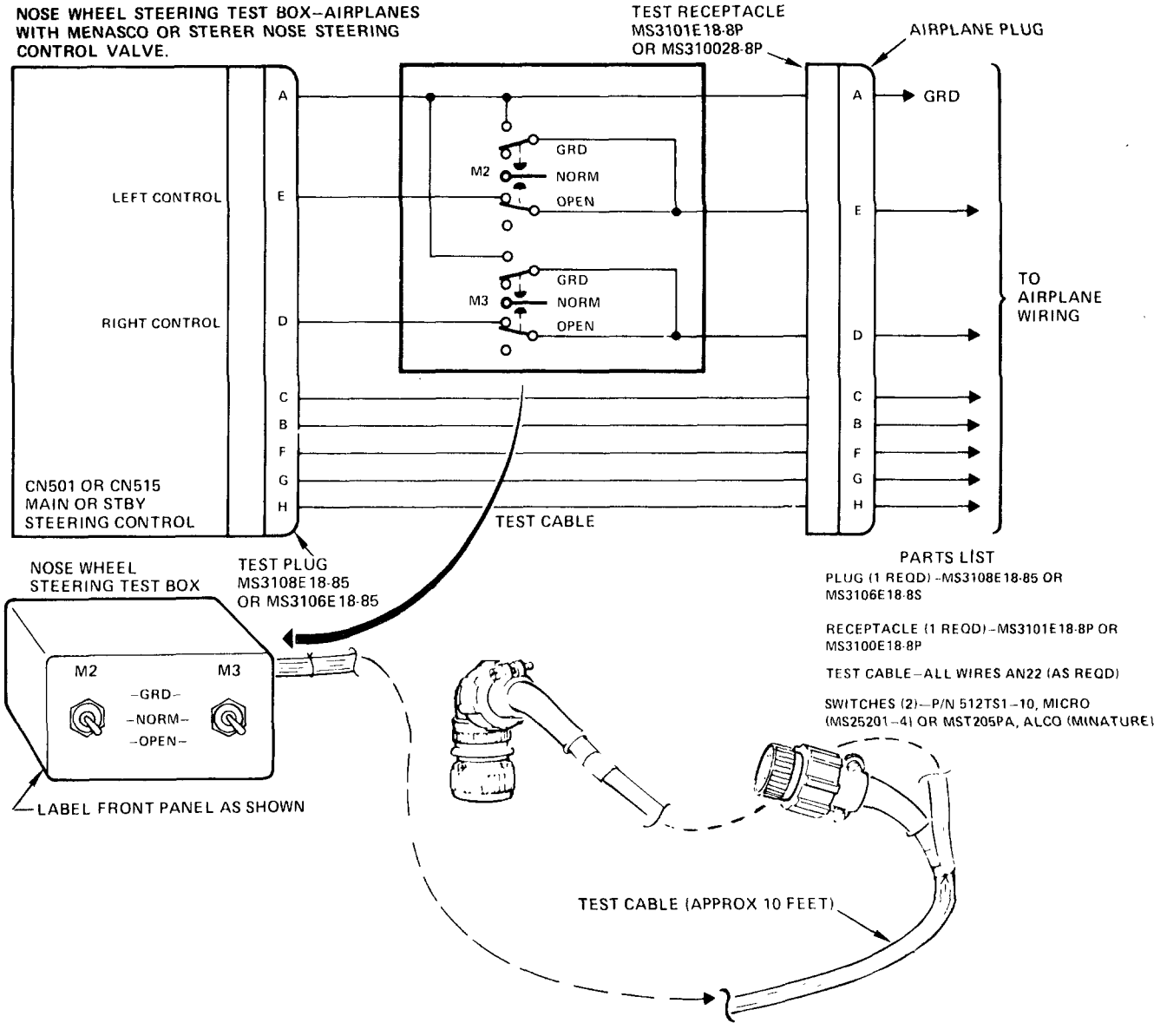

Figure 32-50-501 — Nose Wheel Steering Test Box

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ADJUSTMENT/TEST (Continued)

FUNCTIONAL TEST — NOSE WHEEL STEERING FEEDBACK POTENTIOMETER REDUNDANT WIRING SYSTEM

EQUIPMENT/TOOLS

- 28 VDC Electrical Power Supply
- Airplane Tow Bar, or Dummy Bar

CONSUMABLES

- Safetywire, 0.020 In. Dia.

NOTES

- ***This Functional Test incorporates a test of the main and standby nose wheel steering system and feedback potentiometer redundant cable assembly.***
- ***This Functional Test will require an outside observer monitoring the nose wheel steering collar.***

1. Attach airplane tow bar to disengage and lock-out steering latch on NLG steering collar.
2. Position the landing gear handle DOWN.
3. Disengage gust locks (if engaged).
4. Connect 28 VDC external electrical power to airplane, and turn on.
5. Position the LDG GEAR-POS, NLG STEER-MAIN, and the NLG STEER-STBY circuit breakers in.

NOTE

A hydraulic test stand or the airplane system hydraulic pump may be used to perform this check. Because a hydraulic test stand may not always be available, this test is based on the use of the airplane system hydraulic pump.

CAUTION

Restrict hydraulic pump operations to a minimum to prevent pump overheating.

6. Position the HYDRAULIC PUMP switch to AUTO.
7. Remove the main steer follow-up potentiometer monitor cannon plug (J715) and standby potentiometer monitor cannon plug (J855). See Figure 32-50-502.
8. Position the NOSE GR STEER switch to STANDBY. Observe the NOSE STEER ON lights, MAIN STEER FAIL light, and MASTER CAUTION lights come on.
9. Reset the MASTER CAUTION lights.
10. Position rudder pedals to neutral. Observe that steering collar protractor should be at the 0 degree index mark (±0 degrees).
11. Slowly and smoothly, push rudder pedals full right and hold. Observe that steering collar moves smoothly full right against stop.
12. Position rudder pedals to neutral. Observe that steering collar protractor aligns with the 0 degree index mark (±0 degrees).
13. Slowly and smoothly, push rudder pedals full left and hold. Observe that steering collar moves smoothly full left against stop.
14. Position rudder pedals to neutral. Observe that steering collar protractor aligns with the 0 degree index mark (±0 degrees).
15. Positin the NOSE GR STEER switch to MAIN.
16. Press and release the NOSE WHEEL STEER button. Observe that the NOSE STEER ON lights come on, and the MAIN STEER FAIL light and MASTER CAUTION lights are off.
17. Ensure that the steering collar aligns with the 0 degree index mark (±0 degrees).
18. Slowly and smoothly, push rudder pedals full right and hold. Observe that steering collar moves smoothly full right against stop.
19. Position rudder pedals to neutral. Observe that steering collar protractor aligns with the 0 degrees index mark (±0 degrees).
20. Slowly and smoothly, push rudder pedals full left and hold. Observe that steering collar moves smoothly full left against stop.
21. Position rudder pedals to neutral. Observe that steering collar protractor aligns with the 0 degree index mark (±0 degrees).
22. Reconnect the main steer follow-up potentiometer monitor cannon plug (J715) and standby potentiometer monitor cannon plug (J885). See Figure 32-50-502.
23. Remove main steer follow-up potentiometer monitor cannon plug (J716) and standby potentiometer monitor cannon plug (J856). See Figure 32-50-502.
24. Position the NOSE GR STEER switch to STANDBY. Observe that the NOSE STEER ON light, MAIN STEER FAIL light, and MASTER CAUTION lights come on.
25. Reset the MASTER CAUTION lights.
26. Position the rudder pedals to neutral. Observe the steering collar protractor aligns with the 0 degrees index mark (±0 degrees).

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ADJUSTMENT/TEST (Continued)

FUNCTIONAL TEST — NOSE WHEEL STEERING FEEDBACK POTENTIOMETER REDUNDANT WIRING SYSTEM (Cont'd)

27. Slowly and smoothly, push rudder pedals full right and hold. Observe that steering collar moves smoothly full right against stop.
28. Position rudder pedals to neutral. Observe that steering collar protractor aligns with the 0 degrees index mark (±0 degrees).
29. Slowly and smoothly, push rudder pedals full left and hold. Observe that the steering collar moves smoothly full left against stop.
30. Position rudder pedals to neutral. Observe the steering collar protractor aligns with the 0 degrees index mark (±0 degrees).
31. Position the NOSE GR STEER switch to MAIN.
32. Press and release the NOSE WHEEL STEER button. Observe that the NOSE STEER ON light comes on, and the MAIN STEER FAIL and MASTER CAUTION lights are off.
33. Ensure that the steering collar protractor aligns with the 0 degrees index mark (±0 degrees).
34. Slowly and smoothly, push rudder pedals full right and hold. Observe that the steering collar moves smoothly full right against stop.
35. Position rudder pedals to neutral. Observe that steering collar protractor aligns with the 0 degrees index mark (±0 degrees).
36. Slowly and smoothly, push rudder pedals full left and hold. Observe that steering collar has moved smoothly full left against stop.
37. Position rudder pedals to neutral. Observe that steering collar protractor aligns with the 0 degree index mark (±0 degrees).
38. Reconnect the main steer follow-up potentiometer monitor cannon plug (J716) and standby potentiometer monitor cannon plug (J856). See Figure 32-50-502.
39. Actuate rudder pedals full left and right. Observe full travel of nose steering collar.
40. Position the HYDRAULIC PUMP switch to OFF.
41. Disconnect 28 VDC external electrical power.
42. Actuate rudder pedals full left and right.
43. Engage nose steering latch lock-out.
44. Secure airplane as necessary.

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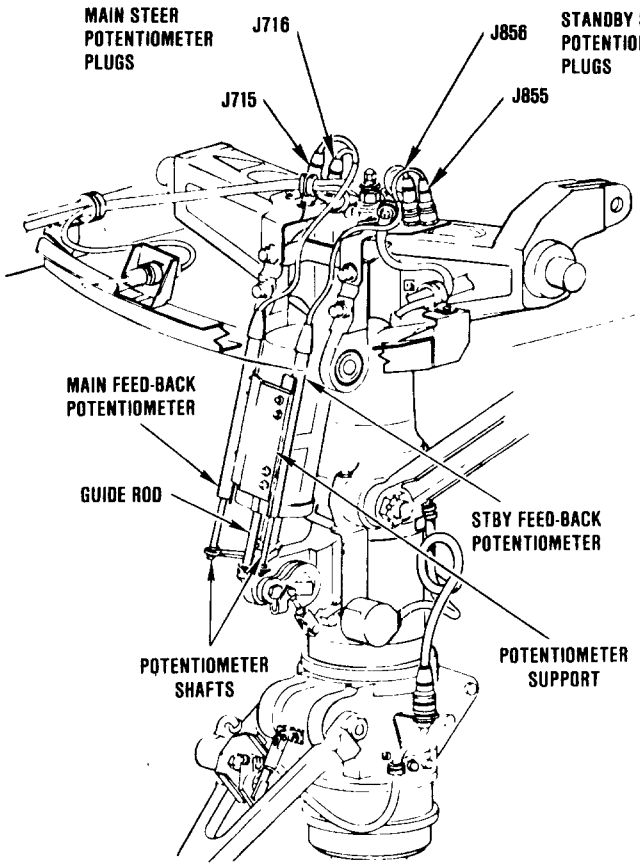
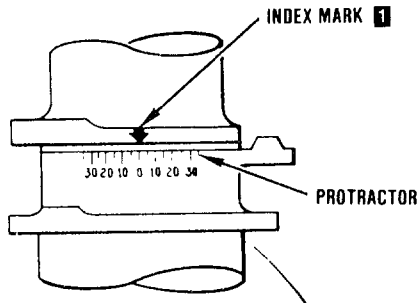
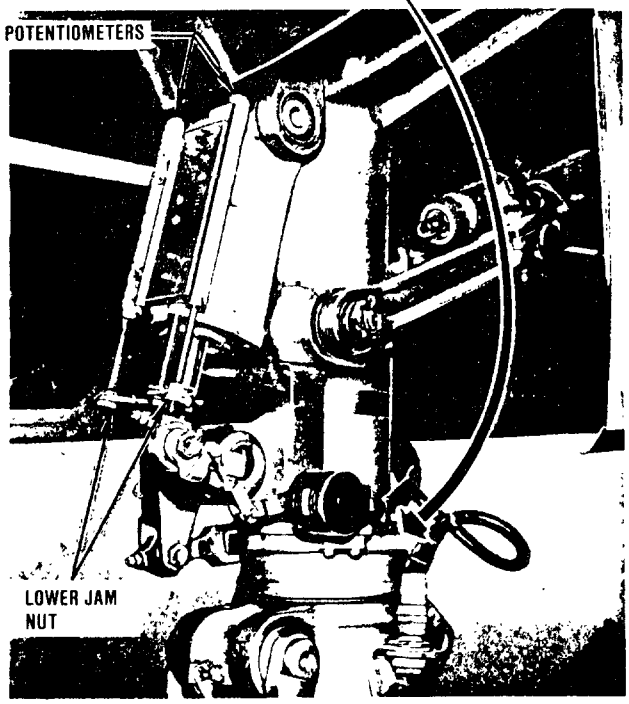


Detail A





Detail B

1 IT IS IMPORTANT THAT INDEX MARK BE VIEWED DIRECTLY, IN A HEAD-ON DIRECTION, TO ENSURE ACCURATE READINGS. VIEWING FROM ANY ANGLE CAUSES A PARALAX VIEW PROBLEM, RESULTING IN ERRONEOUS READINGS.

Figure 32-50-502 — Nose Landing Gear Steering Detail

NEW

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INSPECTION/CHECK

INSPECTION OF NOSE GEAR STEERING SYSTEM (See Figure 32-50-601)

1. On S/N 370-2 thru -6, -8 and -9, raise nose wheel off the ground using jack point on nose gear strut. On S/N 370-1, -7, and 380-1 and Subs, raise nose gear wheels off ground using jacks at all three fuselage points.
2. Check steering link end play. If either spherical bearing exceeds .020 inch, replace link.
3. Check steering bellcrank bushings for excess wear. Radial play should not exceed .004 inch.
4. Torque steering bellcrank attach bolt to 25-30 in.-lbs, then safetywire.
5. Tighten steering link attaching bolt nuts finger tight. Install cotter pin.
6. Lower airplane and remove jack.

NOSE GEAR STEERING LINK INSPECTION (See Figure 32-50-601)

1. Position jack under nose jack point and raise airplane until nose wheel clears ground.
2. Check steering link end play. If either spherical bearing exceeds .020 inch, replace link.
3. Check steering bellcrank bushings for excess wear. Radial play should not exceed .004 inch.
4. Torque steering bellcrank attach bolt to 25-30 in.-lbs, and safetywire.
5. Tighten steering link attaching bolt, nut, and cotter key.
6. Lower airplane and remove jacks.

INSPECT NOSE GEAR STEERING LINK (See Figure 32-50-601)

1. On S/N 370-2 thru -6, -8 and -9, raise nose wheel off the ground using jack point on nose gear strut. On S/N 370-1, -7, and 380-1 and Subs, raise nose gear wheels off ground using jacks at all three fuselage points.
2. Remove bolt attaching nose steering link to nose steering collar.
3. Check steering link end play (dial indicator or equivalent). If spherical bearing inside link exceeds .020 inch, replace link.
4. Install link to steering collar bolt. Tighten nut finger tight and install cotter key.
5. Lower airplane and remove jack or jacks.
6. Perform operational checkout of nosewheel steering system.

NOSE LANDING GEAR SHOCK STRUT BELLCRANK INSPECTION — P/N 3010812-1, -3, -5, -9 and -11 (See Figure 32-50-601)

1. Remove bolt attaching nose steering link to arm.
2. Remove bolt attaching nose steering bellcrank to nose gear steering cylinder rod end.
3. Remove bolt attaching nose steering arm to bellcrank.
4. Loosen nose steering limit switch bracket bolt.
5. Remove bellcrank from nose gear shock strut.
6. Perform a magnetic particle inspection of nose gear steering bellcrank for cracks.

NOTE

If there is evidence of excessive bellcrank shaft or bushings radial wear, perform step 7.

7. Measure bellcrank shaft outside diameter. It must be within 1.1850 to 1.1860 inches. Measure bushings inside diameter. It must be within 1.1870 to 1.1895 inches.
8. Install serviceable bellcrank into nose gear shock strut. Install arm on bellcrank. Ensure that index mark (dot) on end of bellcrank is aligned with index mark on arm.
9. Install arm to bellcrank attach bolt. Torque bolt (25 to 30 in.-lbs) and safety with lockwire.
10. Install link to arm attach bolt. Tighten nut (finger tight) and install cotter key.
11. Position limit switch actuating arm bracket as necessary.
12. Install bolt attaching nose gear steering cylinder rod end to bellcrank. Tighten lock nut snuggly until all end play between bolt head and nut is removed. On S/N 370-1 thru -9 and 380-1 thru -42 not changed by S/L 19, do not apply load to bellcrank clevis. On S/N 370-1 thru -9 and 380-1 thru -42 changed by S/L 19, and 380-43 and Subs, tighten castellated nut finger tight and install cotter pin.
13. Tighten limit switch supporting bracket attach bolt.
14. Check for proper installation and safeties.
15. Perform operational checkout of nose wheel steering system.

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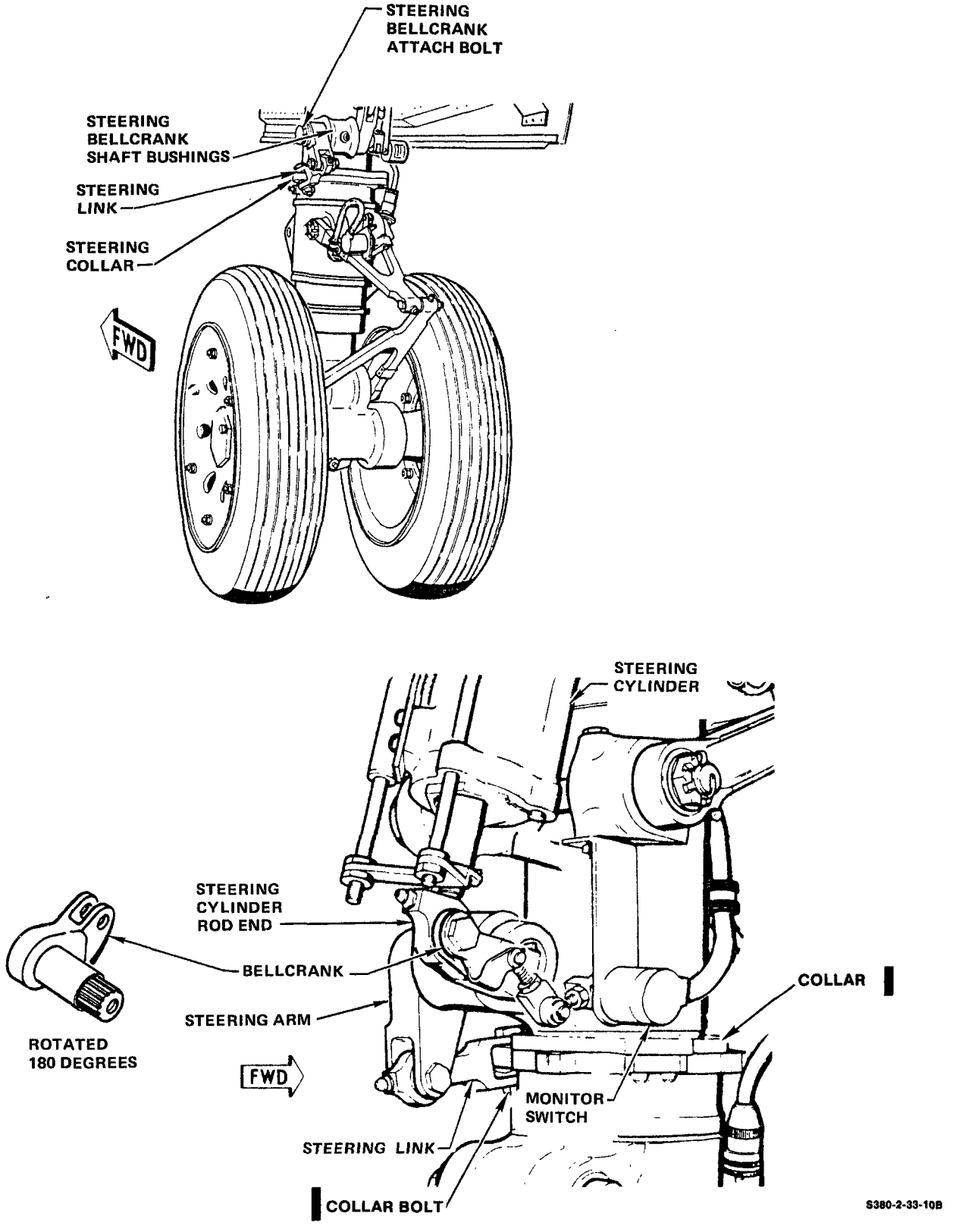

Figure 32-50-601 — Nose Gear Steering Bellcrank and Link Inspection

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INSPECTION/CHECK (Continued)

INSPECT — MLG WHEELS AND NLG WHEEL ASSEMBLIES

The wheel assemblies are a critical item. Due to high energy absorption, supporting varying loads under many different situations, Sabreliner Corporation has determined that a single criteria inspection for the wheel assemblies is inappropriate. The wheel assemblies require extreme care in handling and maintenance for their appropriate service life. For inspection, overhaul, and tire mounting and dismounting, refer to Loral Component Maintenance Manual, AP426 for the main wheel assembly, and AP333 for the nose wheel assembly. Contact Loral Systems Group, 1210 Massillon Rd., Akron, OH 44315 (Tel. 216-796-7028).

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POSITION AND WARNING - DESCRIPTION AND OPERATION

1. Landing Gear Position Indicator Lights

The landing gear position indicator lights consist of three separate indicating units, one for each main gear and one for the nose gear. The three units are mounted in a group below the instrument panel and above the landing gear control handle. Each position light is a lamp assembly containing a green lens. The position lights receive electrical power from the 28-volt essential bus through the "LANDING GEAR - POS" circuit breaker and through the downlock switches to the light. The landing gear position lights may be tested or the light intensity varied. The brightness of the position lights may be dimmed by moving the "FLIGHT INSTRUMENT LIGHTS" rheostat control knob, on the right console panel, to DIM. The lights may be dim-and-bright tested simultaneously by moving the "LIGHTS - CAUTION TEST" switch, on the overhead panel, to DIM and/or BRT.

2. Landing Gear Diodes

The landing gear diodes are a solid state device. These diodes are used as blocking and protective devices for operation and protection of the various components of the landing gear system. The diodes are not adjustable, and when faulty, they must be replaced.

3. Landing Gear Mechanically Actuated Switches

Switches with environmentally sealed metal cases sequence the landing gear position selector valve during operation of the landing gear system. The switches also operate the position indicating and warning system. (See figure 32-00-1.) Each switch is environmentally sealed against moisture and must be replaced as a unit when faulty. Single or multiple switches are mounted in individual cases, and are operated by a cam-and-shaft mechanism. A switch actuating arm is installed on the knurled and grooved end of the switch shaft mechanism. Actuating linkage to each unit must be individually adjusted to precise measurements. (Refer to adjustment procedures of individual switches.)

4. Throttle-actuated Landing Gear Warning Cutout Switches

There are two landing gear throttle-actuated warning cutout switches, one attached to each throttle lever in the engine control quadrant. Each switch is mounted on a bracket below the forward end of the engine control quadrant and is adjustable in relation to a lever attached to the throttle. Movement of the throttle to 95% rpm causes the lever attached to the throttle lever to actuate the warning switch. Once the switches are adjusted when they are installed, they do not require further adjustment.

5. Airspeed and Altitude Warning Switches

The airspeed and altitude warning switches are mounted in a single unit on the right side of the fuselage just forward of the instrument panel in the cockpit. (See figure 32-00-1.) The switches are connected to the pitot and static pressure systems and are operated by a variation in the airplane altitude and airspeed. The setting of switches may be changed, but this should not be done by line maintenance personnel. If the unit fails, it should be replaced.

6. Landing Gear Audible Warning Signal Generator

The landing gear audible warning signal generator is in the bulkhead between the cockpit and the left side of the baggage compartment. The signal generator produces an audible warning signal to be transmitted to the pilot's and copilot's head-sets. The signal is produced by interrupting the incoming dc electrical current by a make-and-break switch (vibrator). This produces a tone signal operating at a frequency determined by the set frequency of the vibrator. The tone signal is then interrupted by the opening and closing of contacts in a relay. The frequency at which the relay operates is determined by the value of a condenser and resistor across the relay coil. This interrupts the tone signal transmitted to the head-sets in each cockpit, causing a tone similar to a buzz-buzz-buzz. There is no method of adjusting the signal generator. Any malfunction is cause for unit replacement.

7. Landing Gear Position Selector Switch

The landing gear control handle contains two landing gear position warning lights that come on when the landing gear is in an unsafe position. On airplanes Serial No. NA-370-2 thru 6 changed by Service Bulletin 72-6, NA-370-1, 7, 8 and 9 and NA-380-1 and later, a main landing gear strut door position switch is installed adjacent to each main landing gear actuating cylinder. These switches monitor the position of the strut doors. If the strut doors do not close after landing gear retraction, electrical power is applied to the red light in the landing gear handle. (See figure 32-00-3.)

POSITION AND WARNING - MAINTENANCE PRACTICES

1. Replacement of Landing Gear Position Indicating and Warning Light Dimming Resistors

A. Removing Dimming Resistor

(1) Remove electrical power, necessary accesses, and relay box cover.

(2) Disconnect resistor electrical leads and remove resistor mounting screws. Remove resistor.

B. Installing Dimming Resistor

(1) Position and install resistor in relay box.

(2) Connect resistor electrical leads.

(3) Install all accesses.

C. Adjusting Dimmer Resistors

(1) Obtain an ohmmeter for this procedure.

(2) Remove electrical power, necessary access panels, and relay box cover.

(3) Loosen clip attaching center electrical lead to resistor. The individual resistors connected to the landing gear position and warning lights are as follows: resistor R514, left gear; R515, right gear; and R516, nose gear. Resistor R510 is connected to the warning light.

(4) Using ohmmeter, set resistors R514, R515, and R516 at 750 ohms each. Set resistor R510 at 150 ohms.

(5) Check intensity of individual lights as follows:

(a) Push in "LANDING GEAR - POS" and "LIGHTS - IND TEST & STEP" circuit breakers on right overhead aft and right overhead forward dc essential circuit-breaker panels, respectively.

(b) Move flight instrument light "INDIRECT" control knob on right console panel to DIM.

(c) To check intensity of landing gear warning light in landing gear control handle, move caution light test switch on overhead center panel to DIM.

(6) Remove electrical power. Check that screw attaching electrical lead to resistor is tight. Install cover on relay box. Install all accesses.

2. Replacement of Main Gear Load Switches

A. Removing Main Gear Load Switch

NOTE: The main gear load switch and the main gear downlock switch are connected by electrical leads (pigtails) to a common disconnect plug. Therefore, to remove either switch, it is necessary to remove the other switch.

(1) Remove electrical power from landing gear circuit. Pull "LANDING GEAR - POS" and "LANDING GEAR - CONT" circuit breakers on right overhead aft dc essential circuit-breaker panel.

(2) Disconnect main gear load and main gear downlock switch linkage at individual switch arms.

(3) Remove clamps securing electrical leads to main gear strut.

(4) Break lockwire and remove jam nut securing main gear load switch to switch bracket. Remove switch from bracket.

(5) Break lockwire and remove jam nut securing main gear downlock switch to switch bracket. Remove switch from bracket.

(6) Remove clamps from aft strut well and disconnect main gear strut electrical harness disconnect plug at aft outboard side of wheel well. Remove electrical harness and main gear load and main gear downlock switches as a unit.

B. Installing and Adjusting Main Gear Load Switch

See figure 32-60-401.

3. Replacement of Main Gear Downlock Switches

A. Removing Main Gear Downlock Switch

NOTE: The main gear load switch and main gear downlock switch are connected by switch electrical leads to a common disconnect plug. Therefore, to remove either switch, it is necessary to remove the other switch.

(1) Remove electrical power from landing gear circuit. Pull "LANDING GEAR - CONT" and "LANDING GEAR - POS" circuit breakers.

(2) Do steps (2) through (5) of procedure for removing main gear load switch and remove switch.

B. Installing and Adjusting Main Gear Downlock Switch

See figure 32-60-402.

4. Replacement of Main Gear Uplock Switches

A. Removing Main Gear Uplock Switch

(1) Remove electrical power from landing gear circuit. Pull "LANDING GEAR - POS" and "LANDING GEAR - CONT" circuit breakers.

(2) Disconnect switch link at switch arm.

(3) Break lockwire and remove jam nut securing switch to switch bracket. Remove switch from bracket.

(4) Remove clamps securing wire bundle along outboard side of wheel well. Remove protective covering from wire bundle.

(5) Separate switch electrical leads from wire bundle, and disconnect switch electrical leads at terminal board in aft wheel well.

(6) Remove switch and electrical leads.

B. Installing and Adjusting Main Gear Uplock Switch

See figure 32-60-403.

5. Replacement of Nose Gear Downlock Switch

A. Removing Nose Gear Downlock Switch

(1) REMOVE electrical power from landing gear circuit. Pull LANDING GEAR - POS and LANDING GEAR - CONT circuit breakers.

(2) Disconnect switch link at switch arm.

(3) Break safetywire and remove jam nut securing switch to switch bracket. Remove switch from bracket.

(4) Disconnect nose wheel steering system potentiometer electrical disconnect plug.

(5) Disconnect nose gear load switch electrical plug at right side of nose gear strut.

(6) Disconnect electrical plug at FS 108, aft and above nose gear strut.

(7) Remove clamps securing electrical harness to strut and drag brace and remove downlock switch and harness as a unit.

B. Installing and Adjusting Nose Gear Downlock Switch

See figure 32-60-404.

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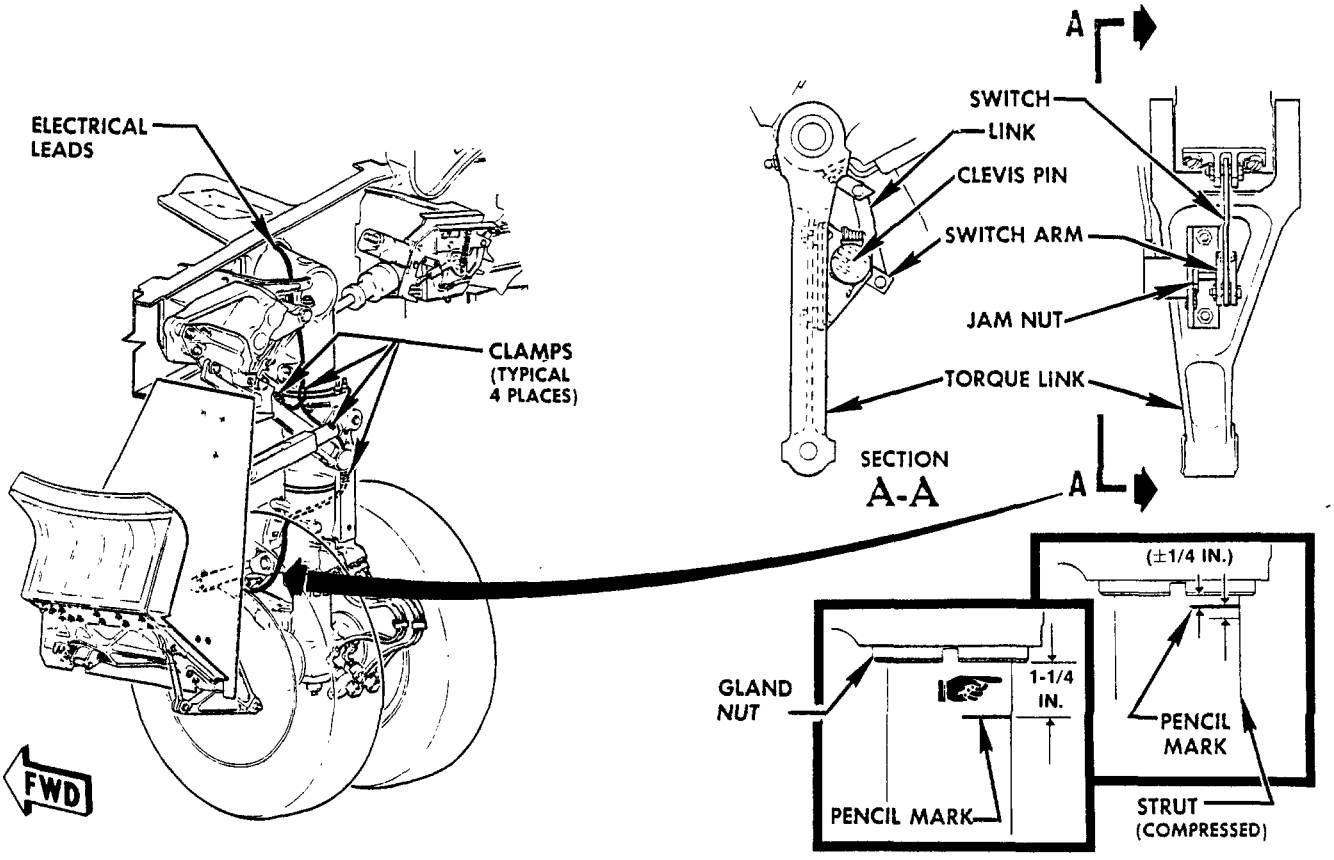

Equipment/Tools

- Continuity Light
- Jacking and Mooring Kit
- Wing Jacks
- Hydraulic Test Stand Capable of Delivering 4 Gpm at 3000 Psi
- 28 Vdc External Electrical Power

NOTE

The following procedure is typical for both main gear struts.

1 Place jacks under airplane and raise airplane enough to permit gear retraction.

2 Position main gear load switch on switch bracket with switch electrical leads toward strut and parallel with torque link.

3 Install, tighten, and safety switch jam nut with safetywire.

4 Connect clamps securing electrical leads to side of strut and side brace.

5 With main gear strut fully extended, draw a pencil line on strut 1-1/4 in. from bottom of gland nut. Place jack under strut and compress strut to pencil mark.

6 Connect a continuity light to pins N and R of electrical plug at end of switch harness.

WARNING

Before applying landing gear hydraulic pressure for any reason (checks, adjustments, etc), deplete normal and auxiliary accumulator hydraulic pressures. This will prevent the auxiliary accumulator power assist retraction system to operate and prevent serious injury to personnel and/or damage airplane.

7 Place switch arm on switch shaft, with arm pointing away from torque link as shown. Do not tighten switch arm to switch shaft. Attach arm to switch link as shown.

8 Looking at shaft end of switch, rotate shaft within switch arm (cw on left gear, ccw on right gear) until continuity light goes out. Continue rotation in same direction until continuity light just comes on. At this point, tighten switch arm on switch shaft.

9 Extend strut. The continuity light should go out before strut is fully extended.

10 Again compress strut. The continuity light should come on when pencil mark is ∫1/4 in. from gland nut.

11 Lower strut and remove jack. Remove continuity light.

12 Install and adjust downlock switch.

13 Install wire harness to aft strut well and connect electrical harness disconnect plug.

14 Check operation of landing gear.

Installing and Adjusting Main Gear Load Switch
Figure 32-60-401

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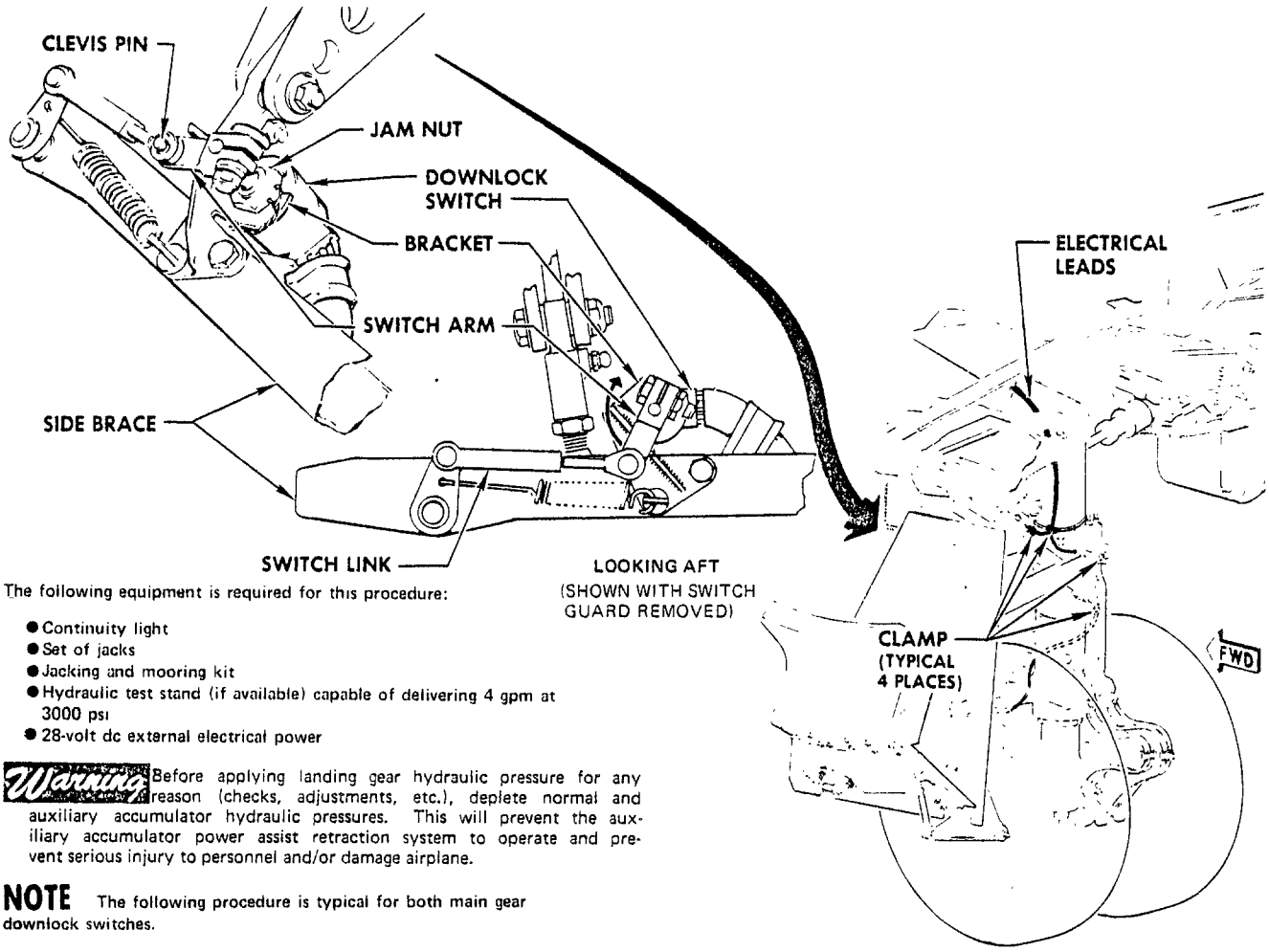


The following equipment is required for this procedure:

- Continuity light
- Set of jacks
- Jacking and mooring kit
- Hydraulic test stand (if available) capable of delivering 4 gpm at 3000 psi
- 28-volt dc external electrical power

Warning Before applying landing gear hydraulic pressure for any reason (checks, adjustments, etc.), deplete normal and auxiliary accumulator hydraulic pressures. This will prevent the auxiliary accumulator power assist retraction system to operate and prevent serious injury to personnel and/or damage airplane.

NOTE The following procedure is typical for both main gear downlock switches.

1 Place jacks under airplane and raise airplane enough to permit gear retraction.

2 Before installing switch, make sure there is one lock washer and one jam nut on switch. Position main gear downlock switch on switch bracket with switch electrical leads inboard and parallel with side brace. Seat switch index pins in holes in bracket. Connect continuity light to pins "A" and "C" of switch electrical harness disconnect plug (P633 LH or P634 RH).

3 Install, tighten, and safety switch jam nut with lockwire.

4 Connect clamps securing electrical leads to side of strut and side brace.

5 Install and adjust main gear load switch.

6 Adjust downlock switch connecting link to 3-1/32 inches (3.03 inches), eye to eye.

7 Place arm on switch shaft with arm pointing down. Do not tighten arm to switch. Attach arm to switch connecting link.

8 Looking at switch from shaft end, rotate switch shaft counter clockwise (clockwise on right-hand main gear downlock switch) until discontinuity is indicated across pins "A" and "C." Continue to rotate shaft (same direction) until continuity is indicated across pins "A" and "C." Tighten locking bolt in arm, securing switch arm to switch shaft.

9 Shorten switch connecting link by 2-1/2 turns of clevis.

10 Install and safety clevis pin through switch arm and switch link clevis.

11 Remove continuity light. Connect and safety electrical plug.

11A Install switch guard with two bolts and washers.

12 Check operation of landing gear, and landing gear position indicating and warning system.

S370-2-33-54C

Installing and Adjusting Main Gear Downlock Switch
Figure 32-60-402

The following equipment is required for this procedure:

- **Set of jacks**
- **Jacking and mooring kit**
- **Continuity light**
- **Hydraulic test stand capable of delivering 4 gpm at 3000 psi**
- **28-volt dc external electrical power**

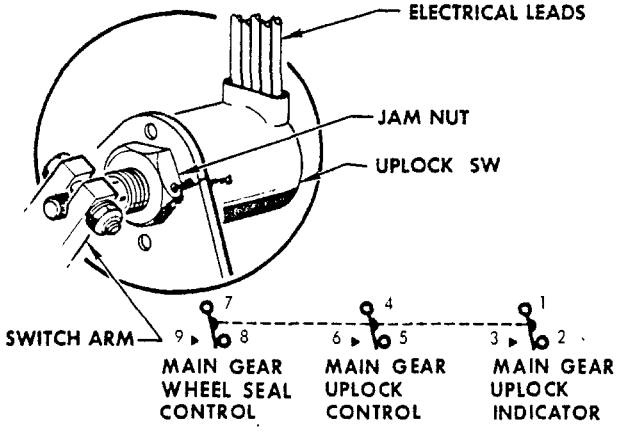
Warning Before applying landing gear hydraulic pressure for any reason (checks, adjustments, etc.), deplete normal and auxiliary accumulator hydraulic pressures. This will prevent the auxiliary accumulator power assist retraction system to operate and prevent serious injury to personnel and/or damage airplane.

NOTE **The following procedure is typical for both main gear uplock switches.**

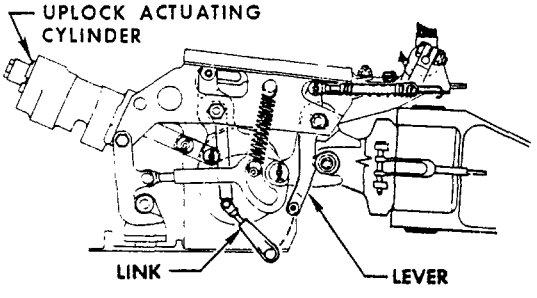
- **The main gear uplock switch is equipped with an electrical lead (pigtail) for each switch contact. Although all electrical leads from the switch are not connected into the landing gear circuit, they must be correctly connected or properly stowed to prevent possible electrical shorts.**

1 Place jacks under airplane and raise airplane enough to permit landing gear retraction. Install main landing gear door maintenance safety locks.

NOTE For better accessibility to the switch mechanism while the gear is up, the wheel and tire assemblies can be removed.



2 Before installing switch, make sure that there is one lock washer and one jam nut on switch. Connect continuity light leads to terminals "1" and "2" of switch (wires 1D47L and 1D48E for left switch or wires 2D47LL and 2D48EE for right switch).

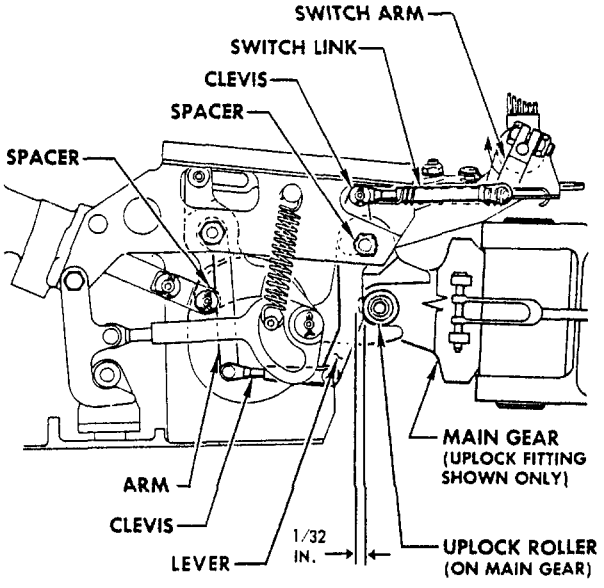


3 Disconnect uplock hook actuating cylinder by removing clevis pin at inboard end of cylinder. Manually close gear uplock hook. To do this, use a screwdriver to pry linkage down from its unlocked overcenter position; then pull inboard on strut door hook until it snaps into locked overcenter position.

4 Position switch on mounting bracket with electrical leads up. Check that index pins seat in index holes in bracket, tighten switch jam nut, and safety with lockwire.

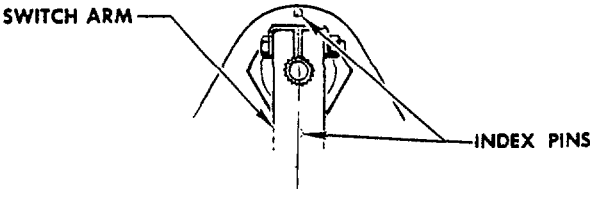
5 Hold arm outboard against spacer. Hold lever outboard against spacer at top. Check that face of lever is 1/32 inch inboard of face of gear uplock hook throat. Adjust length of link to obtain this dimension.

6 Connect link and safety.



7 Position switch arm on shaft loose and hanging down in alignment with center line of switch index pins.

8 Adjust switch link to 2-15/16 inches, eye-to-eye, and connect to switch arm.



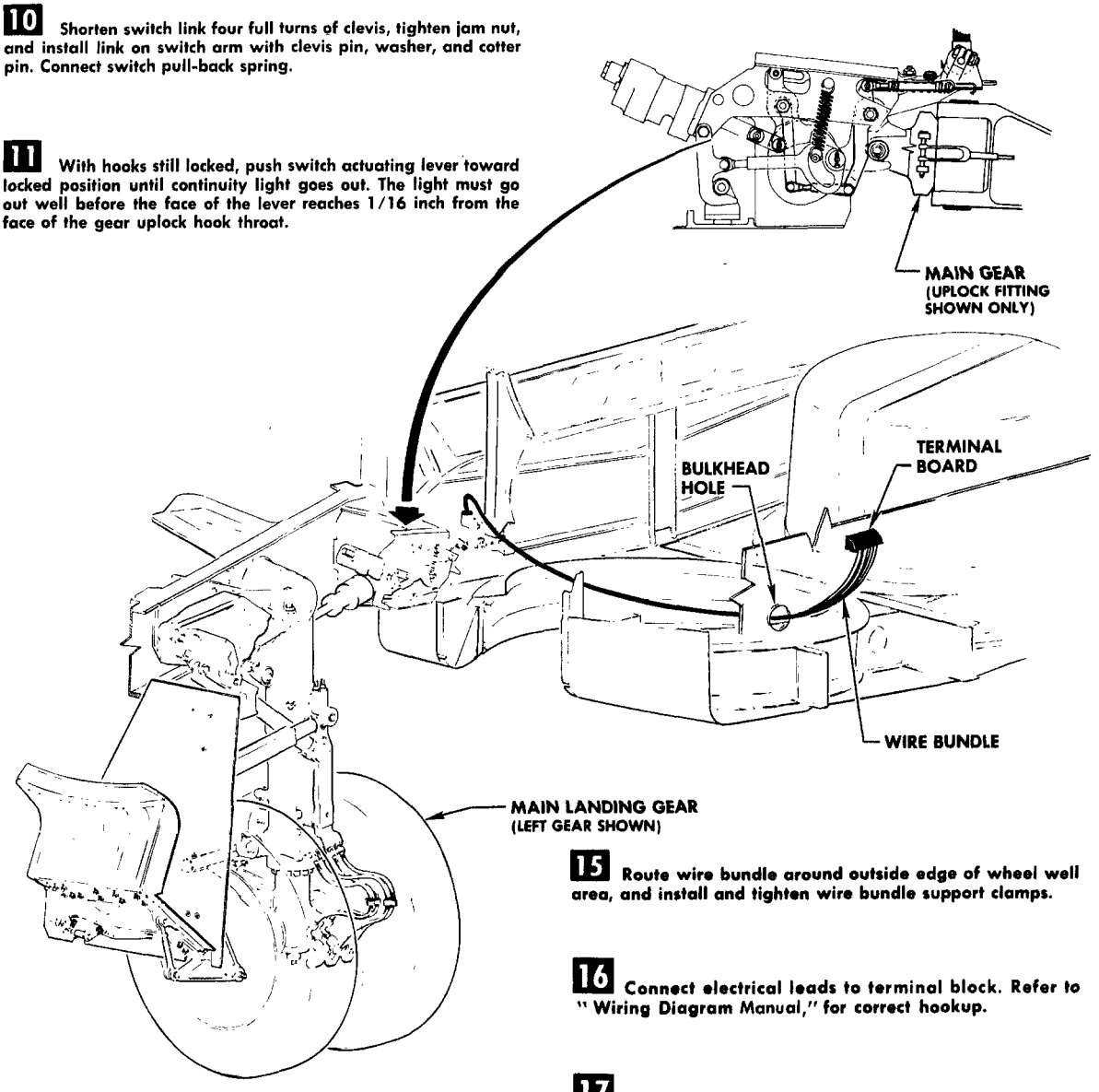
9 Looking at switch from shaft end, rotate switch shaft (clockwise for right switch, counterclockwise for left switch) until continuity light comes on. Continue to rotate switch shaft in same direction until light just goes out. Then tighten locking bolt, securing switch arm to switch shaft.

5370-2-33-55B

Installing and Adjusting Main Gear Uplock Switch
Figure 32-60-403 (Sheet 1 of 2)

10 Shorten switch link four full turns of clevis, tighten jam nut, and install link on switch arm with clevis pin, washer, and cotter pin. Connect switch pull-back spring.

11 With hooks still locked, push switch actuating lever toward locked position until continuity light goes out. The light must go out well before the face of the lever reaches 1/16 inch from the face of the gear uplock hook throat.



12 Connect uplock actuating cylinders to linkage and safety. Check that all other linkage in the area is safetied.

13 Open uplock hooks by pushing forward on emergency gear release lever inboard of uplock switch and remove continuity light.

14 Run electrical leads into wire bundle and install protective cover over bundle.

15 Route wire bundle around outside edge of wheel well area, and install and tighten wire bundle support clamps.

16 Connect electrical leads to terminal block. Refer to "Wiring Diagram Manual," for correct hookup.

17 Manually lower landing gear.

18 Install wheel and strut well close-out panels.

19 Check operation of landing gear retraction and operation of landing gear warning light.

S370-2-33-56

Installing and Adjusting Main Gear Uplock Switch
Figure 32-60-403 (Sheet 2 of 2)

The following equipment is required for this procedure:

- Jacking and mooring kit
- Continuity light
- Set of jacks
- Hydraulic test stand capable of delivering 4 gpm at 3000 psi
- 28-volt dc external electrical power

NOTE The nose gear downlock switch is equipped with an electrical lead (pigtail) for each switch contact. Although all electrical leads from the switch are not connected into the landing gear circuit, they must be correctly connected or properly stowed to prevent possibility of electrical shorts. The electrical leads are routed in a wire bundle (harness) to an electrical disconnect plug above the strut.

1. Position switch on switch bracket with electrical leads aft of, and parallel with, the drag brace.

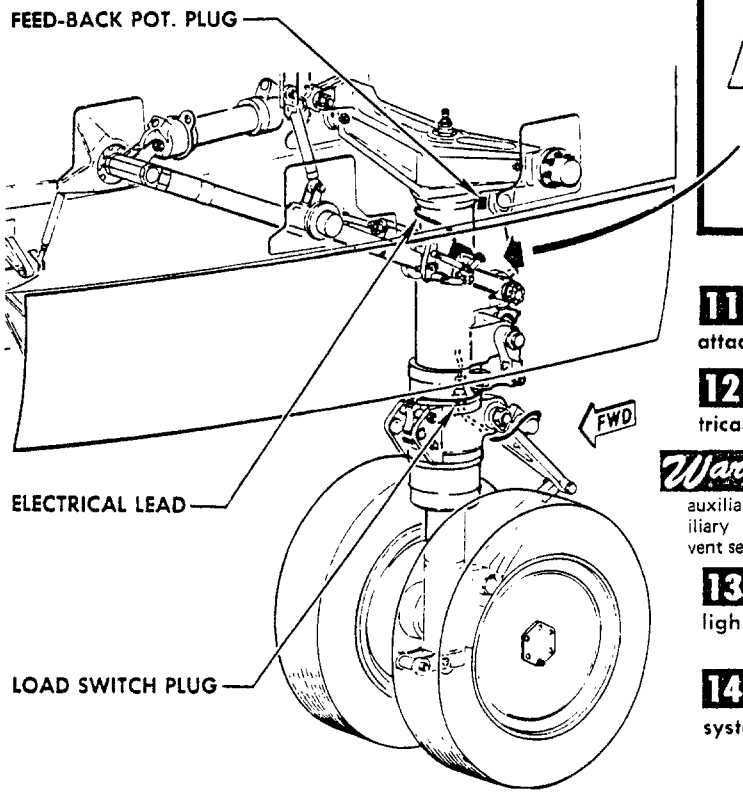
2. Install, tighten, and safety jam nut securing switch to bracket with lockwire.

3. Connect and safety steering feed-back potentiometer and nose gear load switch electrical disconnect plugs with lockwire.

4. Tape and stow unused electrical leads from switch in wire bundle.

5. Install clamps securing electrical leads to strut and drag brace.

6. Connect continuity light to pins "E" and "J" of disconnected electrical plug near top of strut.

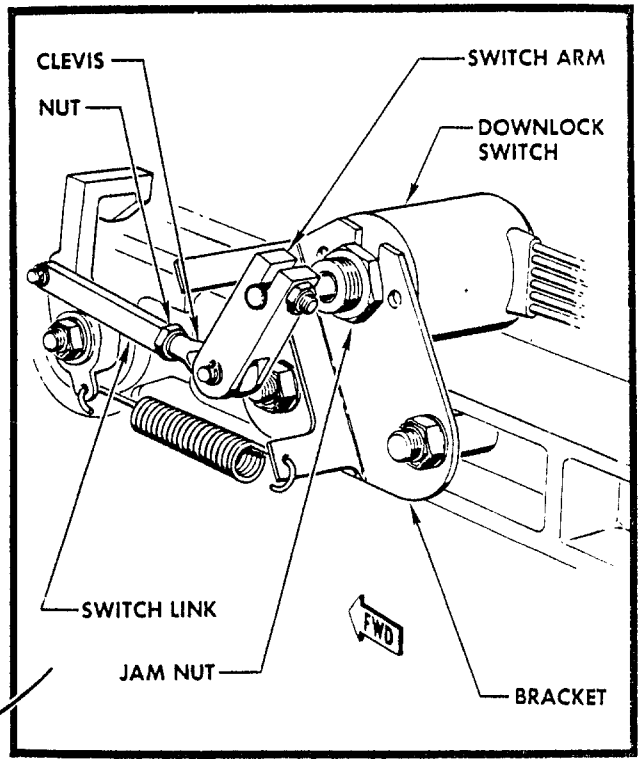


7. Adjust switch adjusting link to 3-3/16 inches eye to eye.

8. Place switch arm on switch shaft, with arm pointing down and slightly forward. Do not tighten locking bolt. Attach arm to switch adjusting link.

9. Looking at switch from shaft end, rotate switch shaft within switch arm counterclockwise until continuity light goes out. Continue to rotate shaft until continuity light comes on. At this point, tighten switch arm locking bolt, securing arm to switch shaft.

10. With nose gear down and locked, lengthen switch adjusting link two turns, and recheck for continuity.



11. Tighten switch link jam nut and safety switch link attach pin.

12. Remove continuity light, and connect and safety electrical disconnect plug with lockwire.

Warning Before applying landing gear hydraulic pressure for any reason (checks, adjustments, etc.), deplete normal and auxiliary accumulator hydraulic pressures. This will prevent the auxiliary accumulator power assist retraction system to operate and prevent serious injury to personnel and/or damage airplane.

13. Check operation of landing gear warning horn and light and nose gear position indicating light.

14. Check functional operation of nose wheel steering system.

Installing and Adjusting Nose Gear Downlock Switch
Figure 32-60-404

6. Nose Gear Uplock Switch Replacement (Refer to figure 32-60-405.)

A. Nose Gear Uplock Switch Removal

WARNINGS:
- All accumulated hydraulic pressure must be depleted to prevent gear retraction that may cause serious injury to personnel and/or airplane damage.
- Pin NLG when working in the NLG wheel well to prevent inadvertent gear retraction that may cause serious injury to personnel.

(1) Remove all electrical and hydraulic power.

(2) Pull LANDING GEAR POS, LANDING GEAR CONT, HYD PUMP CONT, and HYD AUX SYS circuit breakers.

(3) Remove clamps and protective covering around wires.

NOTE: Wire leads of nose gear uplock must be tagged for splicing in replacement switch. Refer to Sabreliner Wiring Diagram Manual, Report No. NA-69-421 (70-Model) or SR-73-26 (80 Model) for wires to be cut.

(4) Cut and tag wire leads of nose gear uplock switch.

NOTE: Note position of switch linkage for reinstallation.

(5) Disconnect switch arm spring. (Refer to figure 32-60-405.)

(6) Disconnect nose gear uplock switch arm.

(7) Remove nose gear uplock switch jam nut.

(8) Remove nose gear uplock switch.

B. Nose Gear Uplock Switch Installation and Adjustment (Refer to figure 32-60-405.)

Equipment/Tools

Set of Jacks

Jacking Pad and Mooring Kit

28 Vdc External Power Source

Continuity Light

Hydraulic Test Stand Capable of Delivering 4 Gpm at 3000 Psi.

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WARNINGS: • All accumulated hydraulic pressure must be depleted to prevent gear retraction that may cause serious injury to personnel and/or airplane damage.

• Pin NLG when working in the NLG wheel well to prevent inadvertent gear retraction that may cause serious injury to personnel.

(1) Pull HYD PUMP CONT, HYD AUX SYS, LANDING GEAR POS, and LANDING GEAR CONT circuit breakers.

(2) Deplete normal and auxiliary accumulator hydraulic pressures.

NOTES: • Do not change adjustment of nose wheel door rod assembly.

• Disconnecting the door rod assembly is necessary to prevent the doors from closing after gear retraction.

(3) Disconnect nose gear wheel door rod assemblies at NLG wheel door actuating arms. Secure rod assemblies to door.

NOTE: The nose gear uplock switch it equipped with an electrical lead for each switch contact. All electrical leads from the switch are not connected into the landing gear circuit, each lead must be either correctly connected or properly stowed to prevent possible electrical shorts.

(4) Check nose gear uplock switch for one jam nut and one lockwasher installed prior to mounting on nose gear.

(5) Install uplock switch on switch mounting bracket with electrical leads fwd.

(6) Check index pins are seated in index holes in bracket.

(7) Install and tighten jam nut, securing switch to bracket, and safetywire nut.

(8) Connect continuity light to wire D47WW and D48AA of switch for check in step 13.

NOTE: Reconfigure nose uplock switch linkage to position prior to disconnecting as noted in removal of uplock switch.

(9) Adjust switch link to 2-13/16 in., eye-to-eye, and attach to switch arm. Do not tighten switch arm to switch shaft at this time.

(10) Line up edge of actuating arm with face of hook throat, insert clevis pin, and hold in position for steps 11 and 12.

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(11) Rotate switch shaft cw within switch arm until continuity light just goes out.

(12) Tighten switch arm to switch shaft.

(13) Operate switch linkage assembly to check continuity light just goes out when edge of actuating arm is moved fwd into alignment with hook throat.

(14) Shorten switch link so that actuating arm is 3/16 in. aft of hook throat when switch contact makes or breaks. Tighten switch link jam nut.

(15) Install washer and cotter pin on switch link clevis pin.

(16) Connect spring to switch arm.

(17) Disconnect continuity light.

(18) Splice switch electrical leads to wires indicated per Sabreliner Wiring Diagram Manual, Report No. NA-69-421 (70 Model) or SR-73-26 (80 Model).

(19) Tape and stow electrical leads not connected into landing gear circuit in wire bundle per wiring diagram manual.

(20) Install wire bundle protective cover.

(21) Install and tighten wire bundle clamps.

(22) Push HYD PUMP CONT, HYD AUX SYS, LANDING GEAR POS, and LANDING GEAR CONT circuit breakers.

(23) Apply electrical and hydraulic power.

(24) Check operation of landing gear.

(25) Disconnect electrical and hydraulic power.

(26) Deplete all accumulated hydraulic pressure.

(27) Connect nose gear wheel door actuating arms.

(28) Apply electrical and hydraulic power.

(29) Check operation of landing gear.

(30) Secure airplane as desired.

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Equipment/Tools

- Set of Jacks
- Jacking and Mooring Kit
- 28 Vdc External Electrical Power
- Continuity Light
- Hydraulic Test Stand Capable of Delivering 4 Gpm at 3000 Psi

WARNING

Before applying landing gear hydraulic pressure for any reason (checks, adjustments, etc.), deplete normal and auxiliary accumulator hydraulic pressures. This will prevent the auxiliary accumulator power assist retraction system to operate and prevent serious injury to personnel and/or damage airplane.

NOTE

The nose gear uplock switch is equipped with an electrical lead for each switch contact. Although all electrical leads from the switch are not connected into the landing gear circuit, they must be correctly connected or properly stowed to prevent possible electrical shorts.

1 Before installing switch, make sure that there is one jam nut and one lock washer on switch. Connect continuity light to terminals (wires D47WW and D48AA) of switch.

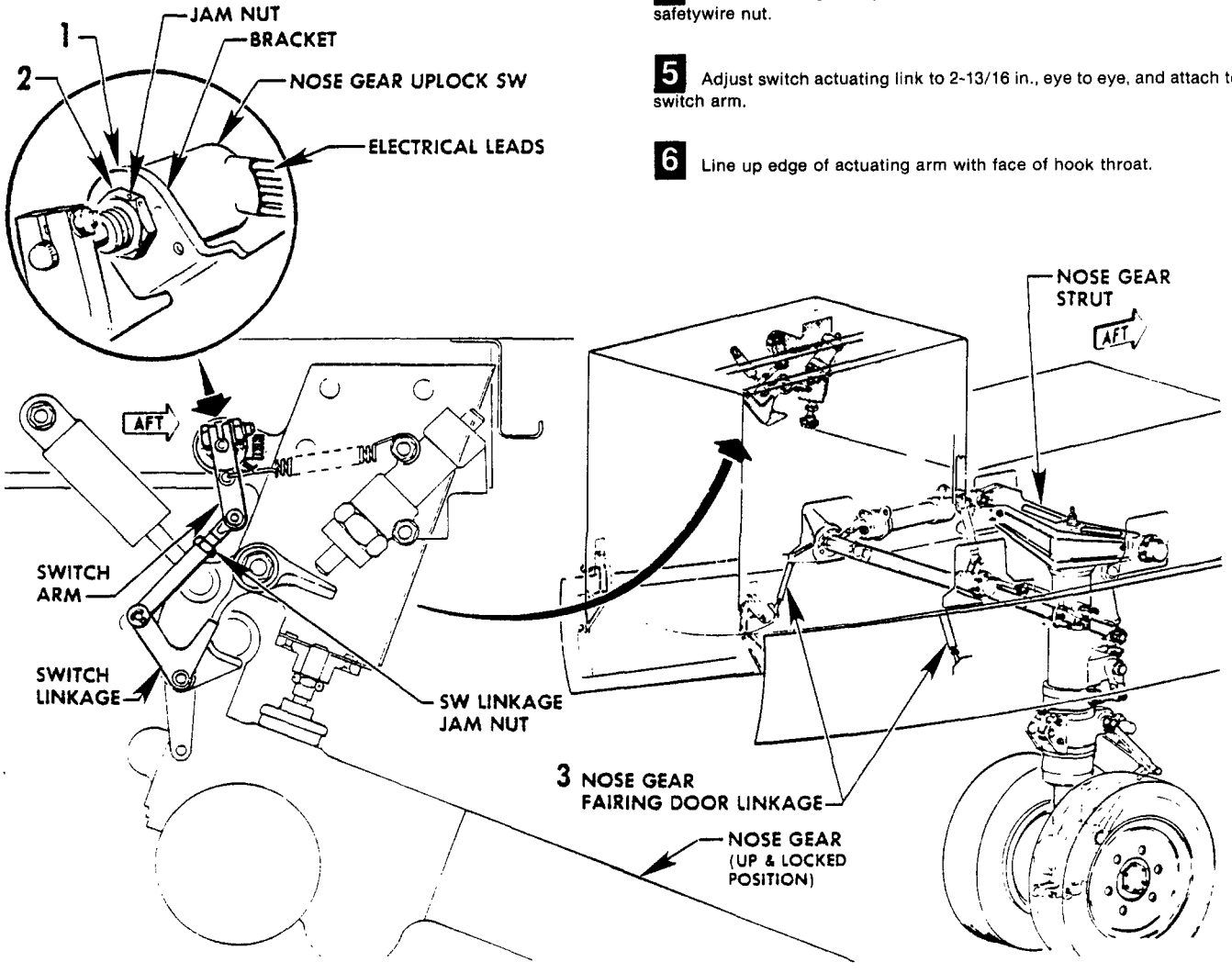
2 Position switch on switch mounting bracket with electrical leads fwd. Make sure that index pins are seated in index holes in bracket.

3 Disconnect nose gear wheel fairing door linkage at fairing door, being careful not to change adjustment of linkage. Disconnecting the door linkage is necessary to prevent the door closing when the nose gear is retracted. Disconnect uplock switch linkage.

4 Install and tighten jam nut, securing switch to bracket, and safetywire nut.

5 Adjust switch actuating link to 2-13/16 in., eye to eye, and attach to switch arm.

6 Line up edge of actuating arm with face of hook throat.



Installing and Adjusting Nose Gear Uplock Switch
Figure 32-60-405 (Sheet 1 of 2)

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7 Looking at switch shaft end, rotate switch shaft within switch arm cw until continuity light just goes out. At this point, tighten switch arm onto switch shaft. Operate linkage to make sure that continuity light just goes out when edge of actuating arm is moved fwd into alignment with throat of hook.

8 Shorten switch link so that actuating arm is 3/16 in. aft of hook throat when switch contact makes or breaks. Install clevis, washer, and cotter pin.

9 Connect spring to switch arm.

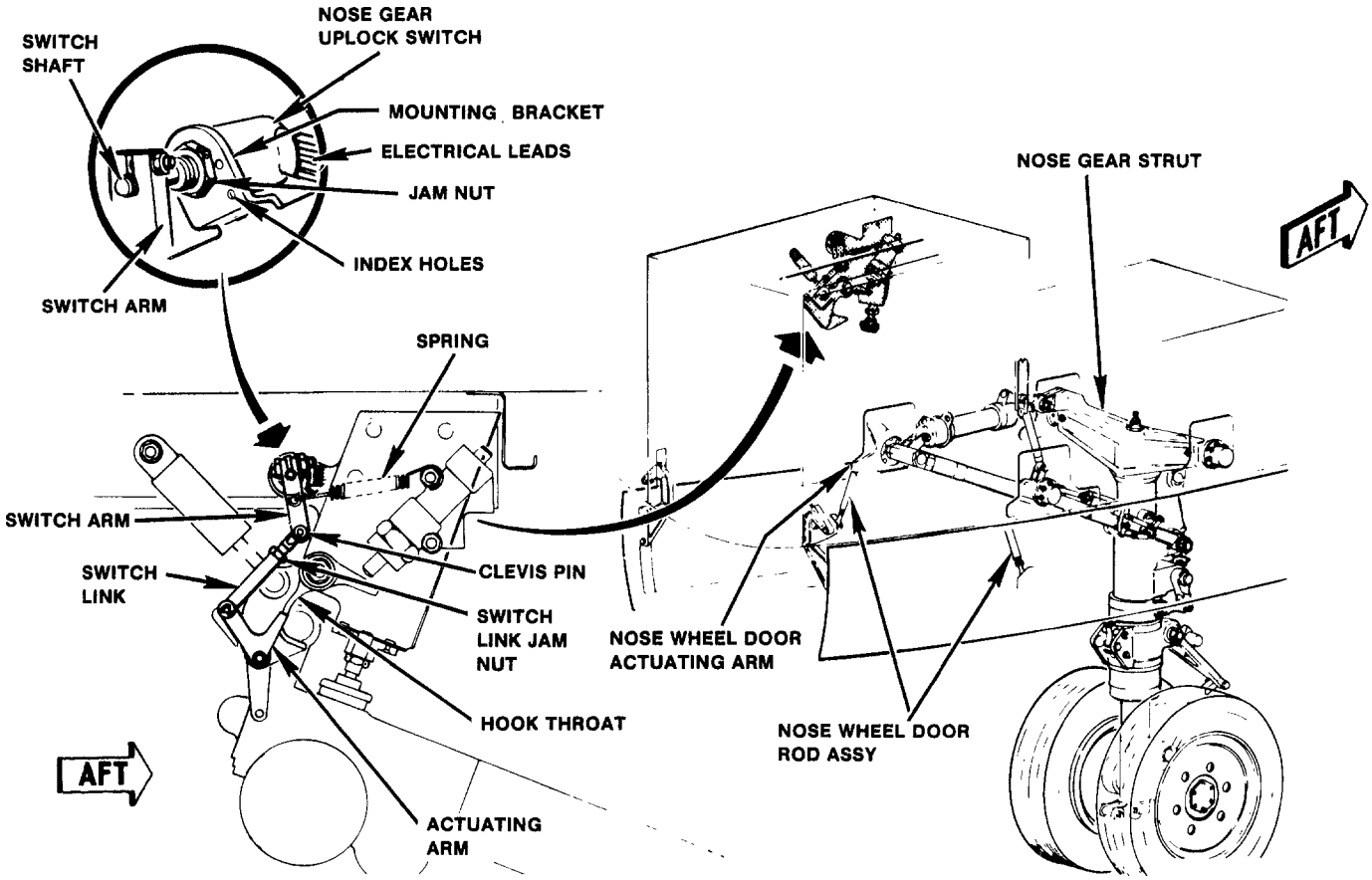
10 Tape ends of switch electrical leads not connected into landing gear circuit. Stow taped electrical leads in wire bundle.

11 Install protective cover over wire bundle, and install and tighten wire bundle clamps.

12 Connect switch electrical leads to terminal strip above nose gear strut.

13 Connect nose gear wheel fairing door linkage.

14 Check operation of landing gear.



MAJOR CHANGE

Installing and Adjusting Nose Gear Uplock Switch
Figure 32-60-405 (Sheet 2 of 2)

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2. Checking Operation of Landing Gear Position Indicating and Warning System

A. The following equipment is needed for this check:

NOTE: The following checks are performed using a hydraulic test stand. If the airplane hydraulic pump is to be used, dump the accumulator pressure before retracting gear. With the hydraulic accumulators dumped, and by cycling the hydraulic pump control switch, this allows a slow retraction of the landing gear.

- Hydraulic test stand capable of delivering 4 gpm at 3000 psi (if available)
- 28-volt dc external electrical power
- MB-1 pitot-static field tester
- Two HS-78 headsets
- Jacking and mooring kit
- Set of jacks
- Air source for inflatable wheel seal - NA370 Airplanes not changed by Service Bulletin 72-8

B. The following circuit breakers must be pushed in:

- "LANDING GEAR - CONT"
- "LANDING GEAR - POS"

CAUTION: TO PREVENT POSSIBLE DAMAGE TO LANDING GEAR AUDIBLE WARNING SIGNAL GENERATOR, DO NOT ALLOW GENERATOR TO OPERATE MORE THAN 2 MINUTES. MOVE THROTTLES TO 95% RPM OR MOMENTARILY PUSH IN LANDING GEAR WARNING HORN CUTOUT BUTTON WHEN AUDIBLE WARNING IS NOT NEEDED. BEFORE RESUMING OPERATION, ALLOW GENERATOR TO COOL A MINIMUM OF 8 MINUTES.

- TO AVOID DAMAGING AIRSPEED INDICATOR, DO NOT APPLY ANY POSITIVE PRESSURE TO STATIC PORT OR OVERPRESSURIZE PITOT PORT. TO PREVENT DAMAGING INDICATORS IN COCKPIT, DO NOT INCREASE OR DECREASE SIMULATED ALTITUDE FASTER THAN 6000 FEET PER MINUTE.
- IF THE AIRCRAFT HYDRAULIC PUMP IS USED, ALLOW SUFFICIENT COOLING TIME BETWEEN CONSECUTIVE RETRACTION CYCLES TO PREVENT DAMAGE TO HYDRAULIC PUMP MOTOR.

C. Perform check as follows:

(1) Check MB-1 field tester for evidence of leakage.

(2) Connect MB-1 field tester to pitot-static system at pitot head as follows:

(a) Connect pressure hose from MB-1 field tester pitot hose connection to pressure port on copilot's pitot tube.

(b) Using suitable fitting, connect vacuum hose from MB-1 field tester to static source (alternate static source on NA370-2 thru -6, -8 and -9).

(3) Connect headsets to pilot's and copilot's microphones and headset cords in cockpit.

(4) Position VHF-1 and VHF-2 switches, on VHF control panel, to PHONE.

(5) On NA370-2 thru -6, -8 and -9, place copilot's static pressure selector valve at ALTERNATE SOURCE.

(6) Perform steps of "General Procedure Before Landing Gear Operation." (Refer to 32-00-3).

(7) Start hydraulic test stand and regulate test stand to 4 gpm at 3000 psi. Reduce hydraulic pressure to zero.

(8) Move landing gear control handle up. The red light in the landing gear control handle should come on regardless of throttle position, indicating that the landing gear is not in the same position as the landing gear control handle.

(9) Increase hydraulic test stand pressure and slowly retract landing gear. As each gear downlock pin is pulled (retracted), the green light in the corresponding gear position indicator in the cockpit will go out, indicating that the gear is not down and locked.

NOTE: Because hydraulic pressure is directed at the same time to all gear downlock pins, the downlock pins will normally be retracted (pulled) at about the same time.

(10) With either or both throttles retarded (less than 95% rpm), the audible warning signal should be heard in each headset and the landing gear warning horn should sound when the first gear downlock pin was retracted (pulled) in step (9). The landing gear audible warning sound and signals may continue until an airspeed above 165 knots and/or analtitude of 13,500 feet is reached.

NOTE: Normally, the landing gear uadible warning signal and sound will not be heard during takeoff because the throttles will be above 95% rpm.

(11) If possible, regulate hydraulic test stand to close and lock one landing gear at a time. When the nose gear is up and locked, the red light in the landing gear control handle should go out, indicating that all gears are up and locked.

(12) Reduce hydraulic test stand pressure to zero. Simulate a landing preparation with airspeed below 145* or 135† knots and altitude below 11,500 feet and by moving both throttles below 95% rpm. The audible warning signal should be heard from the headsets and the landing gear warning horn should sound. The red light in the landing gear control handle should come on in a flashing manner, indicating that the landing gear control handle is not down.

(13) Momentarily depress landing gear warning horn cutout button. The sound from thellanding gear warning horn and in the headsets should stop. The red light in the landing gear control handle should remain on.

(14) Check operation of noncancellable warning horn control as follows:

(a) Lower wing flaps to full down position. Check that warning horn sounds and audible warning signals are heard when flaps are lowered past 20 degrees. Check that red light in landing gear control handle continues to flash.

(b) Depress landing gear warning horn cutout button. Check that warning horn continues to sound and that audible warning signals are still heard. Check that red light in landing gear control handle continues to flash. Place wing flap switch at UP and allow flaps to retract.

(15) Momentarily move both throttles above 95% rpm; then retard both throttles below 95% rpm. The audible warning signal should be heard from the headsets, and the landing gear warning horn should sound. The red light in the landing gear control handle should again come on in a flashing manner. Move both throttles above 95% rpm; the audible warning signal and the warning horn sounds should cease. The red light in the gear control handle should go out. Check each throttle individually. Results should be the same.

(16) Check operation of altitude switch in the airspeed and altitude warning unit as follows:

(a) Move both throttles below 95% rpm. The audible warning signal and the warning horn sound should come on. The red light in the control handle should come on in a flashing manner.

(b) Using MB-1 field tester, simulate an increase in altitude. The audible warning signals should stop, the warning horn sounds should stop and the red light in the landing gear control handle should go out by 13,600 feet on the altimeter in the cockpit.

(c) Slowly decrease altitude. The audible warning signal should be heard, the warning horn should sound, and the red light should come on in a flashing manner when the simulated altitude 12,000 (± 600) feet on the altimeter in the cockpit.

* NA370-1, -7 and 380-1 and later
† NA370-2 thru -6, -8 and -9

(17) Check operation of airspeed switch in airspeed and altitude warning unit as follows:

(a) Using MB-1 field tester, regulate tester to maintain a simulated altitude below 11,000 feet, while increasing a simulated airspeed above 170* or 160† knots. The audible warning signals should stop, the warning horn sounds should stop and the red light in the landing gear control handle should go out by 170* or 160† knots on the airspeed indicator in the cockpit.

(b) Slowly decrease simulated airspeed. When the simulated airspeed is decreased below 150 (±10)* or 140 (±10)† knots, the audible signal from the headsets should be heard and the landing gear warning horn should sound. The red light in the gear control handle should come on in a flashing manner.

(18) Move landing gear control handle down; the audible warning signal and the warning horn sound should continue. The red light in the landing gear control handle should change from a flashing red light to a steady red light.

(19) Increase hydraulic pressure and extend landing gear. The audible warning signals should be heard from the headsets and the warning horn sound should continue until all gear are down and locked. The red light in the landing gear control handle should stay on until the last gear is down and locked. As each gear reaches the down-and locked position, the green light in the corresponding gear position indicator in the cockpit will come on, indicating that the gear is down and locked.

(20) Shut down hydraulic test stand or position HYD PUMP control switch to OFF. Disconnect MB-1 field tester and disconnect headsets.

(21) Ensure that landing gear is down and locked, then lower airplane and remove jacks.

* NA370-1, -7 and 380-1 and later
NA370-2 thru -6, -8 and -9

CHAPTER 33

LIGHTS

CHAPTER 33

LIGHTS

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CHAPTER 33

LIGHTS

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*The asterisk indicates pages revised, added, or deleted by the current revision.

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LIGHTS - DESCRIPTION

1. General

The airplane interior lighting is limited to a caution-warning and indicating light system, cockpit lighting, and emergency lighting. The airplane is provided with wiring to the circuit-breaker panels for installation of lights in the cabin when cabin equipment is installed.

The airplane is equipped with exterior lighting as follows:

- Two flush-mounted extendable landing and taxi lights, on a separate panel on the bottom of the forward fuselage
- Two wing ice check light assemblies, on each side of the fuselage
- Two anticollision lights, one on the tip of the vertical stabilizer and the other on the bottom of the fuselage, forward of the speed brake
- Wing position lights consisting of a red light mounted on the left wing tip, a green light mounted in the right wing tip, and two white lights mounted on the aft end of the fuselage
- On airplanes NA-380-38 and later, three strobe lights are installed. One strobe light on each wing tip (integral part of the position light assembly) and one strobe light located on top of tail cone.
- Four emergency lights, one in cockpit area, and three in cabin area (entrance area, one for LH emergency exit area and one for RH emergency exit area). The emergency system has two 7.2 volt rechargeable batteries in parallel.

FLIGHT COMPARTMENT - DESCRIPTION AND OPERATION

1. General

The flight compartment lighting system provides floodlights for illuminating the right and left instrument panel, pilot's and copilot's consoles, and center pedestal. Over-all illumination of the cockpit is provided by a dome light. In addition to floodlights, the instrument panels, side consoles, overhead panels, and pedestal are illuminated by indirect lighting. Each panel transmits the lighting edgewise throughout the face of the panel to illuminate the panel markings. A thunderstorm light system is provided to reduce the blinding effects of lightning flashes. Included in the flight compartment lighting is the caution-warning and indicating light system. (See figures 33-10-1, 33-10-2, and 33-10-3.)

2. Instrument Panel, Console, Overhead, and Pedestal Floodlights

The flight instrument panel floodlight system lights the left and right instrument panels. The floodlights are on the underside of the instrument panel shroud. They are controlled by the "FLIGHT INSTRUMENT LIGHTS - FLOOD" knobs on the left and right electrical interior and console lighting control panels on the left and right consoles. The left console control panel "FLIGHT INSTRUMENT LIGHTS - FLOOD" knob controls the instrument panel floodlighting for the left side of the instrument panel. The "FLIGHT INSTRUMENT LIGHTS - FLOOD" knob on the right console controls lighting for the right side of instrument panel. Power for the flight instrument floodlights comes from the essential 28-volt dc bus.

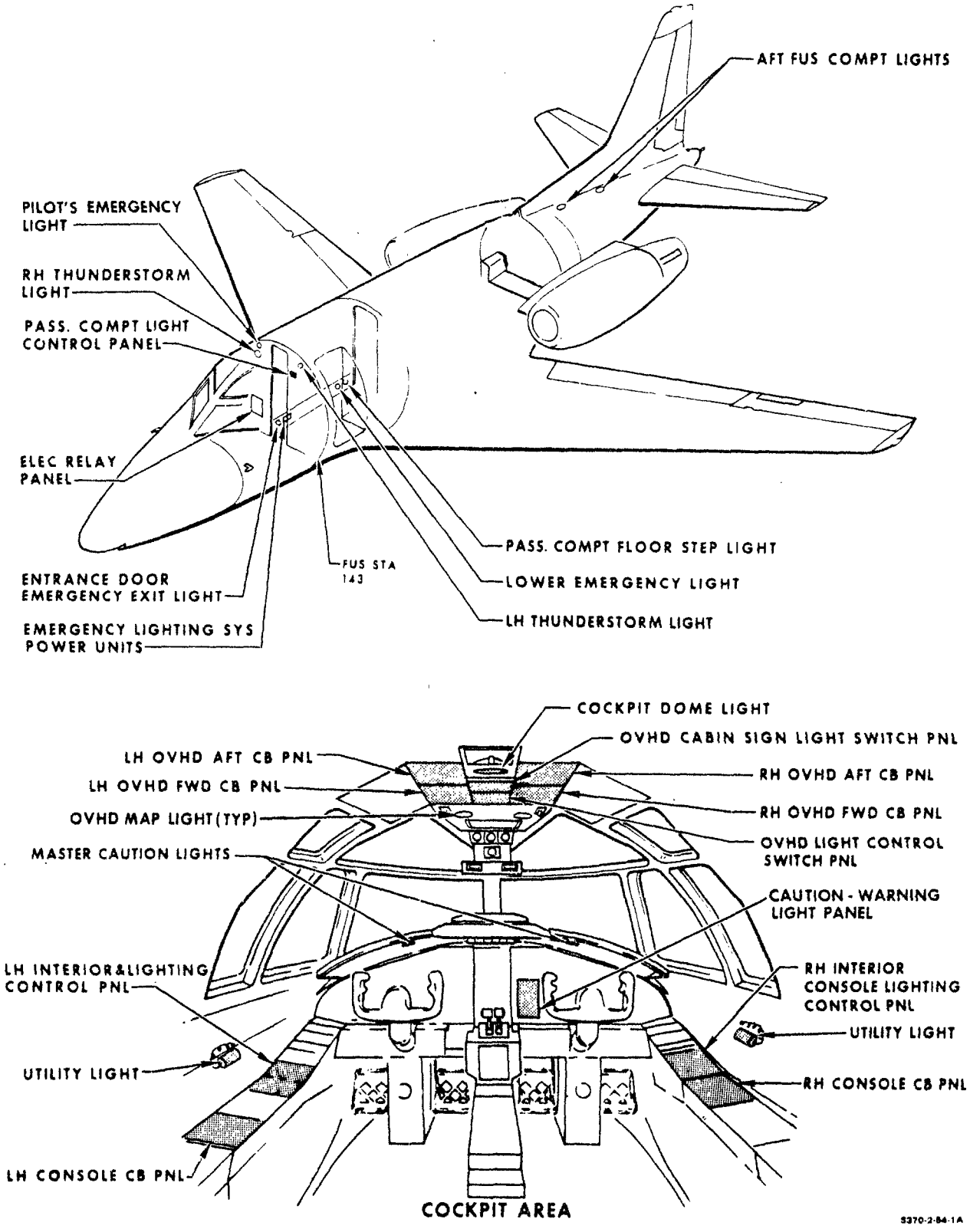
The radio and engine instrument panel floodlight system lights the radio control panels and the engine instruments on the instrument panel. The floodlights are on the underside of the instrument panel shroud. They are controlled by the "RAD & ENG INST - FLOOD" knob on the left console lighting control panel. Power for the radio and engine instrument panel floodlights comes from the essential 28-volt dc bus.

The cockpit console floodlight system provides lighting for the left and right consoles. The floodlights are over the consoles on the frame of the airplane. The "CONSOLE LIGHTS - FLOOD" knob on the left console controls the floodlights over the left console. The "CONSOLE LIGHTS - FLOOD" knob on the right console controls the floodlights over the right console. Power for both the left and right console floodlights comes from the essential 28-volt dc bus.

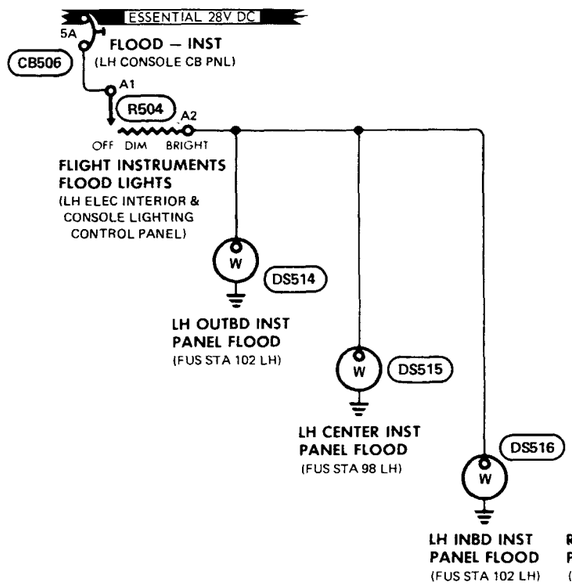
The pedestal floodlight system lights the center pedestal. The floodlights are on the underside of the instrument panel shroud. They are controlled by the "PED & OVHD - FLOOD" knob on the left console lighting control panel. Power for the pedestal floodlights comes from the essential 28-volt dc bus.

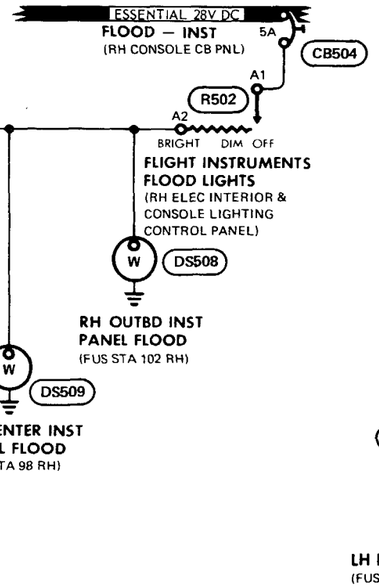
3. Dome Light

The cockpit dome lights provide overhead lighting for the cockpit area. The dome light aft of the center overhead control panel contains a red and white light. This dome light is controlled by "LIGHTS - COCKPIT DOME" switch.

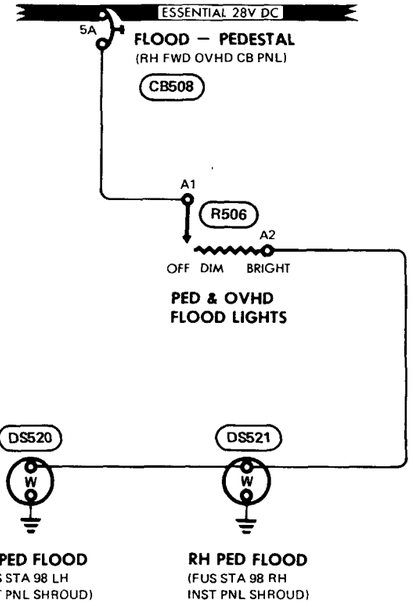


Flight Compartment Lighting Equipment Location
Figure 33-10-1

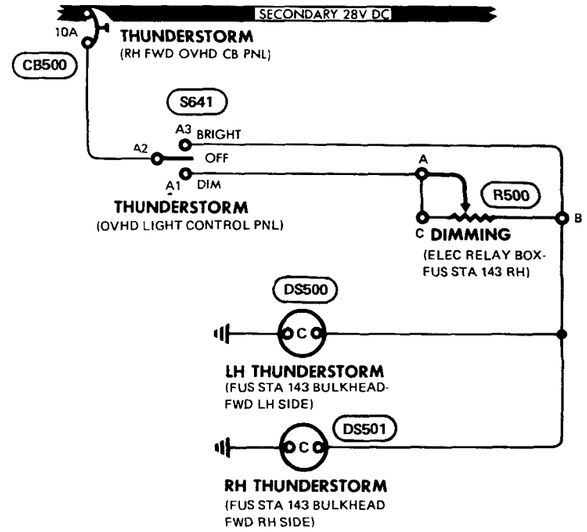




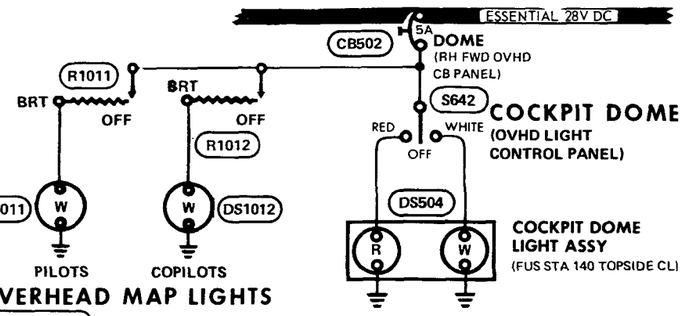
COCKPIT INSTRUMENT PANEL FLOODLIGHT



PEDESTAL FLOODLIGHT

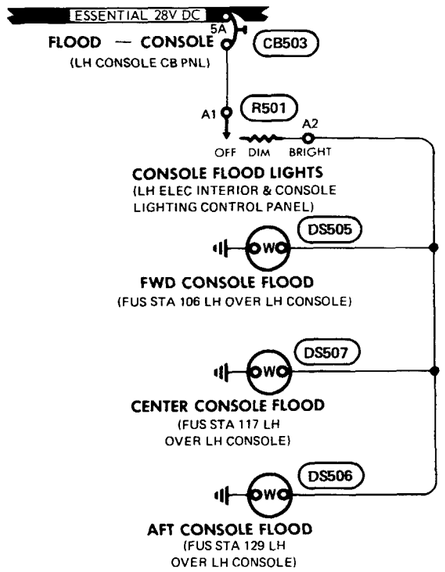


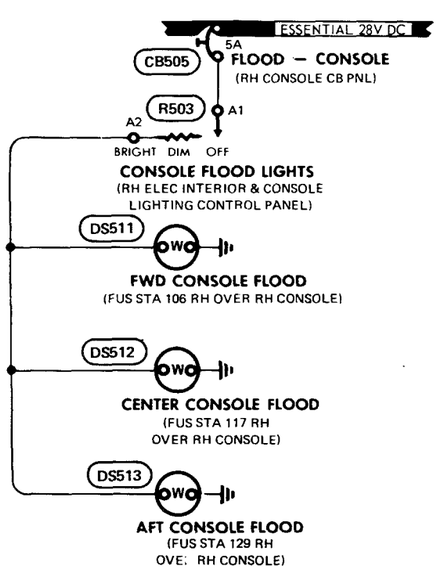
THUNDERSTORM LIGHTS



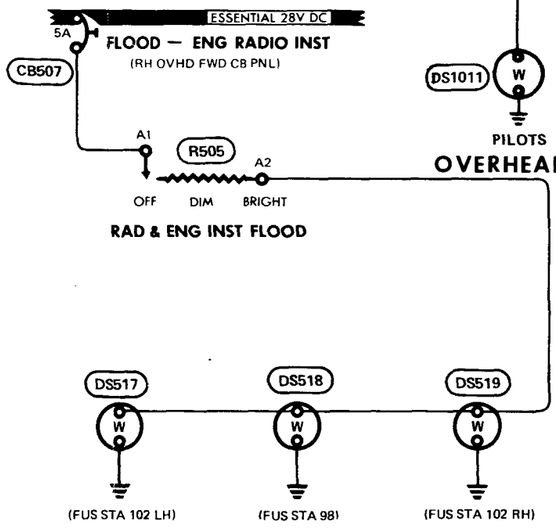
OVERHEAD MAP LIGHTS

COCKPIT DOME AND MAP LIGHTS

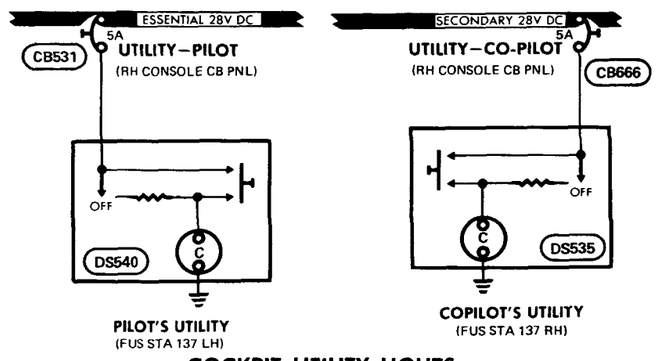




COCKPIT CONSOLE FLOODLIGHTS



RADIO AND ENGINE INSTRUMENT FLOODLIGHTS



COCKPIT UTILITY LIGHTS

S370-2-54-2B

Figure 33-10-2 — Flight Compartment Lighting Schematic (Sheet 1 of 3)

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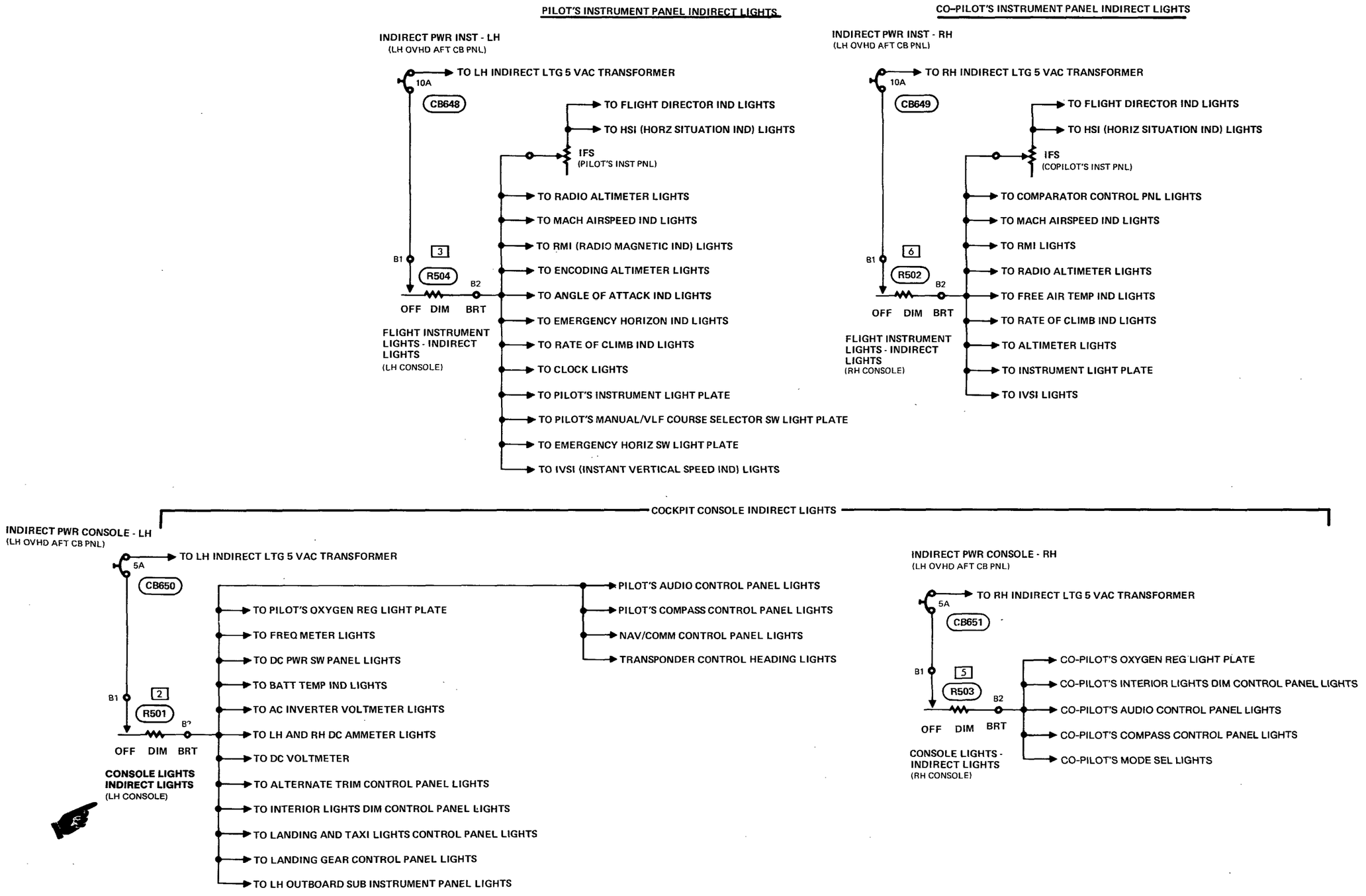

Figure 33-10-2 — Flight Compartment Lighting Schematic (Sheet 2 of 3)

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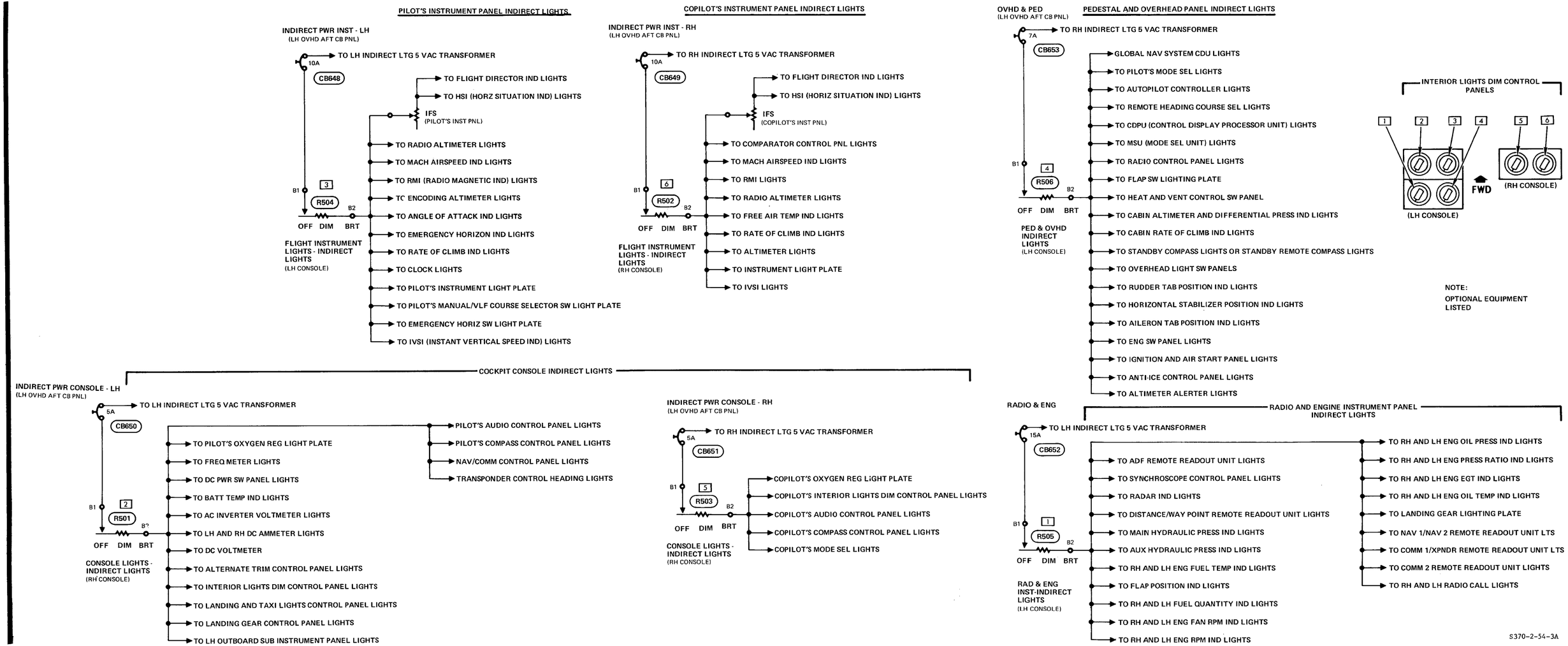

PILOT'S INSTRUMENT PANEL INDIRECT LIGHTS
INDIRECT PWR INST - LH
(LH OVHD AFT CB PNL)
10A
CB648
TO LH INDIRECT LTG 5 VAC TRANSFORMER
TO FLIGHT DIRECTOR IND LIGHTS
TO HSI (HORZ SITUATION IND) LIGHTS
IFS
(PILOT'S INST PNL)
TO RADIO ALTIMETER LIGHTS
TO MACH AIRSPEED IND LIGHTS
TO RMI (RADIO MAGNETIC IND) LIGHTS
TO ENCODING ALTIMETER LIGHTS
TO ANGLE OF ATTACK IND LIGHTS
TO EMERGENCY HORIZON IND LIGHTS
TO RATE OF CLIMB IND LIGHTS
TO CLOCK LIGHTS
TO PILOT'S INSTRUMENT LIGHT PLATE
TO PILOT'S MANUAL/VLF COURSE SELECTOR SW LIGHT PLATE
TO EMERGENCY HORIZ SW LIGHT PLATE
TO IVSI (INSTANT VERTICAL SPEED IND) LIGHTS
B1
3
R504
B2
OFF DIM BRT
FLIGHT INSTRUMENT LIGHTS - INDIRECT LIGHTS
(LH CONSOLE)
COPILOT'S INSTRUMENT PANEL INDIRECT LIGHTS
INDIRECT PWR INST - RH
(LH OVHD AFT CB PNL)
10A
CB649
TO RH INDIRECT LTG 5 VAC TRANSFORMER
TO FLIGHT DIRECTOR IND LIGHTS
TO HSI (HORIZ SITUATION IND) LIGHTS
IFS
(COPILOT'S INST PNL)
TO COMPARATOR CONTROL PNL LIGHTS
TO MACH AIRSPEED IND LIGHTS
TO RMI LIGHTS
TO RADIO ALTIMETER LIGHTS
TO FREE AIR TEMP IND LIGHTS
TO RATE OF CLIMB IND LIGHTS
TO ALTIMETER LIGHTS
TO INSTRUMENT LIGHT PLATE
TO IVSI LIGHTS
B1
6
R502
B2
OFF DIM BRT
FLIGHT INSTRUMENT LIGHTS - INDIRECT LIGHTS
(RH CONSOLE)
OVHD & PED
(LH OVHD AFT CB PNL)
PEDESTAL AND OVERHEAD PANEL INDIRECT LIGHTS
7A
CB653
TO RH INDIRECT LTG 5 VAC TRANSFORMER
GLOBAL NAV SYSTEM CDU LIGHTS
TO PILOT'S MODE SEL LIGHTS
TO AUTOPILOT CONTROLLER LIGHTS
TO REMOTE HEADING COURSE SEL LIGHTS
TO CDPU (CONTROL DISPLAY PROCESSOR UNIT) LIGHTS
TO MSU (MODE SEL UNIT) LIGHTS
TO RADIO CONTROL PANEL LIGHTS
TO FLAP SW LIGHTING PLATE
TO HEAT AND VENT CONTROL SW PANEL
TO CABIN ALTIMETER AND DIFFERENTIAL PRESS IND LIGHTS
TO CABIN RATE OF CLIMB IND LIGHTS
TO STANDBY COMPASS LIGHTS OR STANDBY REMOTE COMPASS LIGHTS
TO OVERHEAD LIGHT SW PANELS
TO RUDDER TAB POSITION IND LIGHTS
TO HORIZONTAL STABILIZER POSITION IND LIGHTS
TO AILERON TAB POSITION IND LIGHTS
TO ENG SW PANEL LIGHTS
TO IGNITION AND AIR START PANEL LIGHTS
TO ANTI-ICE CONTROL PANEL LIGHTS
TO ALTIMETER ALERTER LIGHTS
B1
4
R506
B2
OFF DIM BRT
PED & OVHD INDIRECT LIGHTS
(LH CONSOLE)
INTERIOR LIGHTS DIM CONTROL PANELS
1
2
3
4
5
6
FWD
(LH CONSOLE)
(RH CONSOLE)
NOTE:
OPTIONAL EQUIPMENT LISTED
COCKPIT CONSOLE INDIRECT LIGHTS
INDIRECT PWR CONSOLE - LH
(LH OVHD AFT CB PNL)
5A
CB650
TO LH INDIRECT LTG 5 VAC TRANSFORMER
TO PILOT'S OXYGEN REG LIGHT PLATE
TO FREQ METER LIGHTS
TO DC PWR SW PANEL LIGHTS
TO BATT TEMP IND LIGHTS
TO AC INVERTER VOLTMETER LIGHTS
TO LH AND RH DC AMMETER LIGHTS
TO DC VOLTMETER
TO ALTERNATE TRIM CONTROL PANEL LIGHTS
TO INTERIOR LIGHTS DIM CONTROL PANEL LIGHTS
TO LANDING AND TAXI LIGHTS CONTROL PANEL LIGHTS
TO LANDING GEAR CONTROL PANEL LIGHTS
TO LH OUTBOARD SUB INSTRUMENT PANEL LIGHTS
PILOT'S AUDIO CONTROL PANEL LIGHTS
PILOT'S COMPASS CONTROL PANEL LIGHTS
NAV/COMM CONTROL PANEL LIGHTS
TRANSPONDER CONTROL HEADING LIGHTS
B1
2
R501
B2
OFF DIM BRT
CONSOLE LIGHTS - INDIRECT LIGHTS
(RH CONSOLE)
INDIRECT PWR CONSOLE - RH
(LH OVHD AFT CB PNL)
5A
CB651
TO RH INDIRECT LTG 5 VAC TRANSFORMER
COPILOT'S OXYGEN REG LIGHT PLATE
COPILOT'S INTERIOR LIGHTS DIM CONTROL PANEL LIGHTS
COPILOT'S AUDIO CONTROL PANEL LIGHTS
COPILOT'S COMPASS CONTROL PANEL LIGHTS
COPILOT'S MODE SEL LIGHTS
B1
5
R503
B2
OFF DIM BRT
CONSOLE LIGHTS - INDIRECT LIGHTS
(RH CONSOLE)
RADIO & ENG
RADIO AND ENGINE INSTRUMENT PANEL INDIRECT LIGHTS
15A
CB652
TO LH INDIRECT LTG 5 VAC TRANSFORMER
TO ADF REMOTE READOUT UNIT LIGHTS
TO SYNCHROSCOPE CONTROL PANEL LIGHTS
TO RADAR IND LIGHTS
TO DISTANCE/WAY POINT REMOTE READOUT UNIT LIGHTS
TO MAIN HYDRAULIC PRESS IND LIGHTS
TO AUX HYDRAULIC PRESS IND LIGHTS
TO RH AND LH ENG FUEL TEMP IND LIGHTS
TO FLAP POSITION IND LIGHTS
TO RH AND LH FUEL QUANTITY IND LIGHTS
TO RH AND LH ENG FAN RPM IND LIGHTS
TO RH AND LH ENG RPM IND LIGHTS
TO RH AND LH ENG OIL PRESS IND LIGHTS
TO RH AND LH ENG PRESS RATIO IND LIGHTS
TO RH AND LH ENG EGT IND LIGHTS
TO RH AND LH ENG OIL TEMP IND LIGHTS
TO LANDING GEAR LIGHTING PLATE
TO NAV 1/NAV 2 REMOTE READOUT UNIT LTS
TO COMM 1/XPNDR REMOTE READOUT UNIT LTS
TO COMM 2 REMOTE READOUT UNIT LIGHTS
TO RH AND LH RADIO CALL LIGHTS
B1
1
R505
B2
OFF DIM BRT
RAD & ENG INST-INDIRECT LIGHTS
(LH CONSOLE)
S370-2-54-3A

Flight Compartment Lighting Schematic
Figure 33-10-2 (Sheet 2 of 2)

Two additional overhead lights are located forward of the dome light, one over the pilot's position and the other over the copilot. Intensity controls are located adjacent to each light. Their primary purpose is to provide illumination for map reading, etc. Power for dome and map light illumination comes from the essential 28-volt dc bus.

4. Cockpit Utility Lights

The cockpit utility lights provide two movable light sources for the pilot and copilot to use when lighting is required to read maps, charts, etc. The light assemblies are normally stowed over the left and right consoles. Each light is controlled by a rheostat and momentary-type button switch which are a part of the light assemblies. Power for the pilot's utility light comes from the essential 28-volt dc bus. Power for the copilot's utility light comes from the secondary 28-volt dc bus.

5. Thunderstorm Lights

The thunderstorm lights provide bright or dimmed white lighting for the entire cockpit area. The lights are on the forward face of fuselage station 143 bulkhead, on the left and right sides. The lights are controlled by the "LIGHTS - THUNDERSTORM" switch on the overhead light control panel. Power for the thunderstorm lights comes from the secondary 28-volt dc bus.

6. Instrument Panel, Console, Overhead, and Pedestal Indirect Lighting

The flight instrument indirect lighting system illuminates the flight instruments on the left and right outboard instrument panels. The lights are inside the instrument cases. They are controlled by "FLIGHT INSTRUMENT LIGHTS - INDIRECT" knobs on the left and right electrical interior and console light panel. Power for the flight instrument indirect lighting system comes from the left and right indirect lighting 5-volt ac transformers on the left side of the fuselage at station 143.

The radio and engine instrument panel indirect light system illuminates the faces of the panels in the center portion of the instrument panel. The lights are embedded in the panels. Each panel transmits the light edgewise throughout the face of the panel to illuminate the panel markings. The lights are controlled by the "RAD & ENG INST - INDIRECT" knob on the center pedestal lighting control panel. Power for the operation of the radio and engine instrument panel comes from the left indirect lighting 5-volt ac transformer on the left side of the fuselage at station 143.

The console indirect lighting system illuminates the faces of the panels on the left and right consoles. The lights are embedded in the panels. Each panel transmits the lights edgewise through the face of the panel to illuminate the panel markings. The lights are controlled by the "CONSOLE LIGHTS - INDIRECT" knobs on the left and right electrical interior and console lighting control panels. Power for operation of the console lights comes from the left and right indirect lighting 5-volt ac transformers on the left side of the fuselage at station 143.

The pedestal and overhead indirect lighting system illuminates the faces of the control panels, switch panels, and radio control panels installed on the pedestal and overhead areas. The lights are embedded in the panels. Each

panel transmits the light edgewise throughout the face of the panel to illuminate the panel markings. The stand-by compass indirect light illuminates the inside of the compass. The lights are controlled by the "PED & OVHD - INDIRECT" knob on the left console lighting control panel and powered by the right-hand indirect lighting 5-volt ac transformer on the left side of the fuselage at station 143.

7. Caution-Warning and Indicating Light System

The caution-warning and indicating light system consists of caution-warning light panel, a control switch, four warning test relays, four dim test relays, and two master caution lights. The caution-warning light panel is on the copilot's inboard instrument panel. The master caution lights are recessed into the edge of the instrument panel shroud. The system indicates an individual airplane system trouble or failure by means of the indicator lights on the copilot's inboard instrument panel. When an indicator light comes on, circuitry within the panel also lights both master caution lights on the instrument panel shroud. Either master caution light can be manually pushed in, to reset (extinguish) both master caution lights. This reset feature permits the master caution lights to remain off until a subsequent trouble occurs. The individual system indicator lights on the indicator panel are not resettable. Once these lights come on, they remain on until the trouble is corrected. The "LIGHTS - CAUTION TEST" switch, on the overhead center light control panel, provides the means of testing all indicator lights on the panel (including the master caution lights) at the same time. The lights can be tested bright or dim according to the operation of the test switch. The lights will automatically reset to bright when the pilot's indirect instrument lights are turned off.

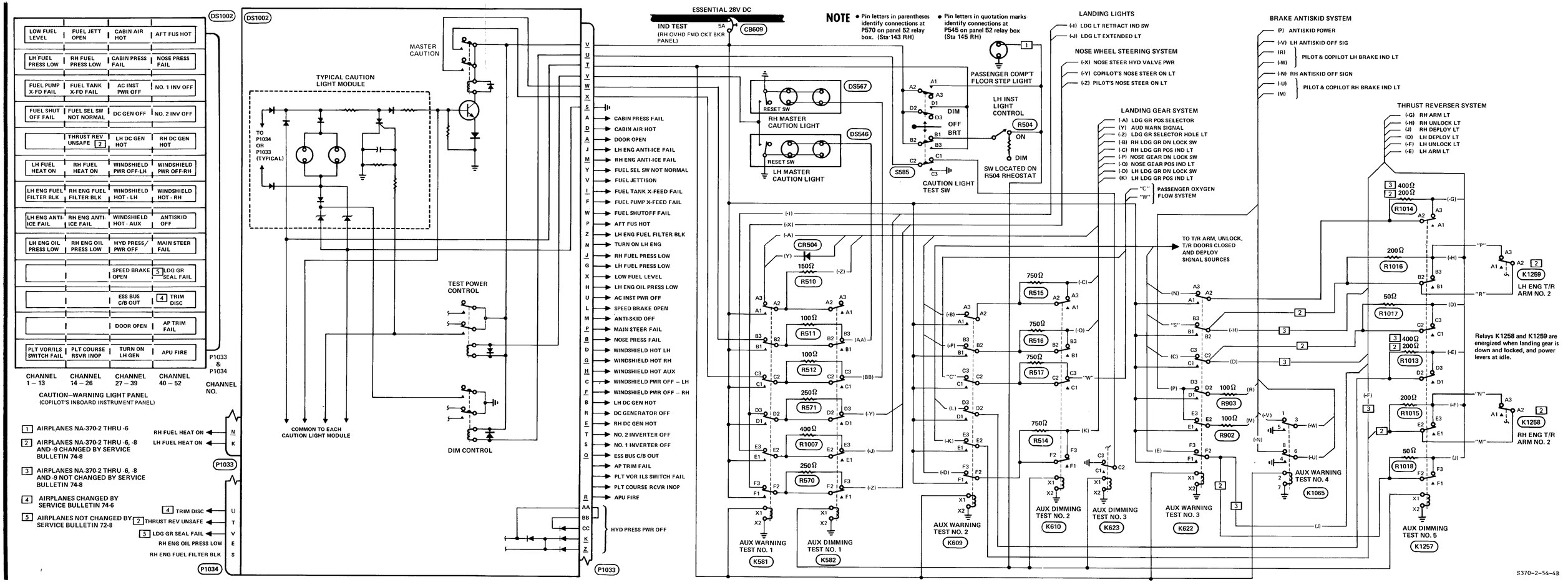
During normal operation, when an airplane system develops a trouble, the system connects 28-volt dc to an indicator light on the panel. At the same time, circuitry is completed to illuminate both master caution lights. The master caution lights are constructed so that the cap operates an internal microswitch when the cap is pushed inward. When actuated, the microswitch completes circuitry that removes the 28-volt dc from both master caution lights. This extinguishes (resets) both master caution lights, allowing the lights to remain off until another airplane system develops trouble.

When the "LIGHTS - CAUTION TEST" switch is held at BRT, the indicator panel dim relay, test relay, and master caution relay are energized. With these relays energized, the individual indicator panel lamps and master caution lights will illuminate in the bright condition. Placing the "LIGHTS - CAUTION TEST" switch to BRT also energizes the four auxiliary warning test relays providing power to illuminate those lamps which are not on the caution-warning light panel.

When the "LIGHTS - CAUTION TEST" switch is held in the DIM position, the indicator panel test relay and master caution relay are energized. The indicator panel dim relay is not energized and the lights illuminate in the dim condition. Placing the "LIGHTS - CAUTION TEST" switch in the DIM position also energizes the four auxiliary dimming test relays and the four auxiliary warning test relays. Power to illuminate those lamps which are not on the caution-warning light panel is forced to go through dimming resistors to illuminate the lamps in a dim condition.

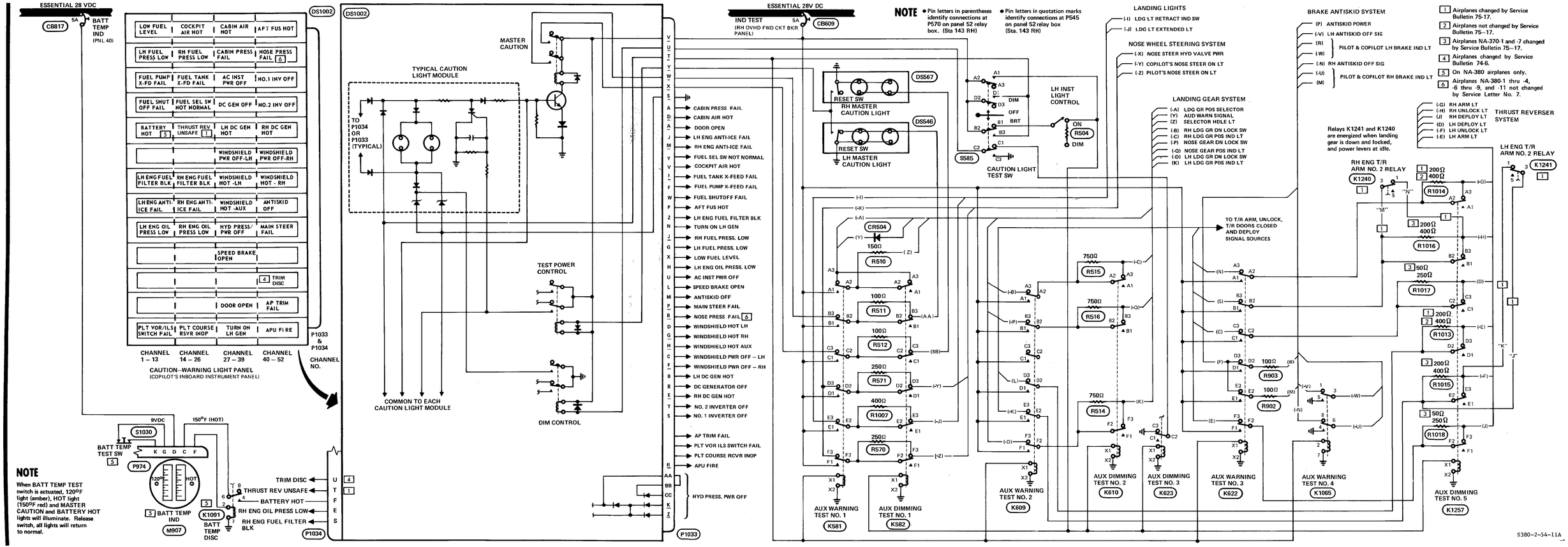
8. Control Wheel Map Lights.

Each control wheel has a map light (behind medallion) and switch (below medallion). Power from UTILITY-PILOT circuit breaker (CB531) on essential bus 28vdc.

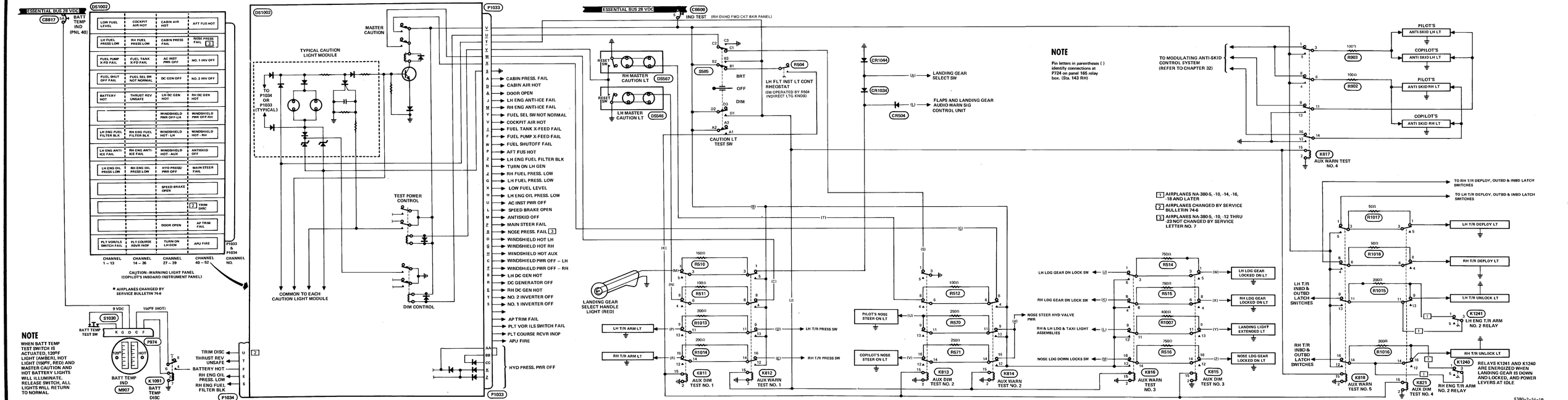

DS1002
LOW FUEL LEVEL
FUEL JETT OPEN
CABIN AIR HOT
AFT FUS HOT
LH FUEL PRESS LOW
RH FUEL PRESS LOW
CABIN PRESS FAIL
NOSE PRESS FAIL
FUEL PUMP X-FD FAIL
FUEL TANK X-FD FAIL
AC INST PWR OFF
NO. 1 INV OFF
FUEL SHUT OFF FAIL
FUEL SEL SW NOT NORMAL
DC GEN OFF
NO. 2 INV OFF
THRUST REV UNSAFE 2
LH DC GEN HOT
RH DC GEN HOT
LH FUEL HEAT ON
RH FUEL HEAT ON
WINDSHIELD PWR OFF-LH
WINDSHIELD PWR OFF-RH
LH ENG FUEL FILTER BLK
RH ENG FUEL FILTER BLK
WINDSHIELD HOT - LH
WINDSHIELD HOT - RH
LH ENG ANTI-ICE FAIL
RH ENG ANTI-ICE FAIL
WINDSHIELD HOT - AUX
ANTISKID OFF
LH ENG OIL PRESS LOW
RH ENG OIL PRESS LOW
HYD PRESS/ PWR OFF
MAIN STEER FAIL
SPEED BRAKE OPEN
5 LDG GR SEAL FAIL
ESS BUS C/B OUT
4 TRIM DISC
DOOR OPEN
AP TRIM FAIL
PLT VOR/ILS SWITCH FAIL
PLT COURSE RSVR INOP
TURN ON LH GEN
APU FIRE
P1033 & P1034
CHANNEL NO.
CHANNEL 1 – 13
CHANNEL 14 – 26
CHANNEL 27 – 39
CHANNEL 40 – 52
CAUTION–WARNING LIGHT PANEL
(COPILOT'S INBOARD INSTRUMENT PANEL)
1 AIRPLANES NA-370-2 THRU -6
2 AIRPLANES NA-370-2 THRU -6, -8 AND -9 CHANGED BY SERVICE BULLETIN 74-8
3 AIRPLANES NA-370-2 THRU -6, -8 AND -9 NOT CHANGED BY SERVICE BULLETIN 74-8
4 AIRPLANES CHANGED BY SERVICE BULLETIN 74-6
5 AIRPLANES NOT CHANGED BY SERVICE BULLETIN 72-8
RH FUEL HEAT ON
LH FUEL HEAT ON
4 TRIM DISC
2 THRUST REV UNSAFE
5 LDG GR SEAL FAIL
RH ENG OIL PRESS LOW
RH ENG FUEL FILTER BLK
P1033
P1034
MASTER CAUTION
TYPICAL CAUTION LIGHT MODULE
TO P1034 OR P1033 (TYPICAL)
COMMON TO EACH CAUTION LIGHT MODULE
TEST POWER CONTROL
DIM CONTROL
CABIN PRESS FAIL
CABIN AIR HOT
DOOR OPEN
LH ENG ANTI-ICE FAIL
RH ENG ANTI-ICE FAIL
FUEL SEL SW NOT NORMAL
FUEL JETTISON
FUEL TANK X-FEED FAIL
FUEL PUMP X-FEED FAIL
FUEL SHUTOFF FAIL
AFT FUS HOT
LH ENG FUEL FILTER BLK
TURN ON LH ENG
RH FUEL PRESS LOW
LH FUEL PRESS LOW
LOW FUEL LEVEL
LH ENG OIL PRESS LOW
AC INST PWR OFF
SPEED BRAKE OPEN
ANTI-SKID OFF
MAIN STEER FAIL
NOSE PRESS FAIL
WINDSHIELD HOT LH
WINDSHIELD HOT RH
WINDSHIELD HOT AUX
WINDSHIELD PWR OFF – LH
WINDSHIELD PWR OFF – RH
LH DC GEN HOT
DC GENERATOR OFF
RH DC GEN HOT
NO. 2 INVERTER OFF
NO. 1 INVERTER OFF
ESS BUS C/B OUT
AP TRIM FAIL
PLT VOR ILS SWITCH FAIL
PLT COURSE RCVR INOP
APU FIRE
HYD PRESS PWR OFF
ESSENTIAL 28V DC
IND TEST (RH OVHD FWD CKT BKR PANEL)
5A
CB609
NOTE • Pin letters in parentheses identify connections at P570 on panel 52 relay box. (Sta 143 RH)
• Pin letters in quotation marks identify connections at P545 on panel 52 relay box (Sta 145 RH)
DS567
RESET SW
RH MASTER CAUTION LIGHT
DS546
RESET SW
LH MASTER CAUTION LIGHT
S585
CAUTION LIGHT TEST SW
DIM
OFF BRT
R504
ON
DIM
SW LOCATED ON R504 RHEOSTAT
PASSENGER COMP'T FLOOR STEP LIGHT
LH INST LIGHT CONTROL
LANDING LIGHTS
(-I) LDG LT RETRACT IND SW
(-J) LDG LT EXTENDED LT
NOSE WHEEL STEERING SYSTEM
(-X) NOSE STEER HYD VALVE PWR
(-Y) COPILOT'S NOSE STEER ON LT
(-Z) PILOT'S NOSE STEER ON LT
LANDING GEAR SYSTEM
(-A) LDG GR POS SELECTOR
(Y) AUD WARN SIGNAL
(-Z) LDG GR SELECTOR HDLE LT
(-B) RH LDG GR DN LOCK SW
(-C) RH LDG GR POS IND LT
(-P) NOSE GEAR DN LOCK SW
(-Q) NOSE GEAR POS IND LT
(-D) LH LDG GR DN LOCK SW
(-K) LH LDG GR POS IND LT
"C" "W" PASSENGER OXYGEN FLOW SYSTEM
BRAKE ANTISKID SYSTEM
(P) ANTISKID POWER
(-V) LH ANTISKID OFF SIG
(R) (-W) PILOT & COPILOT LH BRAKE IND LT
(-N) RH ANTISKID OFF SIGN
(-U) (M) PILOT & COPILOT RH BRAKE IND LT
THRUST REVERSER SYSTEM
(-G) RH ARM LT
(-H) RH UNLOCK LT
(J) RH DEPLOY LT
(D) LH DEPLOY LT
(-F) LH UNLOCK LT
(-E) LH ARM LT
TO T/R ARM, UNLOCK, T/R DOORS CLOSED AND DEPLOY SIGNAL SOURCES
CR504
150Ω R510
100Ω R511
100Ω R512
250Ω R571
400Ω R1007
250Ω R570
750Ω R515
750Ω R516
750Ω R517
750Ω R514
400Ω 200Ω R1014
200Ω R1016
50Ω R1017
400Ω 200Ω R1013
200Ω R1015
50Ω R1018
100Ω R903
100Ω R902
AUX WARNING TEST NO. 1 K581
AUX DIMMING TEST NO. 1 K582
AUX WARNING TEST NO. 2 K609
AUX DIMMING TEST NO. 2 K610
AUX DIMMING TEST NO. 3 K623
AUX WARNING TEST NO. 3 K622
AUX WARNING TEST NO. 4 K1065
AUX DIMMING TEST NO. 5 K1257
K1259 LH ENG T/R ARM NO. 2
K1258 RH ENG T/R ARM NO. 2
Relays K1258 and K1259 are energized when landing gear is down and locked, and power levers at idle.
S370-2-54-4B

Caution-Warning and Indicator Light System Schematic - Airplanes NA-370-2 thru -6, -8 and -9

Figure 33-10-3


ESSENTIAL 28 VDC
CB817
5A
BATT TEMP IND (PNL 40)
DS1002
LOW FUEL LEVEL
COCKPIT AIR HOT
CABIN AIR HOT
AFT FUS HOT
LH FUEL PRESS LOW
RH FUEL PRESS LOW
CABIN PRESS FAIL
NOSE PRESS FAIL 6
FUEL PUMP X-FD FAIL
FUEL TANK X-FD FAIL
AC INST PWR OFF
NO.1 INV OFF
FUEL SHUT OFF FAIL
FUEL SEL SW NOT NORMAL
DC GEN OFF
NO.2 INV OFF
BATTERY HOT 5
THRUST REV UNSAFE 1
LH DC GEN HOT
RH DC GEN HOT
WINDSHIELD PWR OFF-LH
WINDSHIELD PWR OFF-RH
LH ENG FUEL FILTER BLK
RH ENG FUEL FILTER BLK
WINDSHIELD HOT -LH
WINDSHIELD HOT - RH
LH ENG ANTI-ICE FAIL
RH ENG ANTI-ICE FAIL
WINDSHIELD HOT -AUX
ANTISKID OFF
LH ENG OIL PRESS LOW
RH ENG OIL PRESS LOW
HYD PRESS/ PWR OFF
MAIN STEER FAIL
SPEED BRAKE OPEN
4 TRIM DISC
DOOR OPEN
AP TRIM FAIL
PLT VOR/ILS SWITCH FAIL
PLT COURSE RSVR INOP
TURN ON LH GEN
APU FIRE
CHANNEL 1 – 13
CHANNEL 14 – 26
CHANNEL 27 – 39
CHANNEL 40 – 52
CAUTION–WARNING LIGHT PANEL
(COPILOT'S INBOARD INSTRUMENT PANEL)
P1033 & P1034 CHANNEL NO.
DS1002
MASTER CAUTION
TYPICAL CAUTION LIGHT MODULE
TO P1034 OR P1033 (TYPICAL)
TEST POWER CONTROL
DIM CONTROL
COMMON TO EACH CAUTION LIGHT MODULE
V CABIN PRESS FAIL
U CABIN AIR HOT
T DOOR OPEN
Y LH ENG ANTI-ICE FAIL
W RH ENG ANTI-ICE FAIL
X FUEL SEL SW NOT NORMAL
S COCKPIT AIR HOT
A FUEL TANK X-FEED FAIL
D FUEL PUMP X-FEED FAIL
A FUEL SHUTOFF FAIL
J AFT FUS HOT
M LH ENG FUEL FILTER BLK
Y TURN ON LH GEN
V RH FUEL PRESS. LOW
I LH FUEL PRESS. LOW
F LOW FUEL LEVEL
W LH ENG OIL PRESS. LOW
F AC INST PWR OFF
P SPEED BRAKE OPEN
Z ANTISKID OFF
N MAIN STEER FAIL
J NOSE PRESS FAIL 6
G WINDSHIELD HOT LH
G WINDSHIELD HOT RH
H WINDSHIELD HOT AUX
H WINDSHIELD PWR OFF – LH
C WINDSHIELD PWR OFF – RH
F LH DC GEN HOT
B DC GENERATOR OFF
R RH DC GEN HOT
E NO. 2 INVERTER OFF
T NO. 1 INVERTER OFF
S AP TRIM FAIL
PLT VOR ILS SWITCH FAIL
PLT COURSE RCVR INOP
R APU FIRE
AA
BB
CC
K HYD PRESS. PWR OFF
Z
P1033
P1034
9VDC
150°F (HOT)
S1030
BATT TEMP TEST SW
5
K G D C F
P974
120°
HOT
5 BATT TEMP IND
M907
5
K1091
BATT TEMP DISC
TRIM DISC
THRUST REV UNSAFE
BATTERY HOT
RH ENG OIL PRESS LOW
RH ENG FUEL FILTER BLK
U 4
T 1
F
E
S
NOTE
When BATT TEMP TEST switch is actuated, 120°F light (amber), HOT light (150°F red) and MASTER CAUTION and BATTERY HOT lights will illuminate. Release switch, all lights will return to normal.
ESSENTIAL 28V DC
IND TEST 5A
(RH OVHD FWD CKT BKR PANEL)
CB609
NOTE
• Pin letters in parentheses identify connections at P570 on panel 52 relay box. (Sta 143 RH)
• Pin letters in quotation marks identify connections at P545 on panel 52 relay box (Sta. 143 RH)
DS567
RESET SW
RH MASTER CAUTION LIGHT
DS546
RESET SW
LH MASTER CAUTION LIGHT
S585
CAUTION LIGHT TEST SW
DIM
OFF
BRT
LH INST LIGHT CONTROL
ON
R504
DIM
LANDING LIGHTS
(-I) LDG LT RETRACT IND SW
(-J) LDG LT EXTENDED LT
NOSE WHEEL STEERING SYSTEM
(-X) NOSE STEER HYD VALVE PWR
(-Y) COPILOT'S NOSE STEER ON LT
(-Z) PILOT'S NOSE STEER ON LT
LANDING GEAR SYSTEM
(-A) LDG GR POS SELECTOR
(Y) AUD WARN SIGNAL
(Z) SELECTOR HDLE LT
(-B) RH LDG GR DN LOCK SW
(-C) RH LDG GR POS IND LT
(-P) NOSE GEAR DN LOCK SW
(-Q) NOSE GEAR POS IND LT
(-D) LH LDG GR DN LOCK SW
(-K) LH LDG GR POS IND LT
BRAKE ANTISKID SYSTEM
(P) ANTISKID POWER
(-V) LH ANTISKID OFF SIG
(R) (-W) PILOT & COPILOT LH BRAKE IND LT
(-N) RH ANTISKID OFF SIG
(-U) (M) PILOT & COPILOT RH BRAKE IND LT
THRUST REVERSER SYSTEM
(-G) RH ARM LT
(-H) RH UNLOCK LT
(J) RH DEPLOY LT
(D) LH DEPLOY LT
(-F) LH UNLOCK LT
(-E) LH ARM LT
1 Airplanes changed by Service Bulletin 75-17.
2 Airplanes not changed by Service Bulletin 75–17.
3 Airplanes NA-370-1 and -7 changed by Service Bulletin 75–17.
4 Airplanes changed by Service Bulletin 74-6.
5 On NA-380 airplanes only.
6 Airplanes NA-380-1 thru -4, -6 thru -9, and -11 not changed by Service Letter No. 7.
Relays K1241 and K1240 are energized when landing gear is down and locked, and power levers at idle.
RH ENG T/R ARM NO. 2 RELAY
K1240
LH ENG T/R ARM NO. 2 RELAY
K1241
TO T/R ARM, UNLOCK, T/R DOORS CLOSED AND DEPLOY SIGNAL SOURCES
CR504
150Ω R510
100Ω R511
100Ω R512
250Ω R571
400Ω R1007
250Ω R570
750Ω R515
750Ω R516
750Ω R514
200Ω 400Ω R1014
200Ω 400Ω R1016
50Ω 250Ω R1017
200Ω 400Ω R1013
100Ω R903
100Ω R902
200Ω 400Ω R1015
50Ω 250Ω R1018
AUX WARNING TEST NO. 1 K581
AUX DIMMING TEST NO. 1 K582
AUX WARNING TEST NO. 2 K609
AUX DIMMING TEST NO. 2 K610
AUX DIMMING TEST NO. 3 K623
AUX WARNING TEST NO. 3 K622
AUX WARNING TEST NO. 4 K1065
AUX DIMMING TEST NO. 5 K1257
S380-2-54-11A

Caution-Warning and Indicator Light System Schematic - Airplanes NA-370-1 and 7, and NA-380-1 thru -4, -6 thru -9 and -11
Figure 33-10-4


ESSENTIAL BUS 28 VDC
CB817
BATT TEMP IND (PNL 40)
DS1002
LOW FUEL LEVEL
COCKPIT AIR HOT
CABIN AIR HOT
AFT FUS HOT
LH FUEL PRESS LOW
RH FUEL PRESS LOW
CABIN PRESS FAIL
NOSE PRESS FAIL 3
FUEL PUMP X-FD FAIL
FUEL TANK X-FD FAIL
AC INST PWR OFF
NO. 1 INV OFF
FUEL SHUT OFF FAIL
FUEL SEL SW NOT NORMAL
DC GEN OFF
NO. 2 INV OFF
BATTERY HOT
THRUST REV UNSAFE
LH DC GEN HOT
RH DC GEN HOT
WINDSHIELD PWR OFF-LH
WINDSHIELD PWR OFF-RH
LH ENG FUEL FILTER BLK
RH ENG FUEL FILTER BLK
WINDSHIELD HOT - LH
WINDSHIELD HOT - RH
LH ENG ANTI-ICE FAIL
RH ENG ANTI-ICE FAIL
WINDSHIELD HOT - AUX
ANTISKID OFF
LH ENG OIL PRESS LOW
RH ENG OIL PRESS LOW
HYD PRESS/ PWR OFF
MAIN STEER FAIL
SPEED BRAKE OPEN
2 TRIM DISC
DOOR OPEN
AP TRIM FAIL
PLT VOR/ILS SWITCH FAIL
PLT COURSE RCVR INOP
TURN ON LH GEN
APU FIRE
P1033 & P1034
CHANNEL NO.
CHANNEL 1 – 13
CHANNEL 14 – 26
CHANNEL 27 – 39
CHANNEL 40 – 52
CAUTION-WARNING LIGHT PANEL (COPILOT'S INBOARD INSTRUMENT PANEL)
* AIRPLANES CHANGED BY SERVICE BULLETIN 74-6
9 VDC
150°F (HOT)
S1030
BATT TEMP TEST SW
K G D C F
P974
120°F
HOT
BATT TEMP IND
M907
K 1091 BATT TEMP DISC
TRIM DISC
THRUST REV UNSAFE
BATTERY HOT
RH ENG OIL PRESS. LOW
RH ENG FUEL FILTER BLK
P1034
NOTE
WHEN BATT TEMP TEST SWITCH IS ACTUATED, 120°F LIGHT (AMBER), HOT LIGHT (150°F, RED) AND MASTER CAUTION AND HOT BATTERY LIGHTS WILL ILLUMINATE. RELEASE SWITCH, ALL LIGHTS WILL RETURN TO NORMAL
DS1002
TYPICAL CAUTION LIGHT MODULE
TO P1034 OR P1033 (TYPICAL)
MASTER CAUTION
TEST POWER CONTROL
COMMON TO EACH CAUTION LIGHT MODULE
DIM CONTROL
P1033
CABIN PRESS. FAIL
CABIN AIR HOT
DOOR OPEN
LH ENG ANTI-ICE FAIL
RH ENG ANTI-ICE FAIL
FUEL SEL SW NOT NORMAL
COCKPIT AIR HOT
FUEL TANK X-FEED FAIL
FUEL PUMP X-FEED FAIL
FUEL SHUTOFF FAIL
AFT FUS HOT
LH ENG FUEL FILTER BLK
TURN ON LH GEN
RH FUEL PRESS. LOW
LH FUEL PRESS. LOW
LOW FUEL LEVEL
LH ENG OIL PRESS. LOW
AC INST PWR OFF
SPEED BRAKE OPEN
ANTISKID OFF
MAIN STEER FAIL
NOSE PRESS. FAIL 3
WINDSHIELD HOT LH
WINDSHIELD HOT RH
WINDSHIELD HOT AUX
WINDSHIELD PWR OFF – LH
WINDSHIELD PWR OFF – RH
LH DC GEN HOT
DC GENERATOR OFF
RH DC GEN HOT
NO. 2 INVERTER OFF
NO. 1 INVERTER OFF
AP TRIM FAIL
PLT VOR ILS SWITCH FAIL
PLT COURSE RCVR INOP
APU FIRE
HYD PRESS. PWR OFF
ESSENTIAL BUS 28 VDC
CB606
IND TEST (RH OVHD FWD CKT BKR PANEL)
RESET SW
RH MASTER CAUTION LT
DS567
LH MASTER CAUTION LT
DS546
S585
BRT
OFF
DIM
CAUTION LT TEST SW
R504
LH FLT INST LT CONT RHEOSTAT
(SW OPERATED BY R504 INDIRECT LTG KNOB)
CR1044
CR1034
CR504
LANDING GEAR SELECT SW
FLAPS AND LANDING GEAR AUDIO WARN SIG CONTROL UNIT
LANDING GEAR SELECT HANDLE LIGHT (RED)
LH T/R ARM LT
RH T/R ARM LT
R510
R511
R1013
R1014
K811 AUX DIM TEST NO. 1
K812 AUX WARN TEST NO. 1
LH T/R PRESS SW
RH T/R PRESS SW
PILOT'S NOSE STEER ON LT
COPILOT'S NOSE STEER ON LT
R512
R570
R571
K813 AUX DIM TEST NO. 2
K814 AUX WARN TEST NO. 2
LH LDG GEAR DN LOCK SW
RH LDG GEAR DN LOCK SW
NOSE STEER HYD VALVE PWR
RH & LH LDG & TAXI LIGHT ASSEMBLIES
NOSE LDG DOWN LOCKS SW
R514
R515
R1007
R516
K816 AUX WARN TEST NO. 3
K815 AUX DIM TEST NO. 3
LH LDG GEAR LOCKED DN LT
RH LDG GEAR LOCKED DN LT
LANDING LIGHT EXTENDED LT
NOSE LDG GEAR LOCKED DN LT
NOTE
Pin letters in parentheses ( ) identify connections at P724 on panel 165 relay box. (Sta. 143 RH)
TO MODULATING ANTI-SKID CONTROL SYSTEM (REFER TO CHAPTER 32)
R903
R902
PILOT'S ANTI-SKID LH LT
COPILOT'S ANTI SKID LH LT
PILOT'S ANTI SKID RH LT
COPILOT'S ANTI SKID RH LT
K817 AUX WARN TEST NO. 4
1 AIRPLANES NA-380-5, -10, -14, -16, -18 AND LATER
2 AIRPLANES CHANGED BY SERVICE BULLETIN 74-6
3 AIRPLANES NA-380-5, -10, -12 THRU -23 NOT CHANGED BY SERVICE LETTER NO. 7
TO RH T/R DEPLOY, OUTBD & INBD LATCH SWITCHES
TO LH T/R DEPLOY, OUTBD & INBD LATCH SWITCHES
R1017
R1018
R1015
R1016
LH T/R INBD & OUTBD LATCH SWITCHES
RH T/R INBD & OUTBD LATCH SWITCHES
LH T/R DEPLOY LT
RH T/R DEPLOY LT
LH T/R UNLOCK LT
RH T/R UNLOCK LT
K1241 LH ENG T/R ARM NO. 2 RELAY
K1240 RH ENG T/R ARM NO. 2 RELAY
K818 AUX WARN TEST NO. 5
K821 AUX DIM TEST NO. 4
RELAYS K1241 AND K1240 ARE ENERGIZED WHEN LANDING GEAR IS DOWN AND LOCKED, AND POWER LEVERS AT IDLE
S380-2-54-19

Caution-Warning and Indicator Light System Schematic
Airplanes NA-380-5, -10, -12 and Later
Figure 33-10-5

FLIGHT COMPARTMENT - MAINTENANCE PRACTICES

1. General

Cockpit lighting troubles will usually be of a minor nature. Replacement of defective lamps, switches, or rheostats, and the repair of faulty wiring is all that usually will be necessary. If it is determined that lamps and circuitry are not defective, check for shorted or open wiring. Turn off electrical power before replacing lamps or lamp assemblies, especially in caution light system.

2. Testing Flight Compartment Lighting

NOTE: Connect a 28-volt dc external power supply to airplane.

A. Test floodlight systems as follows:

(1) Close the following circuit breakers:

"FLOOD - CONSOLE" - on right-hand ovhd fwd CB panel
"FLOOD - INST" - on right-hand ovhd fwd CB panel
"FLOOD - PEDESTAL" - on right-hand ovhd fwd CB panel
"FLOOD - RADIO & ENG INST" - on right-hand ovhd fwd CB panel

(2) Rotate "CONSOLE LIGHTS - FLOOD" knobs, on left and right console lighting control panels, in a clockwise direction. The intensity of the left and right console floodlights should vary from off to bright.

(3) Rotate "FLIGHT INSTRUMENT LIGHTS - FLOOD" knobs, on left and right console lighting control panels, in a clockwise direction. The intensity of the right and left instrument panel floodlights should vary from off to bright.

(4) Rotate "RAD & ENG INST - FLOOD" knob, on left console lighting control panel, in a clockwise direction. The intensity of the radio and engine instrument floodlights should vary from off to bright.

(5) Rotate "PED & OVHD - FLOOD" knob, on left console lighting control panel, in a clockwise direction. The intensity of the pedestal floodlights should vary from off to bright.

B. Test indirect lights as follows:

(1) Place "INVERTER" switch at NO 1.

(2) Close the following circuit breakers on left overhead aft circuit-breaker panel:

"PILOT XFMR"
"CO-PILOT XFMR"
"INSTRUMENT - LH"

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"INSTRUMENT - RH"
"CONSOLE - LH"
"CONSOLE - RH"
"RADIO & ENG INST"
"OVHD & PED"

(3) Rotate "FLIGHT INSTRUMENT LIGHTS - INDIRECT" knobs, on left and right console lighting control panels, in a clockwise direction. The intensity of the lights on the instruments of the inboard and outboard instrument panels should vary from off to bright. Rotate "IFS" knob on the pilot's instrument panel. The intensity of the lights inside the pilot's HSI (horizontal situation indicator) and flight director indicator should vary.

(4) Rotate "CONSOLE LIGHTS - INDIRECT" knobs, on left and right console lighting control panels, in a clockwise direction. The intensity of the lights embedded in the left and right consoles should vary from off to bright. Rotate "IFS" knob on the copilot's instrument panel. The intensity of the lights inside the copilot's HSI (horizontal situation indicator) and flight director indicators should vary.

(5) Rotate "RAD & ENG INST - INDIRECT" knob, on left console lighting control panel, in a clockwise direction. The intensity of the lights embedded in the radio and engine instrument panels should vary from off to bright.

(6) Rotate "PED & OVHD - INDIRECT" knob, on left console lighting control panel, in a clockwise direction. The intensity of the lights embedded in the control panels on the pedestal and overhead areas should vary from off to bright. The stand-by compass indirect lights should also vary from off to bright.

C. Test thunderstorm lights as follows:

(1) Close "THUNDERSTORM" circuit breaker.

(2) Place "LIGHTS - THUNDERSTORM" switch to BRT. The thunderstorm lights should illuminate with a brilliant white light.

(3) Place "LIGHTS - THUNDERSTORM" switch to DIM. The brilliance of the lights should be noticeably reduced.

D. Test cockpit dome light as follows:

(1) Close "COCKPIT DOME" circuit breaker on right-hand overhead forward circuit-breaker panel.

(2) Place "COCKPIT DOME" switch at WHITE. The cockpit dome light should illuminate with a white light.

(3) Place "COCKPIT DOME" switch at RED. The cockpit dome light should illuminate with a red light.

(4) Rotate overhead map light knobs from "OFF" to "BRIGHT". Light intensity should vary from off to bright.

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E. Test cockpit utility lights as follows:

(1) Close "UTILITY - PILOT" and "UTILITY - COPILOT" circuit breakers on right-hand console circuit-breaker panel.

(2) Press button on each utility light. The utility light should illuminate. Release button.

(3) Rotate rheostat knob from off. The light should come on, and the rheostat should control the brilliancy.

F. Test caution-warning and indicating light system as follows:

(1) Close "IND TEST & STEP" circuit breaker on right-hand overhead forward circuit-breaker panel.

(2) Place "LIGHTS - CAUTION TEST" switch (on cabin sign and light control overhead switch panel) at BRT, and hold. All indicator lights on the copilot's inboard instrument panel should come on bright. Release "LIGHTS - CAUTION TEST" switch.

(3) Notice master caution lights (on edge of instrument panel shroud). Both caution lights should be on.

(4) Push in and release one of the master caution light caps. When released, both master caution lights should go out.

(5) Place "LIGHTS - CAUTION TEST" switch at DIM, and hold. All indicator lights on the copilot's inboard instrument panel should come on dim.

(6) Notice master caution lights. Both caution lights should be dimly lit.

(7) Push in and release other master caution light cap. When released, both caution lights should go out.

3. Caution and Warning Panel Removal and Installation

CAUTION: WHEN REMOVING AND INSTALLING OR TROUBLE SHOOTING THE CAUTION AND WARNING PANEL, ALL ELECTRICAL POWER MUST BE OFF FROM AIRPLANE, TO AVOID SHORTING OUT DIODES AND WELDING RELAY CONTACTS.

A. Removing caution and warning light (C&W) panel. (See figure 33-10-1.)

(1) Remove all electrical power from airplane.

(2) Remove all legend modules (13) from C&W panel, by pulling each module aft until free of panel.

(3) Loosen five instrument panel fasteners (panel that C&W panel is mounted on), and pull panel aft and disconnect two electrical connectors to C&W panel.

(4) Remove 6 C&W panel mounting screws and carefully remove C&W panel from instrument panel.

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B. Installing caution and warning panel.

(1) Remove all electrical power from airplane.

(2) Mount C&W panel behind instrument panel and install 6 mounting screws.

NOTE: All legend modules (13) must be removed from C&W panel before installation.

(3) Connect two electrical connectors to C&W panel.

NOTE: Keyways located at 12 O'clock on receptacles.

(4) Install instrument panel by tightening 5 fasteners.

(5) Install all (13) legend modules on C&W panel, by pressing each module into its indexed receptacle.

NOTE: Legend modules are indexed to be installed in one location only.

(6) After C&W panel installation perform "Test caution - warning and indicating light system", refer to 33-10-1, step 2, F.

4. Removal and Installing Lamp or Legend Plate in Caution and Warning Panel

CAUTION: WHEN REMOVING AND INSTALLING LAMPS OR LEGEND PLATES OR TROUBLE SHOOTING CAUTION AND WARNING PANEL, ALL ELECTRICAL POWER MUST BE OFF FROM AIRPLANE, TO AVOID SHORTING OUT DIODES AND WELDING RELAY CONTACTS.

A. Removing and installing lamp or legend plate as follows:

(1) Remove all electrical power from airplane.

(2) Remove module from panel by pulling aft until free of panel.

(3) Replacing lamp, if required - Remove defective lamp (MS25237-327) from back of module and press in new lamp and perform steps (8) and (9).

(4) Replacing legend plate, if required - Remove trim cover (frames four legends) by prying loose at either end of module near base and perform steps (5) through (8).

(5) Remove long clear plastic (covers four legend plates), legend plate and short clear plastic (size of legend plate).

(6) Install short clear plastic, approved legend plate and long clear plastic.

(7) Place trim cover over long clear plastic and snap into place on both ends of module.

(8) Install module on panel by pushing module into its indexed receptacle.

(9) After lamp or legend plate replacement perform "Test caution - warning and indicating light system", refer to 33-10-1, steps 1 and 2, F.

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PASSENGER COMPARTMENT - DESCRIPTION AND OPERATION

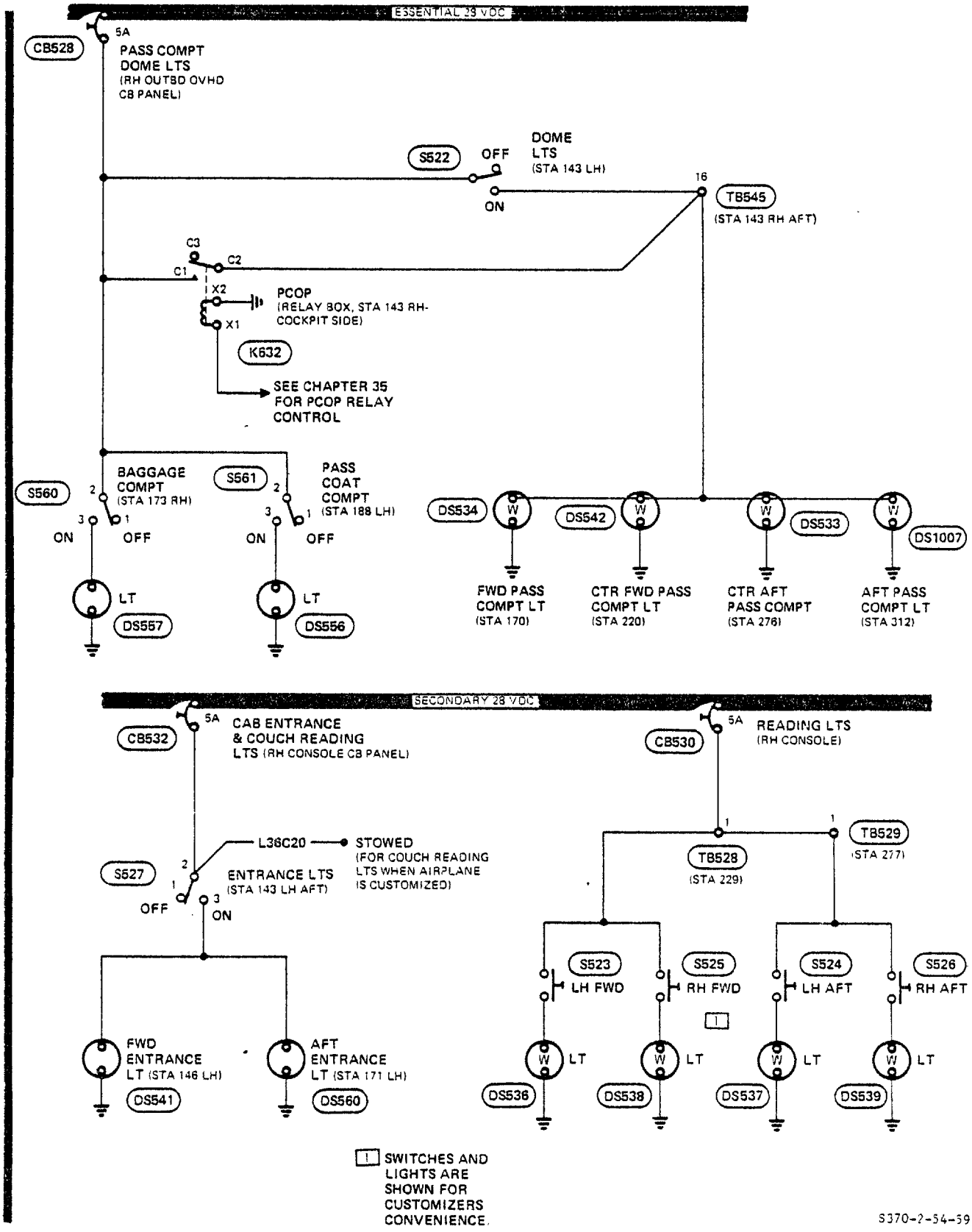
1. Passenger Compartment Warning Sign

The passenger compartment warning sign system automatically, or by manual selection, illuminates the warning signs in the passenger compartment. The "LIGHTS-NO SMOKING" and "LIGHTS-FASTEN BELTS" switches on the cabin sign and light control overhead switch panel permits manual control of the warning signs. Automatic control of the warning signs is done by the operation of an altitude-sensitive switch. The switch is actuated closed when cabin altitude is 13,500 (±500) feet. When the passenger oxygen flow control switch is at AUTO (its normal operating position) the passenger compartment oxygen pressure relay will energize when the altitude switch closes. This completes circuits to light the USE OXYGEN MASKS and NO SMOKING signs in the passenger compartment. The signs illuminate to advise the passengers to refrain from smoking and to put on their oxygen masks. The signs may be illuminated at the pilot's discretion by positioning the passenger oxygen flow control switch to OPEN. (See Chapter 35.)

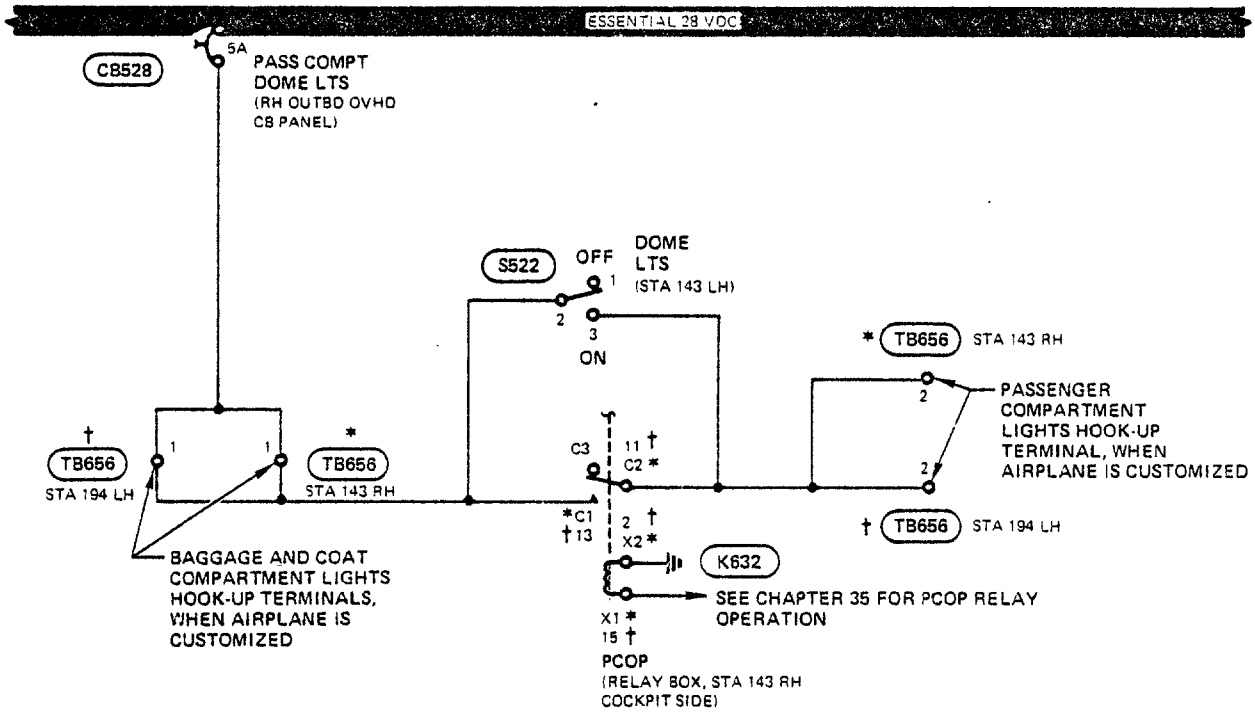
The cabin pressure failure warning microswitch is automatically operated by an aneroid-type sensing element. When the cabin pressurization system fails while the airplane altitude is above 10,000 feet, the sensing element actuates the microswitch. The microswitch energizes the cabin pressure failure relay which connects dc power to the "FASTEN SEAT BELTS" lights in the passenger compartment warning signs. The sign illuminates, advising the passengers to fasten their safety belts in preparation for a rapid airplane descent to an altitude below 10,000 feet. The fast descent is necessary to compensate for the loss of cabin pressurization. (See Chapter 35.)

2. Passenger Dome, Cabin Entrance, Passenger Reading and Baggage Compartments Lighting Systems

See figures 33-20-1 and 33-20-2. These lighting systems are shown for customizer convenience.

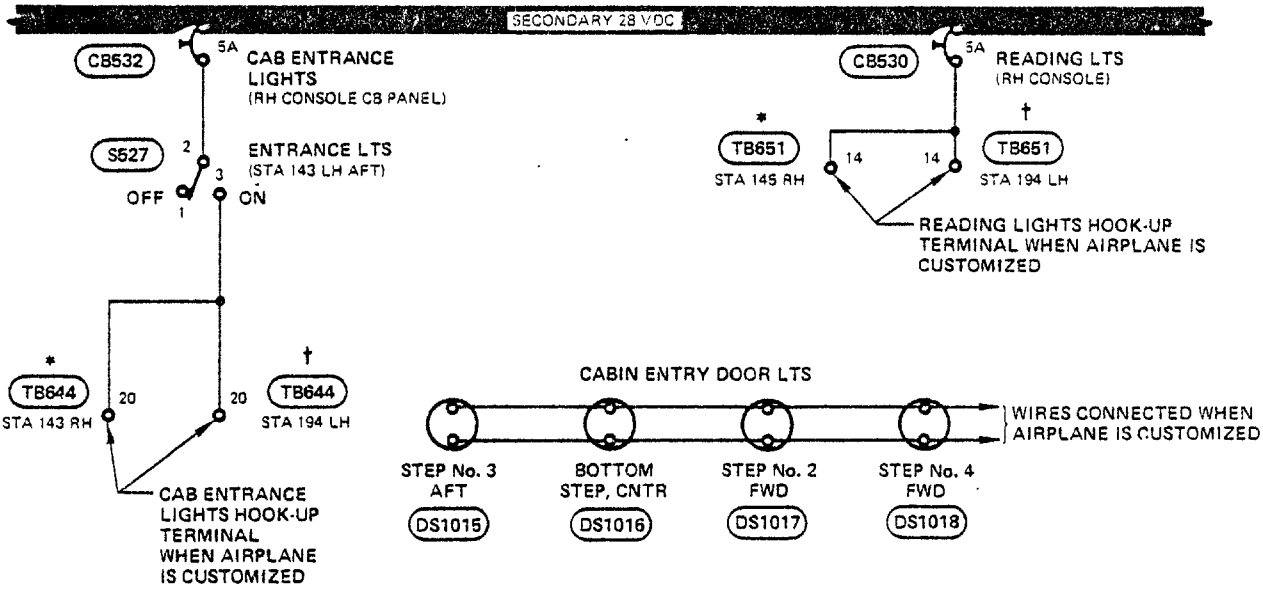


Passenger Dome, Cabin Entrance, Passenger Reading and Baggage Compartment Lighting Schematic - Airplanes NA-370-1 and Later
Figure 33-20-1



* AIRPLANES NA-380-1 THRU -4, -6 THRU -9 AND -11

† AIRPLANES NA-380-5, -10, -12 AND LATER



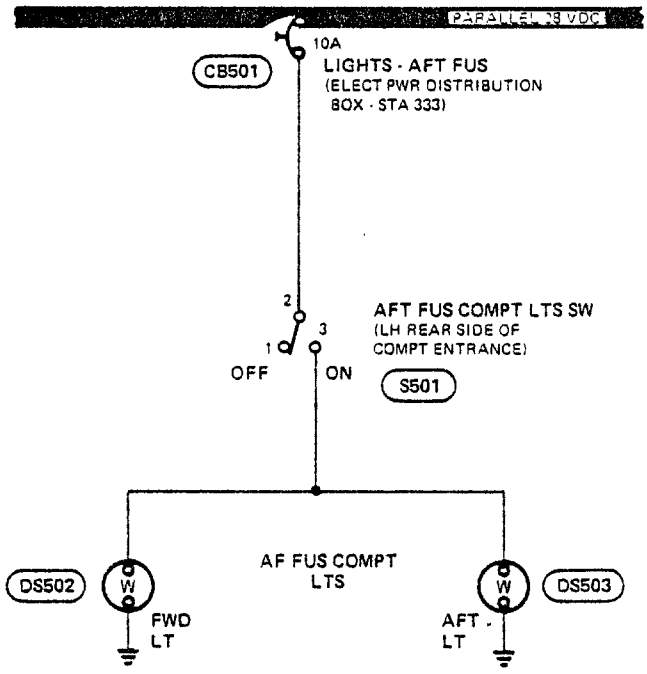
S380-2-54-20

Passenger Dome, Cabin Entrance, Passenger Reading and Baggage Compartment Lighting Schematic - Airplanes NA-380-1 and Later
Figure 33-20-2

CARGO AND SERVICE COMPARTMENT - DESCRIPTION AND OPERATION

1. Aft Fuselage Compartment Lighting

The aft fuselage compartment lighting consists of two overhead dome lights provided to illuminate the compartment. The lights are controlled by a switch on the left rear side of the compartment entrance at fuselage station 390. Power for the aft fuselage compartment lights comes from the paralleling 28-volt dc bus through the "LIGHTS - AFT FUS" circuit breaker, on the electrical power distribution box at fuselage station 333. (See figure 33-30-1.)



S380-2-34-21

Aft Fuselage Compartment Lighting Schematic
Figure 33-30-1

EXTERIOR - DESCRIPTION AND OPERATION

1. General

The exterior lighting system consists of the position lights, anticollision lights, landing and taxi lights, strobe lights (high intensity light), and ice-check lights. The exterior lighting installed on the airplane provides the necessary lighting for landing and taxi operations, for position marking while in flight, for warning approaching airplanes of a possible air collision, and for lighting up the leading edges of the wings to aid in a visual check of the formation of ice on the wings during night flights under bad-weather conditions. (See figures 33-40-1 and 33-40-2.)

2. Landing and Taxi Lights

The landing and taxi light system consists of a left and right retractable landing and taxi light assembly, a landing light ground control relay, and control circuitry. The retractable landing and taxi light assemblies are on the underside of the fuselage at fuselage station 36.

When the control switches in the cockpit are operated, the lights come on and extend to a landing or taxi position, depending on whether the airplane is air-borne or on the ground. With the airplane on the ground and the "LDG & TAXI LTS" switches are placed at BOTH ON and EXTEND, both lights come on, and both light assemblies extend to the taxi position. Upon reaching the taxi position, the taxi microswitches automatically disconnect dc power from the extend motor windings.

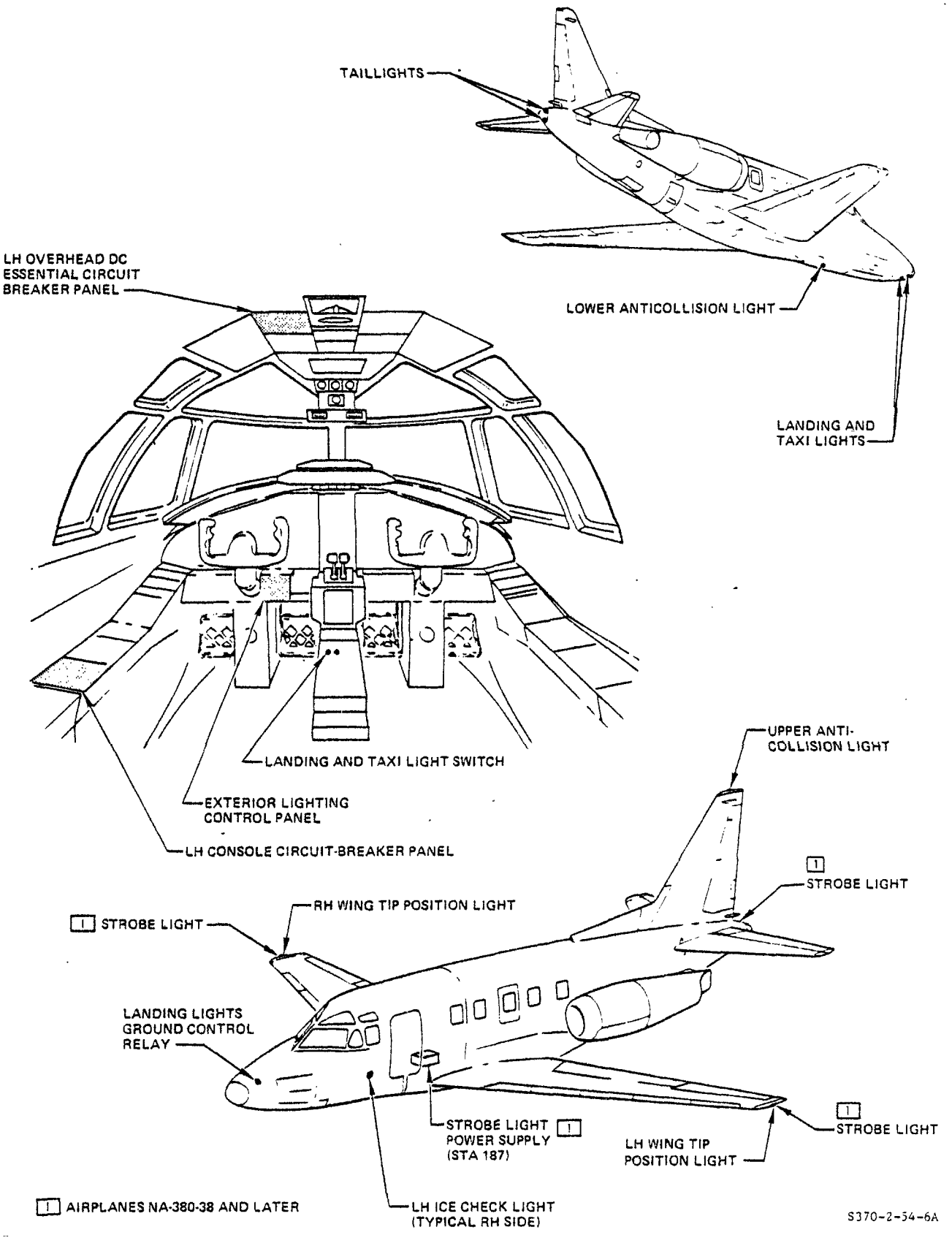
While the airplane is air-borne and the "LDG & TAXI LTS" switches are placed at BOTH ON and EXTEND, both lights come on and both light assemblies extend to the landing position. Upon reaching the landing position, the landing microswitches automatically disconnect dc power from the extend motor windings.

Immediately after touchdown, with lights turned on and in the landing position, the landing light control relay becomes energized and connects dc power to the taxi microswitches. This causes the light assemblies to extend further to the taxi position. In a similar manner, after take-off, the landing light control relay becomes de-energized and connects dc power to the auxiliary microswitches in each light assembly. This causes the landing lights to retract to the landing position. When the control switches are placed at OFF and RETRACT, wehther the airplane is air-borne or on the ground, the lights go out and the light assemblies retract into the fuselage.

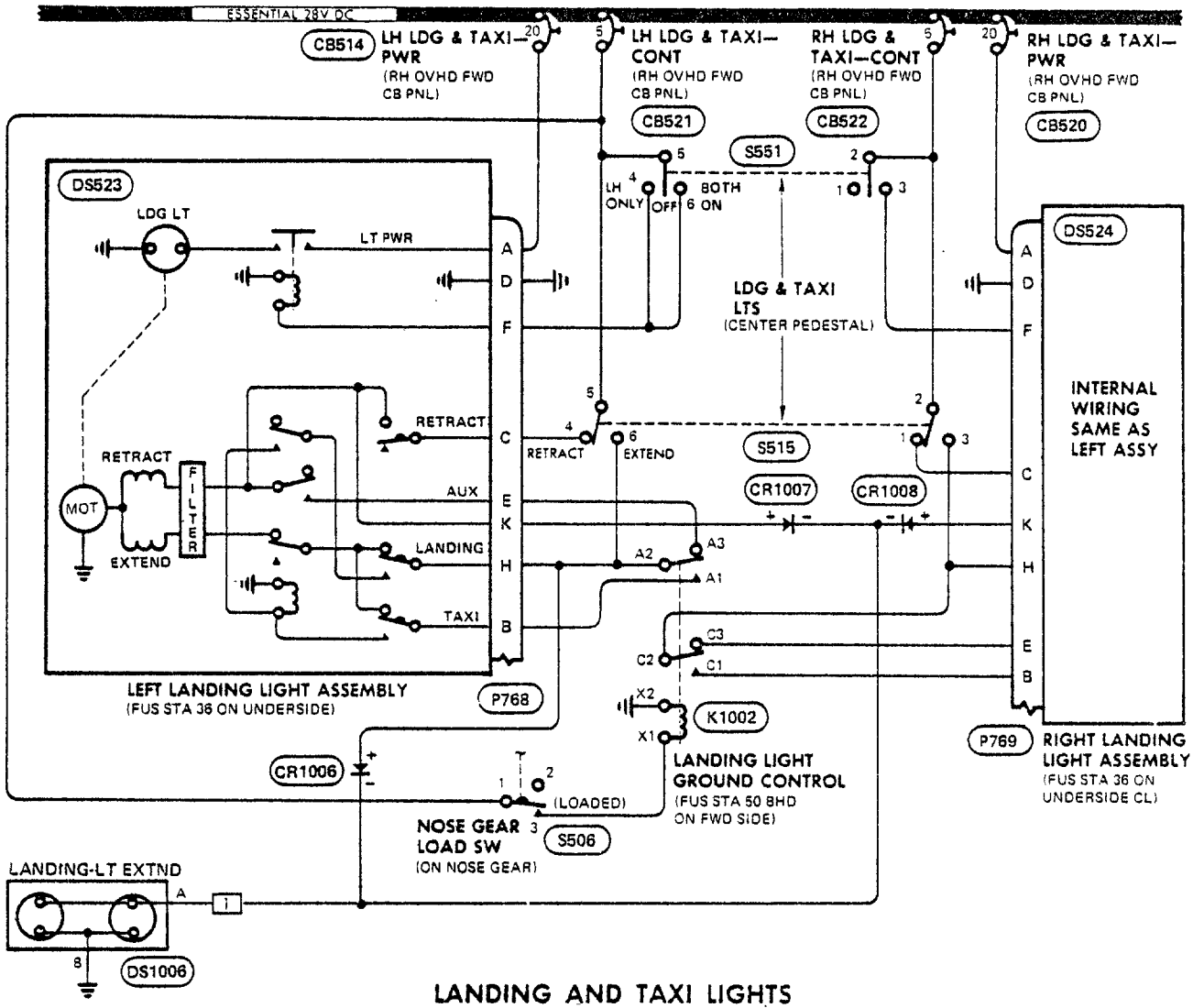
When the "LDG & TAXI LTS" switch is placed in LH ON only, the left landing and taxi light only is turned on. This may be desirable when one light provides adequate illumination for taxi or parking maneuvers, since the lamps are rated for only 8 hours of operation.

3. Position Lights

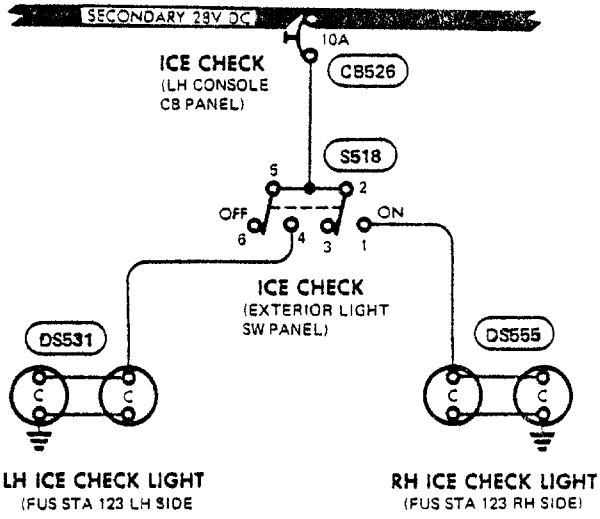
The position light system consists of the left and right wing tip lights and the taillights. When the "EXTERIOR LIGHTS - POSITION" switch is placed


TAILLIGHTS
LH OVERHEAD DC ESSENTIAL CIRCUIT BREAKER PANEL
LOWER ANTICOLLISION LIGHT
LANDING AND TAXI LIGHTS
LANDING AND TAXI LIGHT SWITCH
EXTERIOR LIGHTING CONTROL PANEL
LH CONSOLE CIRCUIT-BREAKER PANEL
UPPER ANTI-COLLISION LIGHT
[1] STROBE LIGHT
[1] STROBE LIGHT
RH WING TIP POSITION LIGHT
LANDING LIGHTS GROUND CONTROL RELAY
STROBE LIGHT [1] POWER SUPPLY (STA 187)
[1] STROBE LIGHT
LH WING TIP POSITION LIGHT
[1] AIRPLANES NA-380-38 AND LATER
LH ICE CHECK LIGHT (TYPICAL RH SIDE)
S370-2-54-6A

Exterior Lighting Equipment Location
Figure 33-40-1



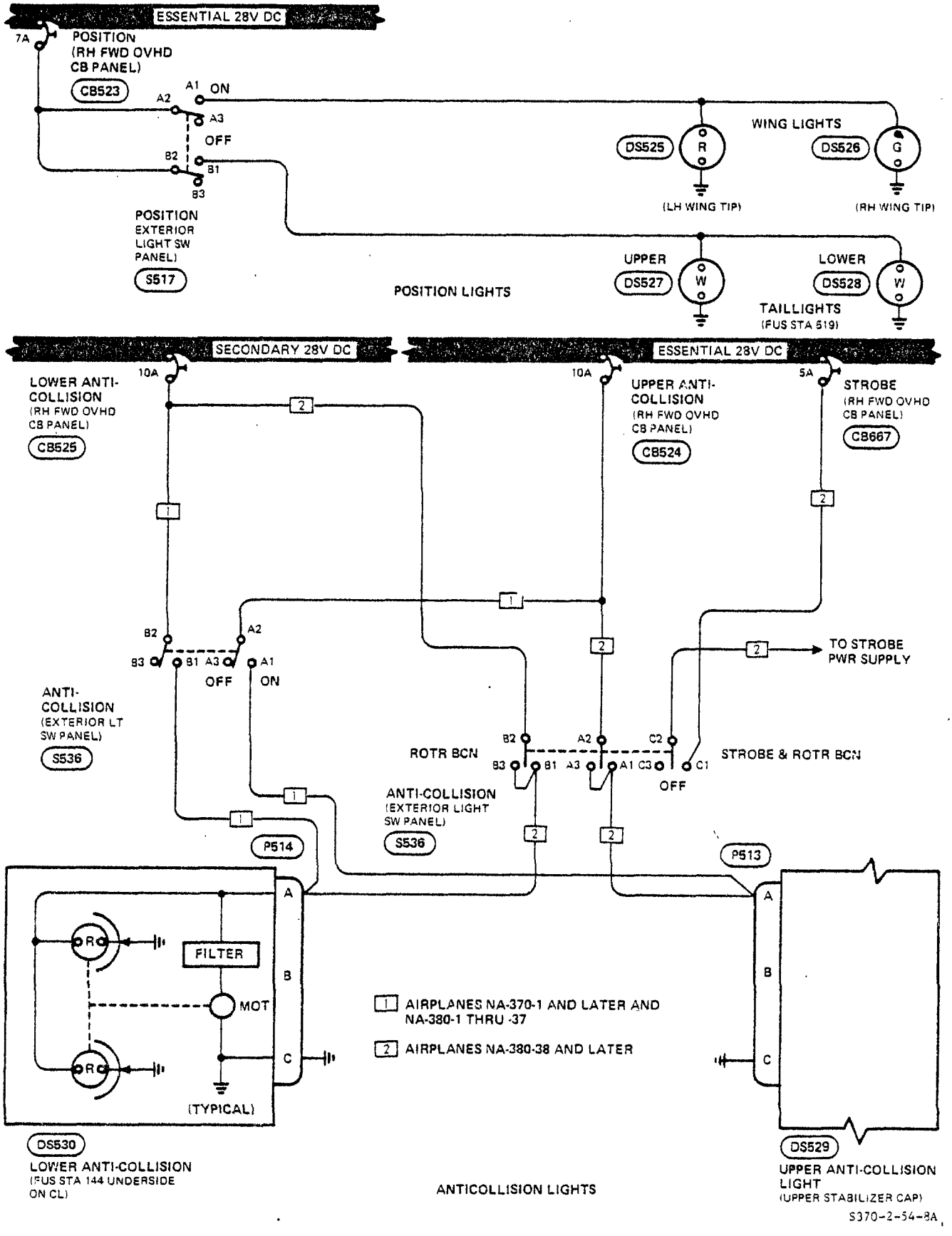
LANDING AND TAXI LIGHTS



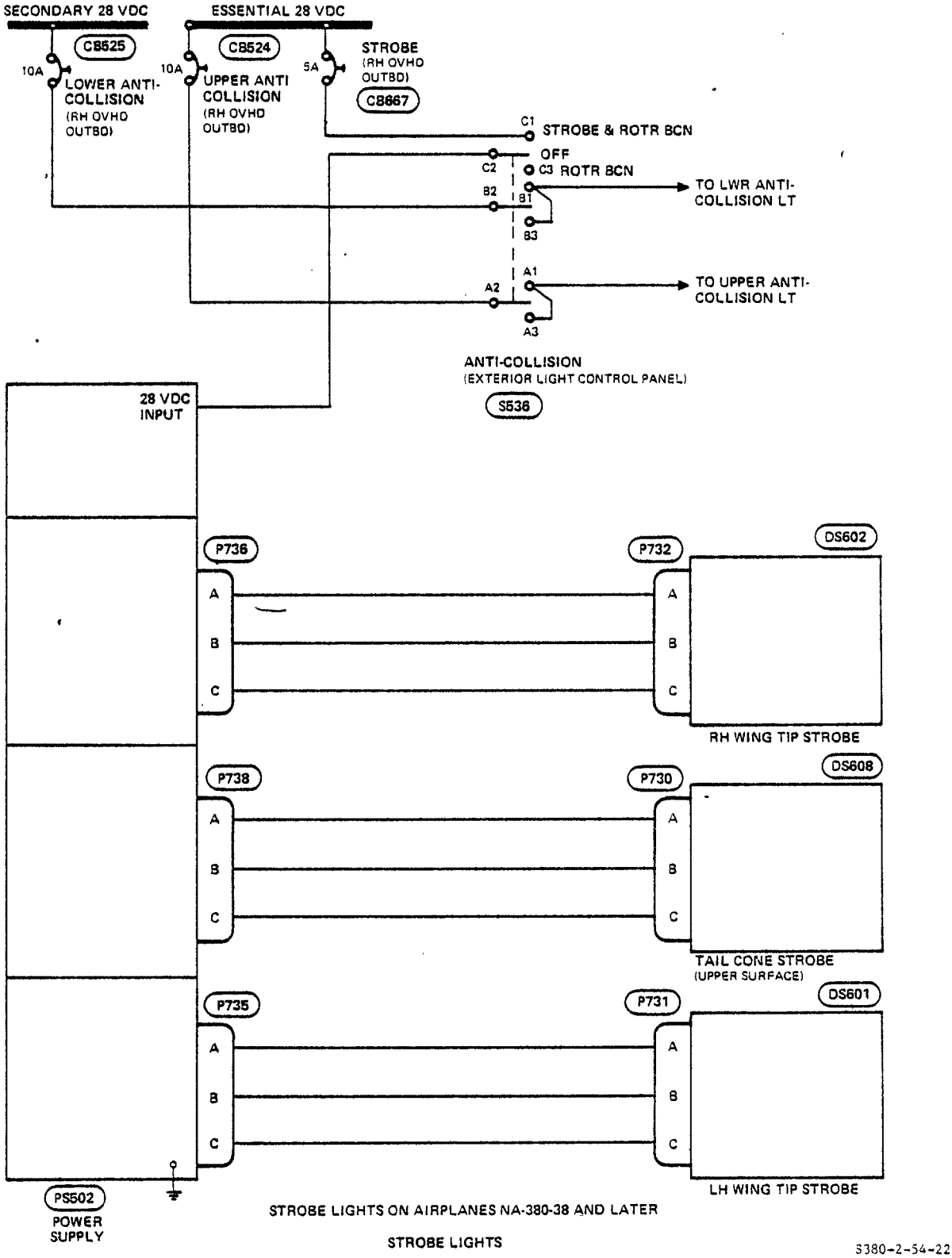
[1] FOR CLARITY, THE EXTEND INDICATING LIGHT TEST AND DIM CIRCUITS ARE NOT SHOWN

ICE CHECK LIGHTS

S370-2-34-7A



Exterior Lighting Schematic
Figure 33-40-2 (Sheet 2 of 3)



Exterior Lighting Schematic
Figure 33-40-2 (Sheet 3 of 3)

at ON, dc power is connected to the position lights. The lights come on and remain on until the switch is placed at OFF.

4. Ice-check Lights

The ice-check light system consists of the left and right ice-check light assemblies and the control switch. The lights are on the left and right sides of the fuselage at station 123. The lights will illuminate the leading edges of the wings when "EXTERIOR LIGHTS - ICE CHECK" switch is placed at ON. (See figure 33-40-2.)

5. Anticollision Lights

The anticollision light system consists of an upper and lower anticollision light assembly and a control switch. Each light assembly contains two lights, a motor, a gear train and connecting linkage, and a radio noise filter. Both lights within each anticollision light assembly are mechanically linked together, and are rotated through 180 degrees by the motor, gear train, and mechanical linkage. Power for the anticollision lights is supplied by the essential and secondary dc busses through the "ANTI-COLLISION - LOWER" and "ANTICOLLISION - UPPER" circuit breakers. The lights are controlled by the "EXTERIOR LIGHTS - ANTI-COLLISION" switch. (See figure 33-40-2.)

6. Strobe Lights (On Airplanes NA-380-38 and later)

In addition to the aircraft position lighting system, three high intensity, white (strobe) lights are installed. The wing tip strobe lights are an integral part of the position light assembly. The tail strobe light is a single lamp assembly and is installed on the upper surface of the tail cone. The system consists of three individual power supplies and a timing circuit (part of sub-chassis). The power supplies are module type units which plug into a sub-chassis assembly which is located at fuselage station 187 CL.

The power supplies are identical units and are transistorized dc to dc converter, which functions to increase the 28 VDC power supplied to the units to approximately 450 VDC. (See figure 33-40-2.) If any one of the power supplies should fail, the remaining two power supplies will function normally Each power supply is labeled as to which light it supplies the high voltage for operation of the inert gas-filled strobe bulb. The electrical wiring from each power supply to its individual strobe light, is special, shielded wire to prevent the high voltage pulses from inductively coupling to any adjacent wiring or equipment. This prevents stray voltages that may render a system inoperative or provide a false indication.

The timing circuit is part of the sub-chassis assembly and is a fully transistorized unit which requires no maintenance. The timer output consists of 50 pulses per minute which provides for 50 flashes at each light assembly.

The sub-chassis provides the mounting base for the power supply modules and provides the 28 VDC electrical power required for distribution to the plug-in modules. A ground wire is also provided on the chassis for an airframe ground.

With the 28 VDC essential bus energized, strobe circuit breaker pushed in and positioning the ANTI-COLLISION switch at STROBE & ROTR BCN, routes 28 VDC power to the power supplies and timing circuit. The timing circuit is then energized, which provides the pulses to the power supplier to trigger each individual light assembly.

EXTERIOR - MAINTENANCE PRACTICES

1. General

Exterior lighting troubles will usually be of a minor nature. Replacement of defective lamps, switches, and relays, and the repair of faulty wiring is all that should be necessary.

2. Testing Exterior Lights

NOTE: Connect a 28-volt dc external power supply to airplane.

A. Test landing and taxi lights as follows:

(1) Close the following circuit breakers on right-hand forward overhead circuit-breaker panel:

"LANDING & TAXI - LH PWR"

"LANDING & TAXI - LH CONT"

"LANDING & TAXI - RH PWR"

"LANDING & TAXI - RH CONT"

(2) Place "LDG & TAXI LTS" switches at EXTEND and BOTH ON. Both landing lights should illuminate and extend to the taxi position.

(3) Simulate an in-flight condition with nose gear load switch in unloaded position. The landing lights should retract to the land position.

(4) Return nose gear load switch to normal ground position. The landing lights should extend to the taxi position.

(5) Place "LDG & TAXI LTS" switch at LH ON. The right landing lamp should go out, and the left should remain on.

(6) Place "LDG & TAXI LTS" switches at OFF and RETRACT. Both lights should be turned off and should retract flush with the underside of the airplane fuselage.

B. Test anticollision, strobe and position lights as follows:

(1) Close the following circuit breakers on right-hand forward overhead circuit-breaker panel.

"ANTI-COLLISION - UPPER"

"ANTI-COLLISION - LOWER"

"LIGHTS - POSITION"

"STROBE" (Airplanes NA-380-38 and Later)

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(2) On airplanes NA-380-1 through -37. Place "EXTERIOR LIGHTS - ANTI-COLLISION" switch at ON. Both upper and lower anticollision lights should operate.

(3) Place "EXTERIOR LIGHTS - ANTI-COLLISION" switch at OFF. Both anticollision lights should go out.

(3A) On airplanes NA-380-38 and later. Place "EXTERIOR LIGHTS - ANTI-COLLISION" switch at ROTR BCN. Both upper and lower anticollision lights should operate.

(3B) Place "EXTERIOR LIGHTS - ANTI-COLLISION" switch at STROBE & ROTR BCN. Both upper and lower anticollision lights should operate and three strobe lights (one on each wing tip and one on tail cone) should operate.

(3C) Place "EXTERIOR LIGHTS - ANTI-COLLISION" switch at OFF. Both anti-collision lights and three strobe lights should go out.

(4) Place "EXTERIOR LIGHTS - POSITION" switch at OFF. The left and right wing tip lights and taillights should go off.

(5) Place "EXTERIOR LIGHTS - POSITION" switch at OFF. The left and right wing tip lights and taillights should go off.

C. Test ice-check lights as follows:

(1) Close "ICE CHECK" circuit breaker on right-hand console circuit-breaker panel.

(2) Place "EXTERIOR LIGHTS - ICE CHECK" switch at ON. The right and left ice-check lights should come on.

(3) Place "EXTERIOR LIGHTS - ICE CHECK" switch at OFF. Both ice-check lights should go off.

3. Adjusting Landing and Taxi Lights

A. See figure 33-40-201, and adjust landing and taxi lights as follows:

(1) Position target board 10 feet ahead of the landing and taxi lamps hinge point, or 78.48 inches forward of nose of the airplane.

(2) Align the target board with the projected FRP and CL of airplane.

NOTE: Alignment of the target board may be accomplished by using a transit or gunners quadrant.

(3) Disengage "NOSE GEAR-MAIN STEER" and "NOSE GEAR-STBY STEER" circuit breakers. Simulate an in-flight condition with nose gear load switch in unloaded position.

(4) Apply electrical power to airplane and position "LDG & TAXI LTS" switches at EXTEND AND BOTH ON.

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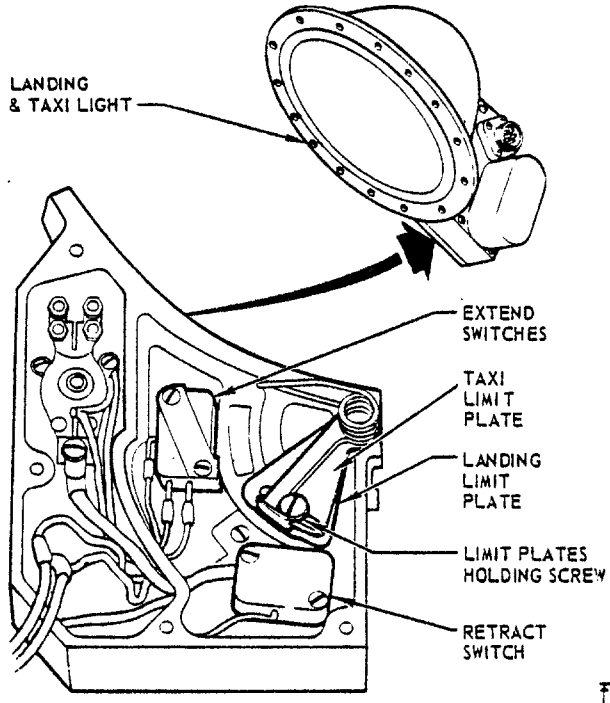

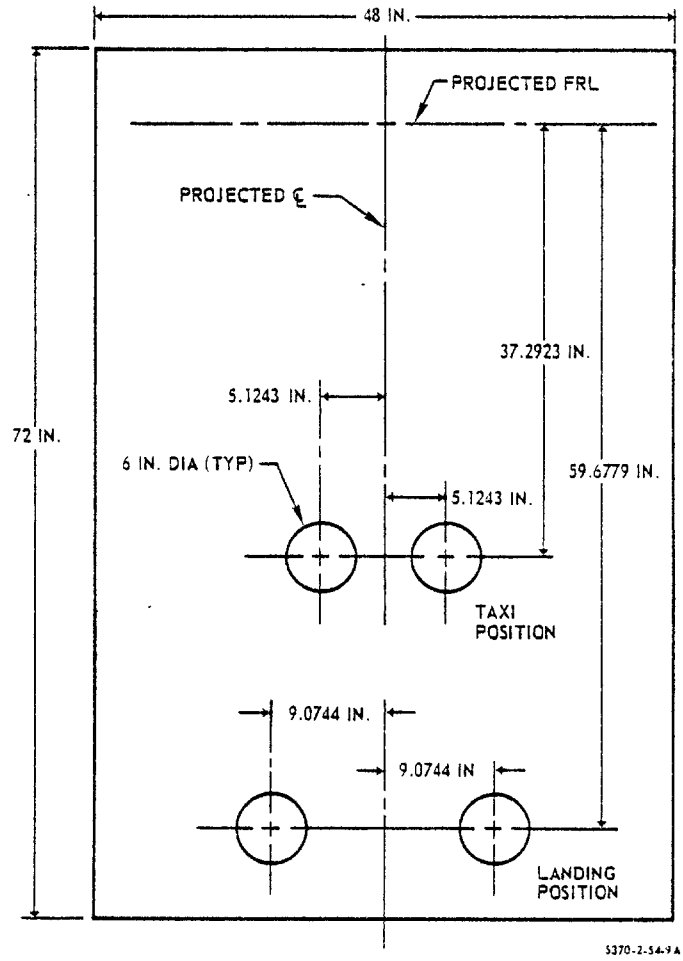

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LANDING AND TAXI LIGHT TARGET BOARD
(NA-370 AND NA-380 AIRPLANES)

Landing and Taxi Light Switch Adjustments
Figure 33-40-201

CAUTION: TO PREVENT LANDING LIGHTS FROM OVERHEATING, TURN LIGHTS OFF WHEN PERFORMING ADJUSTMENTS.

(5) Check positions of landing light beams on target board. The centers of the beams should fall within the landing position circles on target board.

(6) If the light beams are to the left or right of the landing position circles, loosen the mounting screws and manually rotate the light assembly against the mounting screws until the light beam falls into horizontal alignment with the center of the landing position circles on the target board. Tighten mounting screws after alignment.

(7) If the light beams are above or below the landing position circles, remove adjustment cover on side of landing light mechanism housing (side opposite from motor and connector).

(8) Loosen screw holding the landing limit plate and taxi limit plate. Slide the landing limit plate (bottom plate) to desired position so landing light beam is centered on the proper landing light circle of the target board. Temporarily tighten limit plates screw.

NOTE: The extend limit switches require adjustment only if the taxi and landing limit plates interfere with the housing in the retract position.

(9) Connect and safety nose gear load switch linkage.

(10) Check position of taxi light beam on target board. The centers of both beams should fall within the taxi position circles on target board.

(11) If light beams are above or below the taxi position circles loosen screw holding limit plates. Hold the landing limit plate and adjust the taxi limit plate (top plate) so that light beams are centered on taxi light circle of target board. Tighten limit plate screw.

(12) Engage "NOSE GEAR-MAIN STEER" and "NOSE GEAR-STBY STEER" circuit breakers.

(13) Position "LDG & TAXI LTS" switches at RETRACT. The landing and taxi lights should retract until just flush and mechanism is not under strain. Adjust retract position by moving retract limit switch up or down in desired direction.

(14) Operate landing light to ensure light operates satisfactorily at both extremes.

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4. Removing and Installing Strobe Lights

A. Removing LH or RH wing strobe lights.

(1) Remove clear lens from base assembly by removing two screws.

(2) Remove two screws, mounting position light lens retainer to base assembly, and remove lens retainer.

(3) Remove two screws just aft of strobe light and pull base assembly outboard and disconnect electrical connector from strobe light assembly.

(4) Remove strobe light from base assembly by removing two screws.

B. Installing LH or RH wing strobe light.

(1) Install strobe light (Hoskins, P/N 31-0265-1) to base assembly with two mounting screws.

(2) Connect electrical connector to strobe light assembly and safety wire.

(3) Position base assembly on wing tip and install two mounting screws aft of strobe light.

(4) Place position light lens retainer on base assembly and install two mounting screws.

(5) Install clear lens on base assembly with two screws and washers.

(6) After strobe light installation, perform "Test anticollision, strobe and position lights", refer to 33-40-1, step 2, B.

C. Removing tail cone strobe light.

(1) Remove LH tail cone access panel.

(2) Disconnect electric connector from strobe light assembly.

(3) Remove strobe light lens from base assembly by removing one screw.

(4) Remove strobe light base assembly by removing three screws, nuts and washers.

D. Installing tail cone strobe light.

(1) Install strobe light assembly (Hoskins, P/N 30-0002) on tail cone with three mounting screws, two bevel washers (on aft mounting screws, inside of tail cone), flat washers and nuts.

(2) Connect electrical connector to strobe light assembly and safety.

(3) Install strobe light lens on base assembly with one screw and washer.

(4) Install tail cone access panel.

(5) After strobe light installation, perform "Test anticollision, strobe and position lights", refer to 33-40-1, step 2, B.

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5. Removing and Installing Upper Anti-Collision Light Assembly

CAUTION: PLACE AFT FUSELAGE SUPPORT UNDER AFT FUSELAGE. THIS WILL PREVENT POSSIBLE TAIL DAMAGE.

A. Removing Upper Anti-Collision Light Assembly (Grimes, P/N 9950-1)

(1) Remove all electrical power from airplane.

(2) Gain access to top of vertical stabilizer.

(3) Remove red lens from base assembly, by removing two screws and lens retainer (under aft screw).

(4) Remove fillet sealant from periphery of light mounting base with a non-metallic tool.

(5) Remove four screws from mounting base assembly, and pull base assembly upwards and disconnect electrical connector from assembly.

B. Installing Upper Anti-Collision Light Assembly (Grimes, P/N 9950-1)

(1) Remove red lens from base assembly by removing two screws and nut, and lens retainer (under aft screw).

(2) Connect electrical connector and install base assembly to top of vertical stabilizer with four screws.

(3) Install red lens on base assembly with two screws and lens retainer.

(4) Apply fillet seal ("3M" EC801 or equivalent) around the periphery of the base assembly.

(5) After anti-collision light installation, perform "Test Anti-Collision, Strobe and Position Lights", refer to 33-40-1, step 2, B.

6. Removing and Installing Lower Anti-Collision Light Assembly

A. Removing Lower Anti-Collision Light Assembly (Grimes, P/N 9950-1)

(1) Remove all electrical power from airplane.

(2) Remove red lens from base assembly, by removing two screws and lens retainer (under aft screw).

(3) Remove fillet sealant from periphery of light mounting base with a non-metallic tool.

(4) Remove four mounting screws from mounting base assembly and pull downwards and disconnect electrical connector from assembly.

B. Installing Lower Anti-Collision Light Assembly (Grimes, P/N 9950-1)

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(1) Remove red lens from base assembly by removing two screws and nut and lens retainer (under aft screw).

(2) For drainage, remove Allen type set screw from lens center mounting screw.

(3) Connect electrical connector and install base assembly with four mounting screws.

(4) Install red lens on base assembly with two screws and lens retainer.

(5) Apply fillet seal ("3M" EC801 or equivalent) around the periphery of base assembly.

(6) After anti-collision light installation, perform "Test Anti-Collision, Strobe and Position Lights", refer to 33-40-1, step 2, B.

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EMERGENCY LIGHTING - DESCRIPTION AND OPERATION

1. General

The emergency lighting system furnishes emergency lighting for the cockpit area, entrance area, and exit areas when the normal lighting systems become inoperative. During normal night-time operation ("LIGHTS - EMERG" switch at ARM), the emergency lighting system lights are out and the 7.2-volt batteries inside the emergency power units are being charged by dc power from the essential bus. The relays inside the power units are energized during normal operation by the essential bus power that charges the 7.2-volt batteries. The energized relays disconnect battery power from the emergency lights, which prevents the lights from being illuminated during normal operation. If there is a dc power failure, the relays inside the emergency power units are de-energized. The de-energized relays connect the 7.2-volt batteries to the emergency lights. The diodes inside the emergency power units isolate the 7.2-volt batteries from the dc essential bus loads in order to prevent these loads from discharging the power unit batteries. When the "LIGHTS - EMERG" switch is placed at TEST, an airplane dc power failure is simulated. The operation is the same as previously explained. When the "LIGHTS - EMERG" switch is placed at DISARM, the emergency lights are disconnected from the power units. (See figure 33-50-1.)

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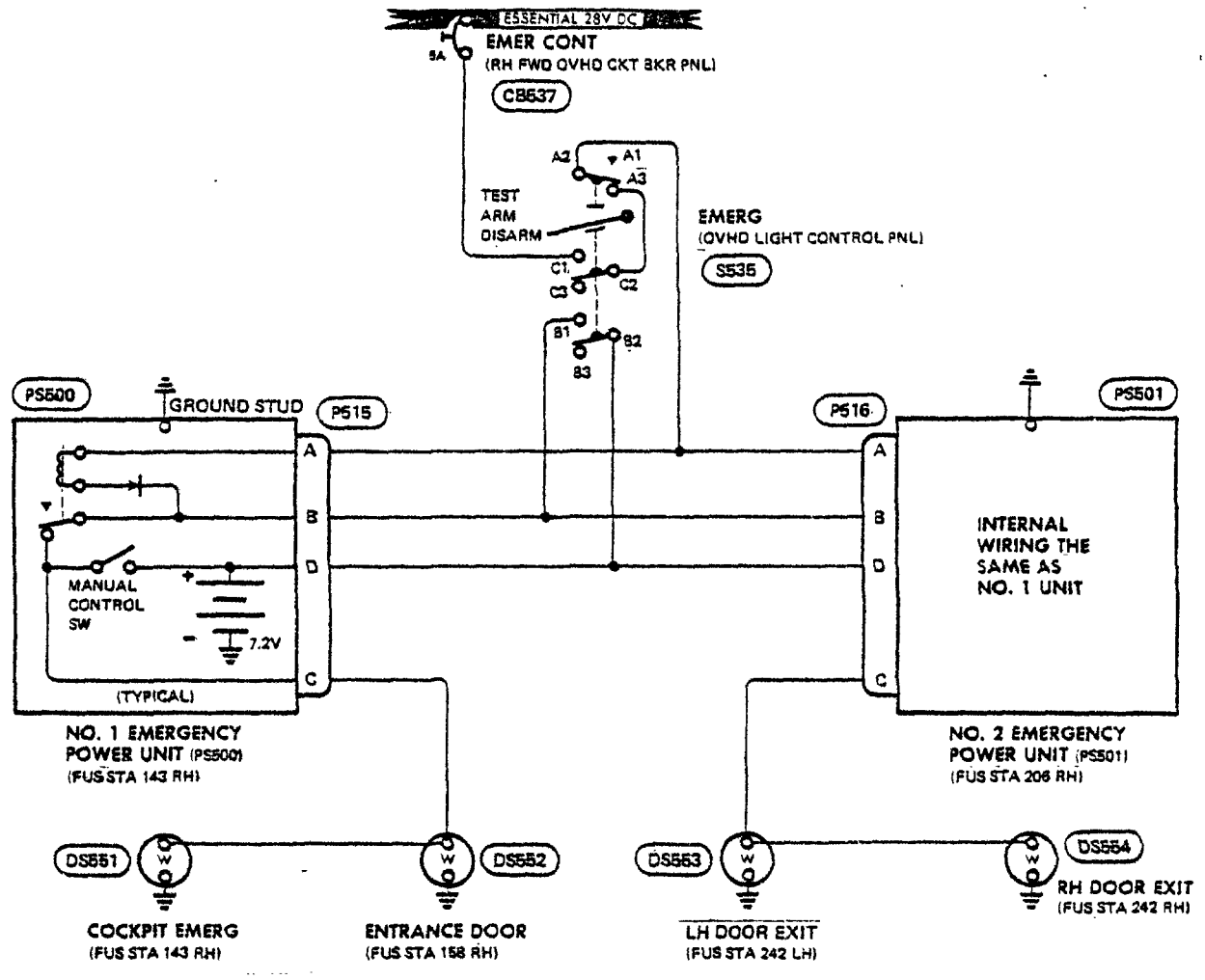

S370-2-54-10A

Emergency Lighting System Schematic
Figure 33-50-1

EMERGENCY LIGHTING

MAINTENANCE PRACTICES

GENERAL

The emergency lights should not be used for any ground maintenance or any purposes other than intended. If the batteries are in a discharged condition, they may be recharged with a 28-VDC source of power. Approximately 16 hours of charging is required if the batteries are in a completely discharged state.

TESTING EMERGENCY LIGHTS

1. Connect 28-VDC external power to airplane.
2. Close EMERG CONTROL circuit breaker on right-hand forward overhead circuit breaker panel.
3. Place LIGHTS—EMERG switch at DISARM position. All emergency lights should be off.
4. Place LIGHTS—EMERG switch at ARM. All emergency lights should be off.
5. Disconnect or turn off external power. All emergency lights should be on.
6. Reconnect or turn on external power. All emergency lights should be off.
7. Place LIGHTS—EMERG switch at TEST and hold. All lights should be on.
8. Place LIGHTS—EMERG switch at DISARM, and disconnect external power.

EMERGENCY LIGHTING BATTERY INSPECTION (S/N 370-1, -7, 380-1 and Subs)

1. Gain access to the No. 1 and No. 2 emergency power units (fuselage station 143 RH and 206 RH).
2. Disconnect connector and remove 4 screws securing the power supply.
3. Remove 4 screws from the case to gain access to the batteries inside the power supply.
4. Inspect the emergency lighting 7.2v batteries for condition and corrosion. Deep cycle batteries or replace as necessary.
5. Reinstall the 4 screws to the emergency power supply case and cover.
6. Reinstall No. 1 and No. 2 emergency power units and all equipment removed to gain access.

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CHAPTER

34

NAVIGATION

CHAPTER 34

NAVIGATION

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NAVIGATION

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NAVIGATION - DESCRIPTION

1. General.

The basic Sabreliner navigational system is limited to the pitot-static system, stand-by compass, free air temperature indicating system, and flame-out and stall warning system. Additional (optional) navigation equipment consists of the weather penetration radar system, distance measuring equipment (DME), dual VHF omnidirectional range receivers (VOR/ILS), the marker beacon receiver, dual automatic direction finders (ADF), dual air traffic control (ATC)transponders, dual flight director systems, dual magnetic compass systems, an altitude alerting system, a comparator warning monitor system, and a radio altimeter. Location of the system components is shown in figure 34-00-1.

NOTE: On NA-370 airplanes, the equipment listed and shown in figure 34-00-1 is not necessarily installed in these airplanes.

For information concerning installation, operation, maintenance, and trouble shooting of the equipment, refer to the manuals listed in paragraph 2. (Instruction manuals combine installation, maintenance, overhaul, and illustrated parts breakdown manuals. System No. 1 is the pilot's; system No. 2 is the copilot's.)

NOTE: If these manuals are not furnished with the equipment, they may be obtained from the Collins Radio Company, Cedar Rapids, Iowa, unless otherwise noted.

2. Navigation Equipment Manuals

A. The following manuals are available for the WP-103G weather penetration radar system:

(1) Pilot's Guide, 523-0242000

(2) Maintenance and Installation Manual, 523-0760726

(a) 537F-7A antenna

- Overhaul Manual, 523-0760919

(b) 493A-5 Brite tube indicator

- Overhaul Manual, 523-0760889

(c) 561G-3/4 control (modified by distributor)

- Overhaul Manual, 523-0759410

(d) 776C-4 synchronizer

- Overhaul Manual, 523-0760549

B. The following manuals are available for the pilot's FD-109H flight director system and the copilot's FD-109F flight director system:

NOTE: The pilot's flight director system is integrated with the Collins AP-104 autopilot system providing displays on the pilot's instrument panel. Some common units are listed in Chapter 22.

(1) Pilot's Guide, 523-0759326

(2) Maintenance and Installation Manual, 523-0759319

(a) 161E-2 No. 1 and No. 2 mode couplers

- Overhaul and Illustrated Parts Breakdown, 523-0759313

(b) 329B-8G No. 1 and No. 2 flight director indicators

- Overhaul Manual, 523-0759268

(c) 331A-8G No. 1 and No. 2 course indicators

- Overhaul Manual, 523-0760251

(d) 332D-11 No. 1 and No. 2 vertical gyros

- Maintenance Manual, 523-0757988
- Overhaul and Illustrated Parts Breakdown, 523-0757989

(e) 344C-1G No. 1 and No. 2 instrument amplifiers

- Overhaul and Illustrated Parts Breakdown, 523-0759833

(f) 345A-7 No. 1 and No. 2 turn rate gyros

- Overhaul Manual, 523-0759314

(g) 562A-5M No. 1 and No. 2 steering computers

- Overhaul and Illustrated Parts Breakdown, 523-0759311

(h) 614E-20D No. 1 mode selector

- Overhaul and Illustrated Parts Breakdown, 523-0759316

(i) 614E-22 remote heading/course selector

- Overhaul and Illustrated Parts Breakdown, 523-0759922

(j) 614E-23 No. 2 mode selector

- Overhaul and Illustrated Parts Breakdown, 523-0759923

C. The following manuals are available for the pilot's and copilot's MC-102 magnetic compass systems:

(1) 332C-4 No. 1 and No. 2 radio magnetic indicators

(a) Installation, Maintenance, Overhaul, and Illustrated Parts Breakdown, 523-0760303

(2) 327C-2 No. 1 and No. 2 slave indicators, 328A-2A No. 1 and No. 2 compass amplifiers, 332E-2 No. 1 and No. 2 directional gyros, and 620359/620783 No. 1 and No. 2 flux detector and compensator

(a) Maintenance Manual, 520-5842004

(b) Installation Manual, 520-5842006

(c) Overhaul Manual, 520-5842003

(d) Illustrated Parts Breakdown, 520-5842005

D. The following manuals are available for the pilot's and copilot's 51RV-1 VOR/ILS systems:

(1) 51RV-1 No. 1 and No. 2 receivers

(a) Maintenance and Installation Manual, 523-0736004

(b) Overhaul and Illustrated Parts Breakdown, 523-0756524

E. The following manual is available for the 51Z-4 marker beacon system:

(1) 51Z-4 marker beacon receiver

(a) Instruction Manual, 523-0755843

F. The following instructions are available for the altitude alerting system:

NOTE: This system is manufactured by Intercontinental Dynamics Corporation.

(1) 16100-2 copilot's reported altitude indicator, 18137B-2 pilot's altitude reminder, 18711B-14 pilot's servoed altimeter indicator, and 18410A-2 altitude alerting tone generator

(a) Maintenance Checkout Sheet, C018410A

G. The following manuals are available for the AL-101 radio altimeter system:

(1) Maintenance and Installation Manual, 523-0757791

(a) 860F-1 radio altimeter receiver-transmitter

- Overhaul and Illustrated Parts Breakdown, 523-0758884

H. The following manuals are available for the dual DF-203 ADF system:

(1) Maintenance and Installation Manual, 523-0755450

(2) Overhaul Manual, 523-0756096

(3) Illustrated Parts Breakdown, 523-0755919

(4) RF Cables Overhaul Manual, 523-0755504

(a) 51Y-4 ADF receiver

- Instruction Manual, 523-0755843

(b) 614L-8 control unit

- Overhaul and Illustrated Parts Breakdown, 523-0755493

(c) 137A-5 fixed loop antenna

- Overhaul and Illustrated Parts Breakdown, 523-0755617

(d) 179J-8 sense antenna coupler

- Overhaul and Illustrated Parts Breakdown, 523-0755502

(e) Electromechanical components

- Overhaul Manual, 523-0757895

I. The following manuals are available for the 54W-1 comparator warning monitor system:

(1) Maintenance and Installation Manual, 523-0759041

(a) 54W-1 comparator warning monitor

- Overhaul and Illustrated Parts Breakdown, 523-0759042

J. The following manuals are available for the 860E-2 distance measuring equipment (DME):

(1) 860E-2 receiver-transmitter

(a) Pilot's Guide, 523-0755915

(b) Maintenance and Installation Manual, 523-0755912

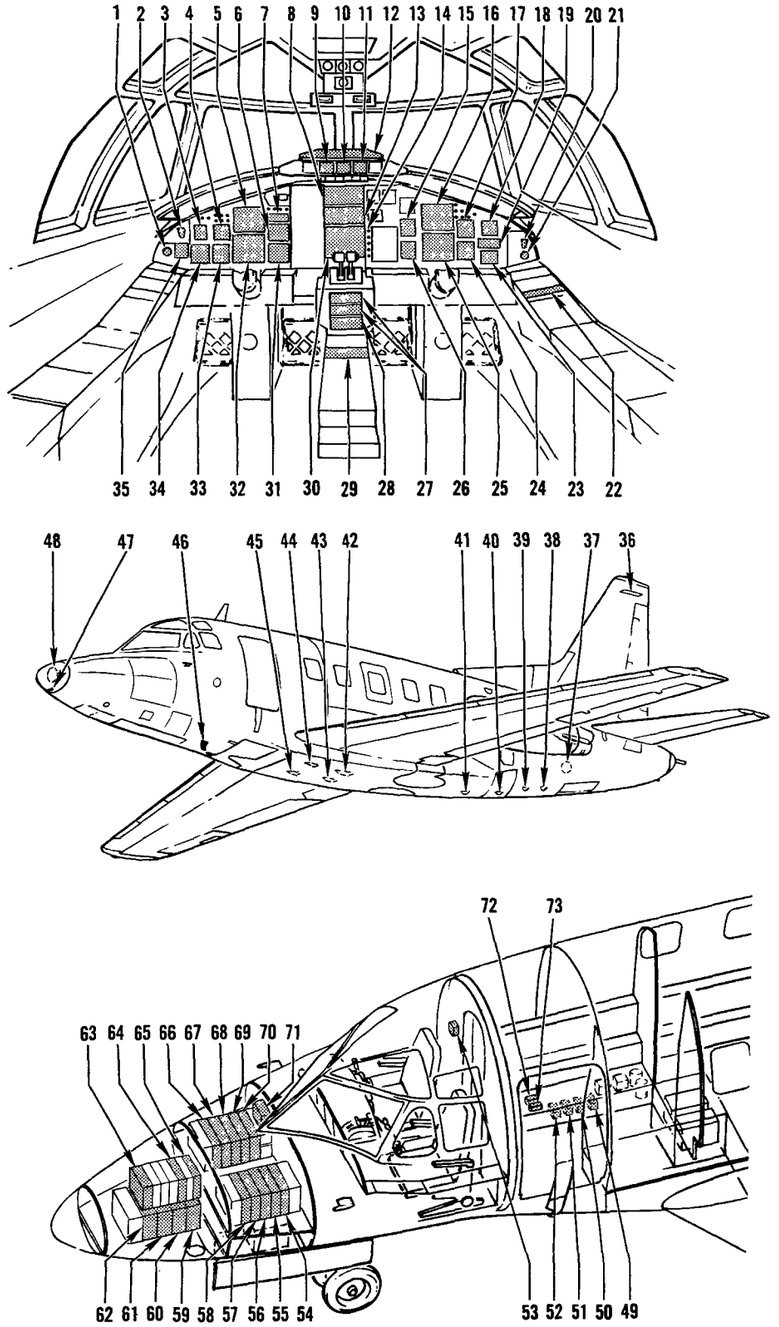
(c) Overhaul and Illustrated Parts Breakdown, 523-0756626

K. The following manuals are available for the air traffic control (ATC) transponder:

(1) 621A-6 transponder

(a) Maintenance and Installation Instruction, 523-0759617

(b) Overhual and Illustrated Parts Breakdown, 523-0759616



1. COMPASS SLAVE INDICATOR
2. COMPASS SLAVE SWITCH
3. CLOCK
4. MACH AIRSPEED INDICATOR
5. FLIGHT DIRECTOR INDICATOR
6. SERVOED DRUM ALTIMETER
7. ALTITUDE REMINDER
8. NO. 1 ADF CONTROL
9. NO. 1 VOR CONTROL
10. ATC TRANSPONDER CONTROL
11. NO. 2 VOR CONTROL
12. COMM/NAV SWITCH PANEL
13. NO. 2 ADF CONTROL
14. COMPARATOR MONITOR LIGHTS
15. MACH AIRSPEED INDICATOR
16. FLIGHT DIRECTOR INDICATOR
17. ALTIMETER PRESSURE
18. CLOCK
19. REPORTED ALTITUDE INDICATOR
20. COMPASS SLAVE SWITCH
21. COMPASS SLAVE INDICATOR
22. CO-PILOTS FLIGHT DIRECTOR MODE SELECTOR PANEL
23. ACCELEROMETER
24. VERTICAL SPEED INDICATOR
25. COURSE INDICATOR
26. RADIO MAGNETIC INDICATOR
27. PILOTS FLIGHT DIRECTOR MODE SELECTOR PANELS
28. AUTOPILOT CONTROLLER PANEL
29. WEATHER RADAR CONTROL PANEL
30. WEATHER RADAR INDICATOR
31. VERTICAL SPEED INDICATOR
32. COURSE INDICATOR
33. RADIO MAGNETIC INDICATOR
34. RADIO ALTIMETER INDICATOR
35. ACCELEROMETER
36. VOR/LOCALIZER ANTENNA
37. NO. 1 DIRECTIONAL GYRO
38. ATC ANTENNA
39. MARKER BEACON ANTENNA
40. RADIO ALTIMETER ANTENNA
41. RADIO ALTIMETER ANTENNA
42. NO. 1 ADF LOOP ANTENNA (LH)
43. NO. 2 ADF LOOP ANTENNA (RH)
44. NO. 1 ADF SENSE ANTENNA (LH)
45. NO. 2 ADF SENSE ANTENNA (RH)
46. DME ANTENNA
47. GLIDE SLOPE ANTENNA
48. RADAR ANTENNA
49. NO. 1 DIRECTIONAL GYRO
50. NO. 2 DIRECTIONAL GYRO
51. NO. 1 TURN RATE GYRO
52. NO. 2 TURN RATE GYRO
53. ALTITUDE ALERTER TONE GENERATOR
54. NO. 1 VOR/ILS RECEIVER
55. NO. 1 ADF RECEIVER
56. NO. 1 INSTRUMENT AMPLIFIER
57. NO. 1 COMPASS AMPLIFIER
58. WARNING MONITOR
59. DME RECEIVER TRANSMITTER
60. NO. 2 ATC TRANSPONDER
61. RADAR RECEIVER TRANSMITTER
62. RADAR SYNCHRONIZER
63. MARKER BEACON RECEIVER
64. NO. 1 STEERING COMPUTER
65. NO. 1 ATC TRANSPONDER
66. NO. 2 VOR/ILS RECEIVER
67. NO. 2 ADF RECEIVER
68. NO. 2 STEERING COMPUTER
69. NO. 2 INSTRUMENT AMPLIFIER
70. NO. 2 COMPASS AMPLIFIER
71. NO. 2 MODE COUPLER
72. AUDIO JUNCTION BOX
73. INSTRUMENT SWITCHING UNIT

S370-2-72-1

FLIGHT ENVIRONMENT DATA - DESCRIPTION

1. Pitot and Static Pressure Systems - Airplanes NA-370-2 Thru -6

The pitot-static systems furnish pitot (impact) pressure and static (atmospheric) pressure to the pressure-operated flight instruments and to other systems requiring pitot and static pressures for operation. (See figure 34-10-1.) Both pitot pressure and static pressure operate the mach/airspeed indicator, airspeed and altitude warning switches, and air data computer. Instruments operated by static pressure only are the rate-of-climb indicator, altimeter, and cabin altimeter and differential pressure indicator. An independently operated pitot-static system is provided for each set of pilot and copilot instruments. (See figure 34-10-1.) Alternate static-pressure sources, common to both sets of instruments, are also provided. A positive-positioning toggle-type selector valves, on both and left and right subpanels, are provided for transferring from primary static-pressure source to the alternate static-pressure source. Dual, electrically heated pitot-static pressure heads are installed, one on each side of the fuselage.

Pitot pressure is sensed at the cup-shaped recess in the pitot-static head tip. Pitot pressure (impact pressure), which is built up inside the pitot head tip, is transmitted through pitot lines to the various instruments and systems. Static pressure is sensed through holes flush with the pitot-static head and routed through static lines to the various instruments. Drain lines, for each alternate static system, collect moisture that may condense inside the static lines, and route it overboard. Normal static and pitot system lines drain through the pitot-static head. The pitot-static head anti-icing system prevents ice from forming on the pitot-static heads by heating the heads with electrical heating elements.

CAUTION: DO NOT SCRATCH, NICK, DENT, OR IN ANY OTHER WAY DAMAGE PITOT-STATIC HEADS. KEEP PITOT-STATIC HEADS FREE OF DIRT. INSTALL DUST COVERS WHEN AIRPLANE IS PARKED.

1A. Pitot and Static Pressure Systems - Airplanes NA-370-1, -7, and NA-380-1 and Later

The Pitot-Static Probes and the Selector valves allow both the pilot and copilot to select either the right probe, the left probe or both probes. The three valve positions are labeled L, R, and BOTH. For all normal operations, the valves are switched to the BOTH position. (See figure 34-10-1A.)

The right and left compensated pitot heads each have two static ports, Static No. 1 and Static No. 2. With the pilot's static source selector switch in the BOTH position, the pilot's flight instrument static source is obtained from the left Static No. 1 and the right Static No. 2. With the copilot's static selector switch in the BOTH position, the copilot's flight instrument static source is obtained from the left Static No. 2 and the right Static No. 1. The L and R positions are provided to select a single static source and are used in the event of either pitot head becoming inoperative (damage from bird strike, icing, etc.). The pilot's L position selects the Static No. 1 on the left pitot head, and the R position selects Static No. 2 on the right pitot head. The copilot's L position selects the Static No. 2 on

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the left pitot head, and the R position selects Static No. 1 on the right pitc head.

Pitot pressure is sensed at the cup-shaped recess in the pitot-static head tip. Pitot pressure (impact pressure), which is built up inside the pitot head tip, is transmitted through pitot lines to the various instruments and systems. Static pressure is sensed through holes flush with the pitot-static head and routed through static lines to the various instruments. Drain lines, for each alternate static system, collect moisture that may condense inside the static lines, and route it overboard. Normal static and pitot system lines drain through the pitot-static head. The pitot-static head anti-icing system prevents ice from forming on the pitot-static heads by heating the heads with electrical heating elements.

CAUTION: DO NOT SCRATCH, NICK, DENT, OR IN ANY OTHER WAY DAMAGE PITOT-STATIC HEADS. KEEP PITOT-STATIC HEADS FREE OF DIRT. INSTALL DUST COVERS WHEN AIRPLANE IS PARKED.

2. Mach and Airspeed Audible Warning System

A Mach and airspeed sensor switch (figure 34-10-2) is on the fuselage left side at station 78, and is connected to the pilot's pitot-static system. A horn and an audible signal unit (for crew headsets) are on the forward side of the fuselage station 143 bulkhead, just behind the pilot. The Mach and airspeed sensor switch supplies 28 volts dc to the horn and audible signal unit. The Mach and airspeed sensor switch is set to sound the warning horn and supply audible signals to headsets when the airplane is flying at certain correlated conditions of airspeed and altitude, approaching unsafe limits. These limits are a function of the true Mach number limitation of the airplane. The true Mach number limitation of the airplane is indicated by the "maximum allowable" pointer on the Mach/airspeed indicator. The sensor switch is a sealed unit and cannot be adjusted in the field. A push-to-test switch ("MACH-A/S WARNING TEST") is installed on the pilot's main outboard instrument subpanel.

The audible warning system is activated when the airplane is flying below an indicated altitude of 21,210 feet, at an indicated airspeed of 370 (±3) knots; or flying above an indicated altitude of 21,210 feet and reaching the true Mach number limitation of 0.800.

3. Free Air Temperature Indicating System

The free air temperature indicating system includes the temperature indicator, temperature bulb, and electrical wiring connected through the "FREE AIR TEMP" circuit breaker to the 28-volt dc essential bus. The bezel-mounted free air temperature indicator, indicates the outside air temperature as sensed by the temperature bulb on the fuselage at station 113. The indicator is an electrically operated resistance thermometer using a moving coil movement. It is graduated from -70°C to 80°C. When 90.38 ohms resistance is connected (by the temperature bulb or other means) between the sensing connection and ground with 28-volt dc on the power terminal, the indicator should be zero. The sensing element in the temperature bulb uses changes in resistance with

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the change of temperature to which it is exposed. This variable resistance actuates the indicator to give temperature indication. The resistance of the bulb is normally 97 to 101 ohms at a temperature of 20°C to 30°C (68°F to 86°F).

4. Stall Warning System

A. Description and Operation - NA-370-2 Through -6, -8 and -9 Airplanes

The stall warning system warns the pilot or copilot when the airplane approaches stall speed or when the airplane approaches a speed which may induce engine flame-out. The warning system operating point is biased by the position of the wing flaps. The warning is accomplished by vibrating both the pilot's and copilot's control column with the control shakers. The system consists of two lift transducers, a flap monitor switch, a signal summing unit, two control column shakers, and associated electrical control circuits. (See figure 34-10-3.)

A "STALL WARNING TEST" switch, on the pilot's instrument subpanel, is provided for operational check-out of the system on the ground or in the air. The switch provides a means of bypassing the nose gear and main gear load switches when the airplane is on the ground.

B. Description and Operation - NA-370-1, -7, and NA-380 Airplanes

The stall warning system warns the pilot and copilot when the airplane approaches a stall speed or when the airplane approaches a critical attitude and/or a speed which may induce an engine stall. The warning system operating point is biased by the position of the wing flaps. The stall warning indication is accomplished by vibrating both the pilot's and copilot's control column with control shakers, an audible warning and a flashing red stall warning light. The system consists of four lift transducers (2 on each wing), a flap monitor switch, two signal summing units, two control column shakers, two flasher units, two warning lights, two engine bleed valve actuated switches (one for each engine), two warning horns and the associated relays and electrical wiring to complete the circuit.

A stall warning test switch is provided for operational checkout of the aircraft stall system on the ground or in the air. This switch provides a means of bypassing the nose gear and main gear load switches when the airplane is on the ground. To perform an operational check of the engine stall warning system, the engines must be operating above 85% rpm.

As the RH and LH systems are identical in operation, only the LH system operation will be covered in this section.

With the weight of the airplane on the landing gear, the main landing gear load monitor relay (K732) is energized. (See figure 34-10-3A.)

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This removes the ground circuit from the LH rotation stall warning relay (TM509). During takeoff, as soon as the main landing gear load switches are actuated, relay K732 is de-energized, which in turn completes a ground circuit to the rotation stall warning relay (TM509). After approximately 3 seconds relay TM509 energizes providing a ground circuit for the summing unit which activates the stall warning system. The time delay of TM509 eliminates a stall warning indication (shakers or horn) on initial rotation (lift-off) of the airplane. After the stall warning system has been actuated and when the aircraft is in a stall condition, a signal from the transducers in the wing is applied to the summing unit. This signal is amplified and routed to the stall monitor relay coil (K735) and the continuous ignition relay coil (TM507). Relay K735 and TM507 are then energized which in turn activates the engine ignition system and the control stick shakers. Preceding (approximately 2 seconds) the stick shaker operation, a signal from the summing unit is applied to the engine stall warning light and audible warning horn. This warn-in alerts the flight crew of an engine stall condition. As the aircraft attitude is changed to correct for the stall condition, the stall warning transducers remove the stall signals from the summing unit, which in turn de-actuates the control stick shakers, engine stall warning indicator light and warning horn, subsequently returning the system to normal operation. The continuous ingition relay TM507) will remain energized for approximately 3 seconds after the stall warning condition is cor-rected. This time delay keeps the engine exciters energized (provides engine ignition) which ensures adequate time to fully recover from the stall condition.

5. Mach/Airspeed Indicator

A Mach/airspeed indicator, on the pilot's outboard instrument panel, provides airspeed indication and Mach number. A single pointer indicates the Mach number on a rotating dial, and airspeed on a fixed dial. A maximum allowable pointer is positioned by airspeed limit at lower altitudes and Mach number limit at the higher altitudes. The Mach/airspeed indicator incorporates in-tegral lighting.

6. Accelerometer

An accelerometer, on the pilot's and copilot's outboard instrument panel, in-dicates the load on the structure of the airplane in terms of gravitational (G) units. The accelerometer indicates the load along the vertical axis of the airplane. Three pointers inform the pilot as to the following: The main pointer continuously indicates gravitational load. It indicates 1 G when the airplane is in level flight. The pointer indicates positive G when the air-plane accelerates upward. The pointer indicates negative G when the airplane accelerates downward. The other two pointers indicate the greatest positive and greatest negative acceleration, respectively, of the airplane during maneuvers. By truning the knob on the lower left of the instrument, the pilot can at any time reset the positive and negative pointers to normal. The ac-celerometer is a bezel-mounted instrument, incorporating integral lighting.

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7. Cabin Altimeter and Differential Pressure Indicator

The cabin altimeter and differential pressure indicator, in the center pedestal, is vented both to outside air pressure and to pressure within the cabin. These pressures drive two corresponding pointers, each traveling over a separate arc of calibrations. The outer pointer shows cabin pressure altitude. The inner pointer, fixed to a rotating band, shows both cabin altitude and differential pressure (between cabin and outside air) in pounds per square inch. The outer pointer will normally read between zero and 8000 feet. As altitude increases, the differential pointer reading will increase until at 45,000 feet (cabin pressure pointer at 8000 feet), a differential of 8.8 psi will be indicated, provided this pressure schedule is maintained by the cabin pressure regulating system. Refer to Chapter 21 for information pertaining to air conditioning and pressurization systems.

8. Cabin Rate-of-Climb Indicator

The rate-of-climb indicator, on the center pedestal, shows the rate of ascent or descent of the airplane cabin in feet per minute. The instrument is actuated by the changes in atmospheric pressure within the cabin that come with a change in altitude. A screw in the left corner of the case permits the instrument pointer to be set to zero when the airplane is on the ground. The instrument is bezel-mounted and incorporates integral lighting. Static pressure for operation of the instrument is obtained by venting the static port to cabin pressure.

9. Lift Transducers

On NA-370-2 through -6, -8 and -9 airplanes, two transducers are installed on the lower right wing near the leading edge at approximately the center of the wing. The vane of the inboard transducer is swep 7.5 degrees greater than the wing leading edge, while the vane of the outboard transducer is swept 20 degrees greater than the wing leading edge. On NA-370-1, -7 and NA-380 airplanes, an additional set of transducers is installed on the lower left wing near the leading edge at approximately the center of the wing.

The lift transducers, in the stall warning system, consists of a vane which is spring-loaded to a central position, and a transformer with an E-type core. The excitation winding, on the central core of the transformer, is provided with ac voltage from the summing unit. The output signal windings are wound on the outer cores. The vane moves an armature across the face of the poles of the E-type transformer which changes the magnetic coupling between the excitation winding and the output windings. This provides the lift signal that is sent to the summing unit.

A heater is provided in the transducer for maintaining a constant viscosity of the damping fluid and deicing the vane in flight. Power for the heater is obtained from the 28-volt dc essential bus, and is controlled by the "PITOT & STALL WNG" switch, main gear load monitor relay, and stall transducer heat relay. The heaters on both transducers operate at reduced voltages when the airplane is on the ground.

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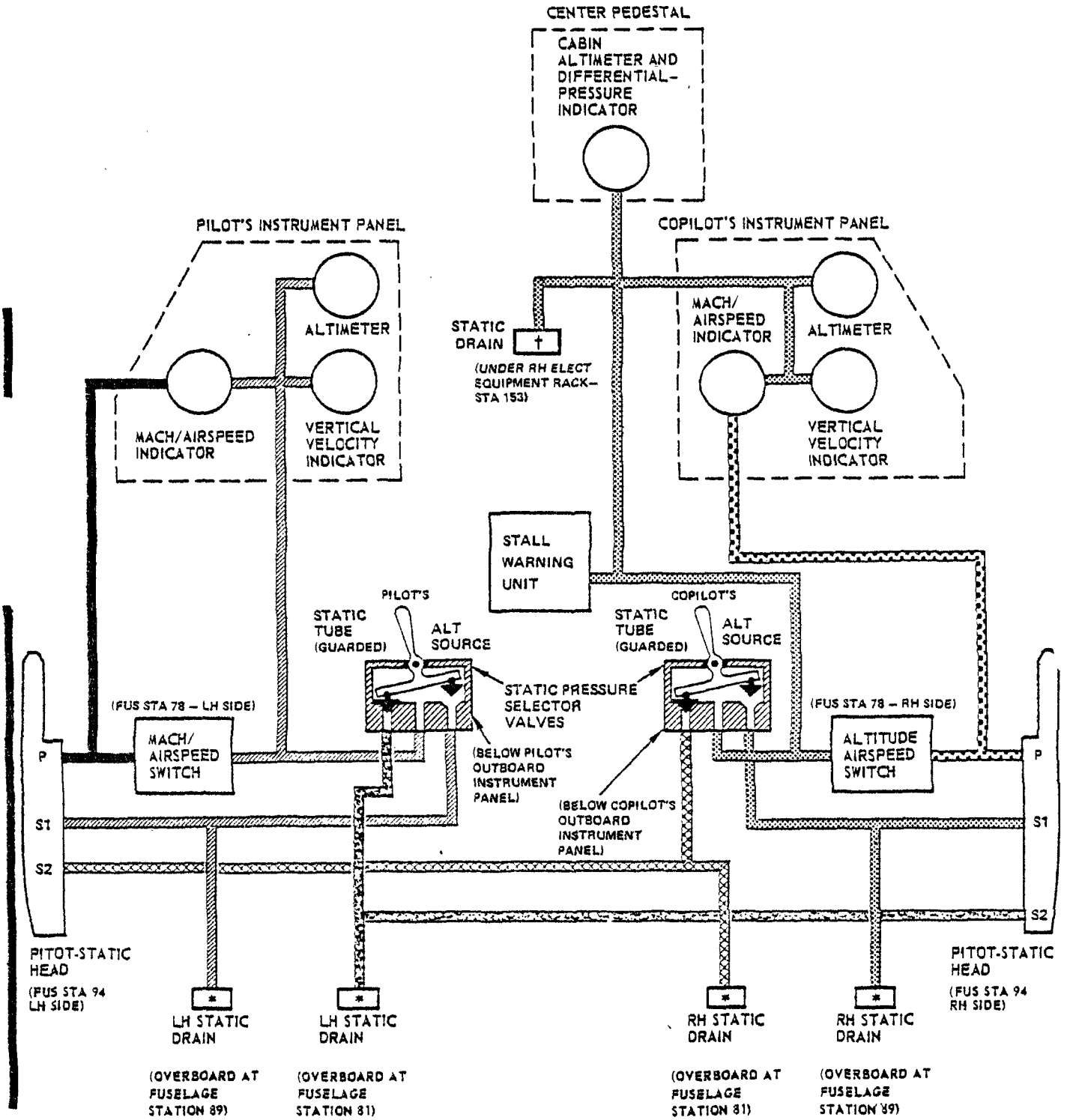

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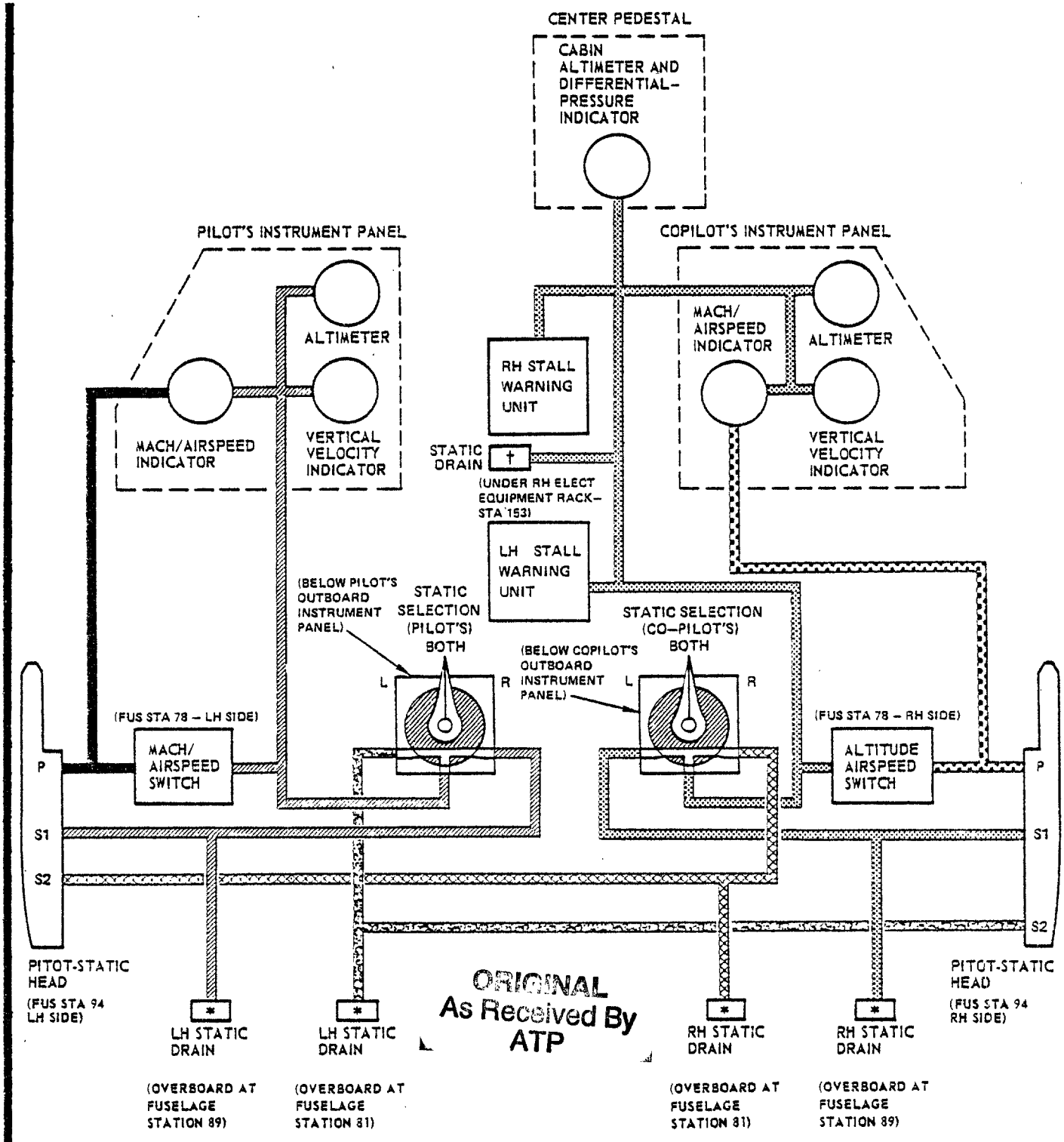
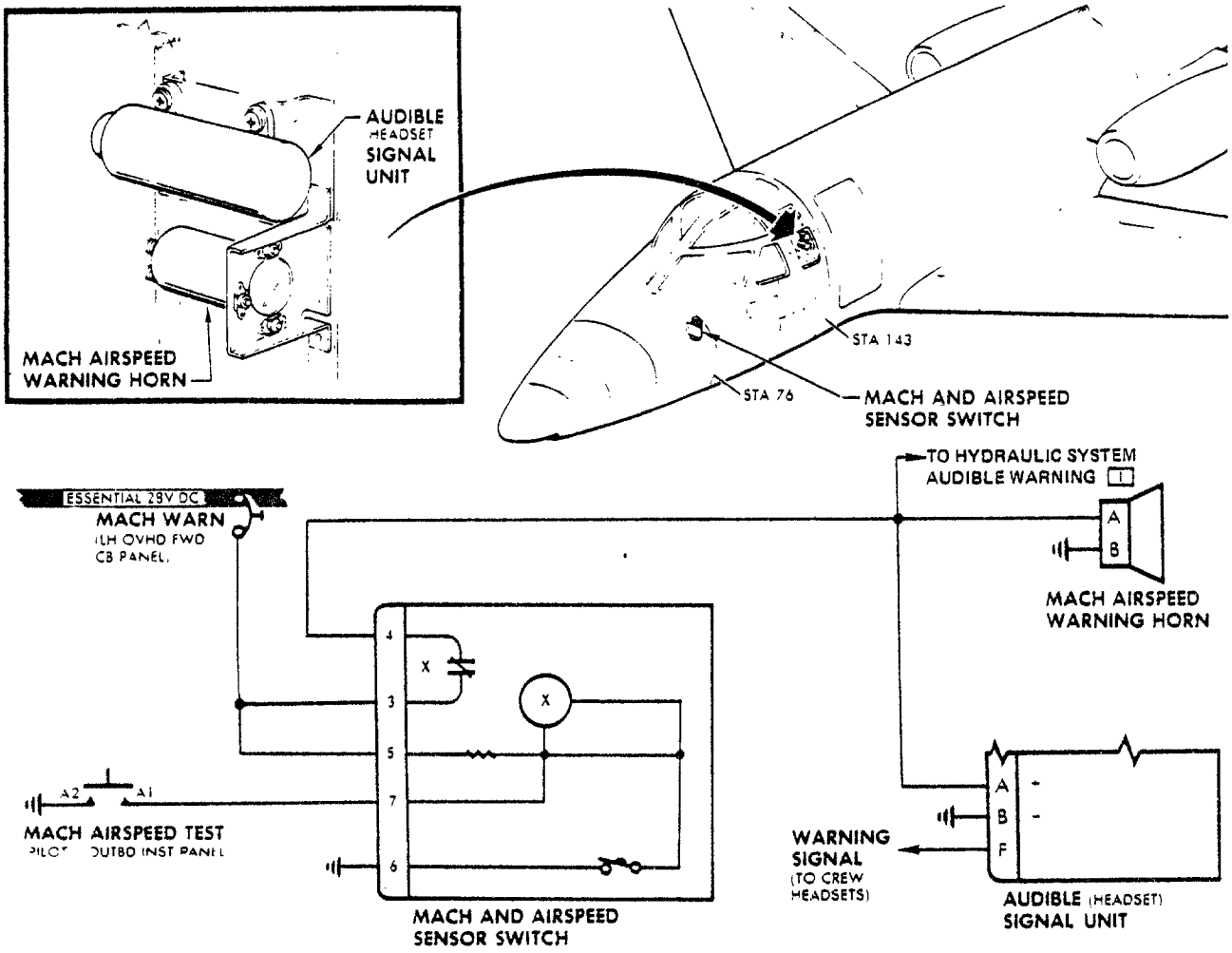
Pitot-Static System Schematic
Airplanes NA-370-2 Thru -6,-8, and -9
Figure 34-10-1



Pitot-Static System Schematic-Airplanes NA-370-1 and -7 and NA-380-1 and Later
Figure 34-10-1A



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[1] NA380-1 AND LATER AIRPLANES

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Mach and Airspeed Audible Warning System
Figure 34-10-2

10. Control Column Shaker

The control column shaker, in the stall warning system, is a motor-driven unit on the outboard side of the pilot's and copilot's control column. The unit consists of a coaxially mounted 28-volt dc motor and an eccentrically weighted ring assembly. Vibrations of the weighted ring are transferred to the control column to give the shaking action.

11. Signal Summing Unit - Aircraft NA-370-2 Through -6, -8 and -9

The signal summing unit, in the stall warning system, is on the electrical equipment rack at fuselage station 195. The signal summing unit receives signals from the lift transducers, provides ac voltage to the excitation windings of the lift transducers, and controls the 28-volt dc that is sent to the motor of the control column shakers.

The signals received from the two lift transducers are mixed in the signal summing unit and controlled by an aneroid potentiometer. The potentiometer is so adjusted as to provide stall warning only below 15,000 feet, and warning of an impending engine flame-out only above 42,000 feet. Between these two altitudes, the mixture is controlled by the mixing circuit within the signal summing unit. The resultant signal is then used to actuate the control stick shakers.

11A. Signal Summing Units - Aircraft NA-370-1, -7, NA-380-1 and Later

On NA-380-Airplanes, the signal summing units in the stall warning system are on the electrical equipment rack at fuselage station 150. The signal summing units receive signals from the lift transducers, provide ac voltage to the excitation windings of the lift transducers, and control the 28-volt dc power that is routed to the motor of the control column shakers and to the engine stall warning lights and horns. (See figure 34-10-3A.)

11B. Engine Bleed Valve Actuated Stall Warning Horn Inhibitor Switches - Aircraft NA-370-1, -7, and NA-380-1 and Later

The engine bleed valve actuated stall warning horn inhibitor switches (S1031 and S1032) are installed on the right and left hand engine. The switches are actuated open whenever the bleed valves are open. (See figure 34-10-3A.) When the switches are actuated open, the engine stall warning flasher units and warning horns are rendered inoperative. At high engine power settings (approximately 85% rpm) the switches are closed, which completes the electrical circuit to the flasher units and warning horns. If at this time an engine stall condition is approached (as sensed by the wing transducer), the warning lights will flash and the warning horn will sound. In a stall condition, this warning will follow the control column shaker warning by approximately 2 seconds.

12. Flap Travel Monitor Switch

The flap travel monitor switch, in the stall warning system, is a rotary-type microswitch in the flap well just outboard of the right main gear. The switch is connected to the flap and follows the movement of the flap travel.

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The primary purpose of the switch is to decrease the speed at which a stall warning occurs when the flaps are extended. The switch is actuated at 13° (±30')*or 11(±30')† degrees of entended flap travel. When actuated, a compensating circuit within the signal summing unit maintains calibration of the system at this position. When flaps are retracted beyond 13°(±30')* or 11(±30')† degrees, the switch causes the system to return to normal.

13. Altitude Reminder

An altitude reminder visually and aurally advises the pilot when a preselected altitude is approached, by lighting a light on the instrument panel and sounding a horn for 2 seconds. The altitude select knob, on the instrument panel, may be turned until the desired altitude is read on a counter. With the mode switch on seek, the light will come on and the horn will sound 1000 feet before the selected altitude is reached, alerting the pilot. The horn sounds for 2 seconds, and the light stays on until the airplane is within 300 feet of the selected altitude. With the mode switch positioned on hold, the light will come on and the horn will sound for 2 seconds if the airplane deviates up or down from the selected altitude by 300 feet. The light will stay on as long as the mode switch is left on hold, unless the selected altitude is regained within 300 feet.

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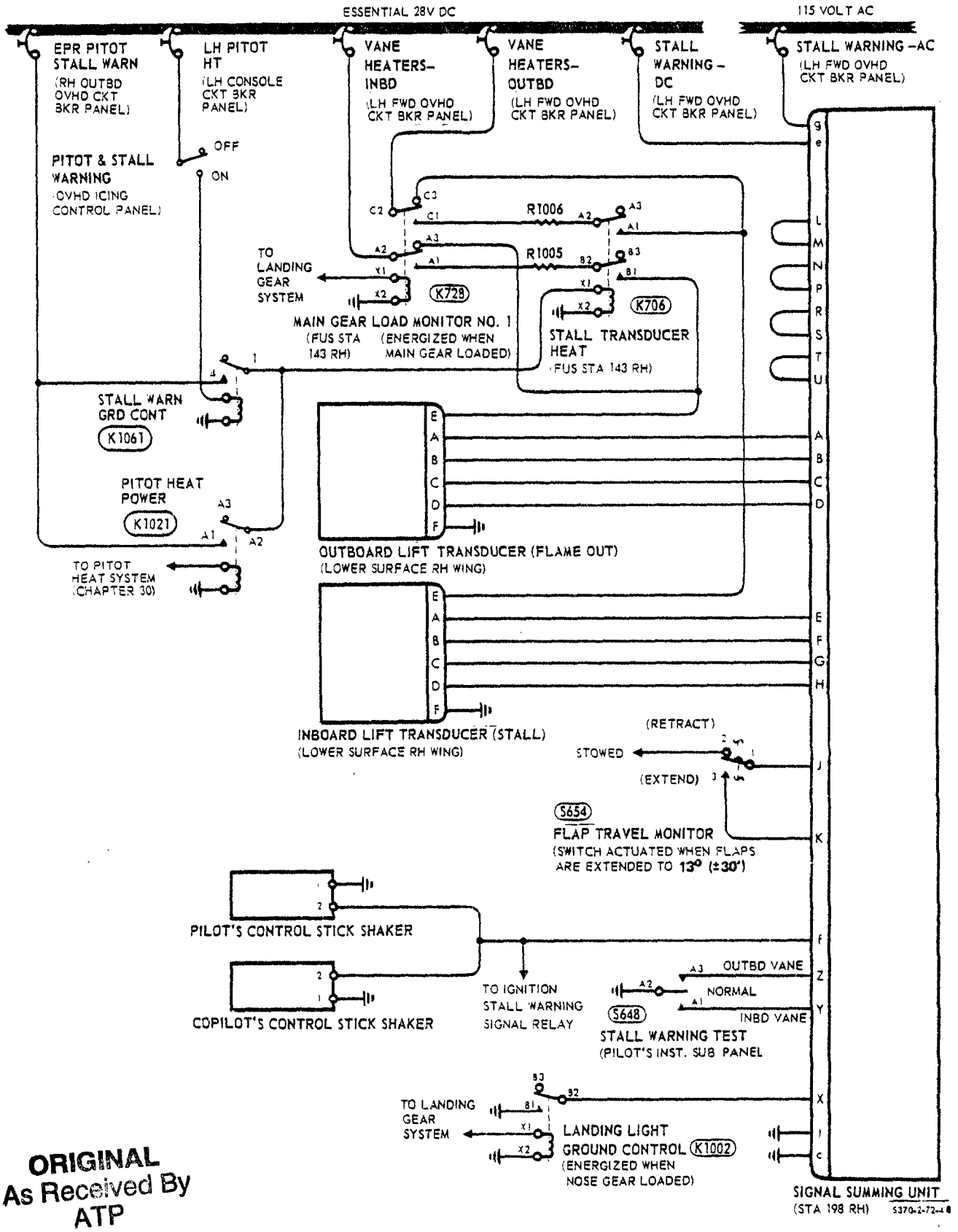
* Airplanes Serial No. NA-370-2 thru -6
† Airplanes Serial No. NA-370-1, -7 and later and NA-380-1 and Later airplanes


ESSENTIAL 28V DC
115 VOLT AC
EPR PITOT STALL WARN (RH OUTBD OVHD CKT BKR PANEL)
LH PITOT HT (LH CONSOLE CKT BKR PANEL)
VANE HEATERS-INBD (LH FWD OVHD CKT BKR PANEL)
VANE HEATERS-OUTBD (LH FWD OVHD CKT BKR PANEL)
STALL WARNING-DC (LH FWD OVHD CKT BKR PANEL)
STALL WARNING-AC (LH FWD OVHD CKT BKR PANEL)
PITOT & STALL WARNING (OVHD ICING CONTROL PANEL)
OFF
ON
R1006
R1005
TO LANDING GEAR SYSTEM
K728
MAIN GEAR LOAD MONITOR NO. 1 (FUS STA 143 RH) (ENERGIZED WHEN MAIN GEAR LOADED)
K706
STALL TRANSDUCER HEAT (FUS STA 143 RH)
STALL WARN GRD CONT
K1061
PITOT HEAT POWER
K1021
TO PITOT HEAT SYSTEM (CHAPTER 30)
OUTBOARD LIFT TRANSDUCER (FLAME OUT) (LOWER SURFACE RH WING)
INBOARD LIFT TRANSDUCER (STALL) (LOWER SURFACE RH WING)
(RETRACT)
STOWED
(EXTEND)
S654
FLAP TRAVEL MONITOR (SWITCH ACTUATED WHEN FLAPS ARE EXTENDED TO 13° (±30'))
PILOT'S CONTROL STICK SHAKER
COPILOT'S CONTROL STICK SHAKER
TO IGNITION STALL WARNING SIGNAL RELAY
OUTBD VANE
NORMAL
INBD VANE
S648
STALL WARNING TEST (PILOT'S INST. SUB PANEL)
TO LANDING GEAR SYSTEM
LANDING LIGHT GROUND CONTROL K1002 (ENERGIZED WHEN NOSE GEAR LOADED)
SIGNAL SUMMING UNIT (STA 198 RH)
S370-2-72-4B

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Flame-out and Stall Warning Schematic -
Aircraft NA-370-2 Through -6, -8 and -9
Figure 34-10-3

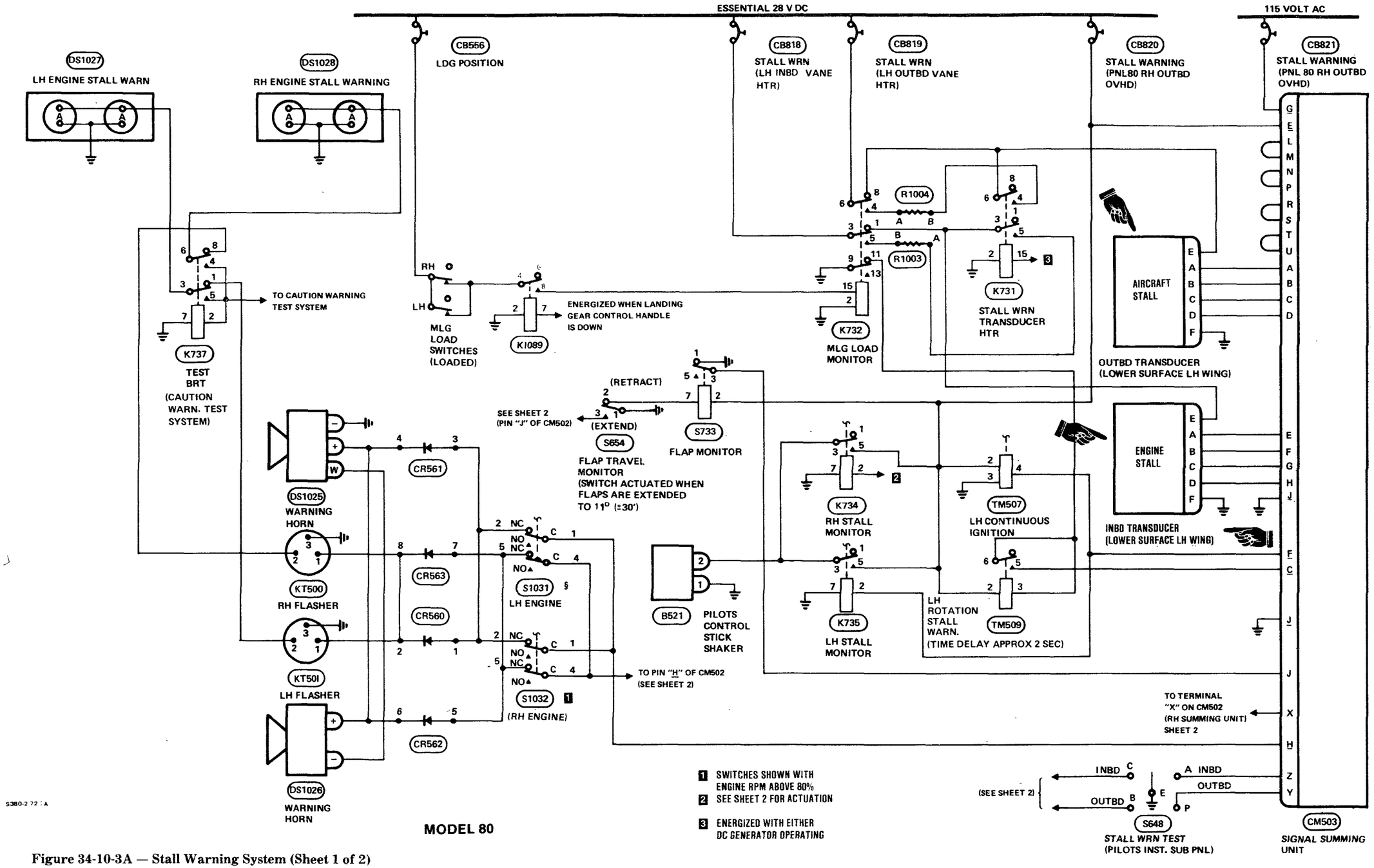

Figure 34-10-3A — Stall Warning System (Sheet 1 of 2)

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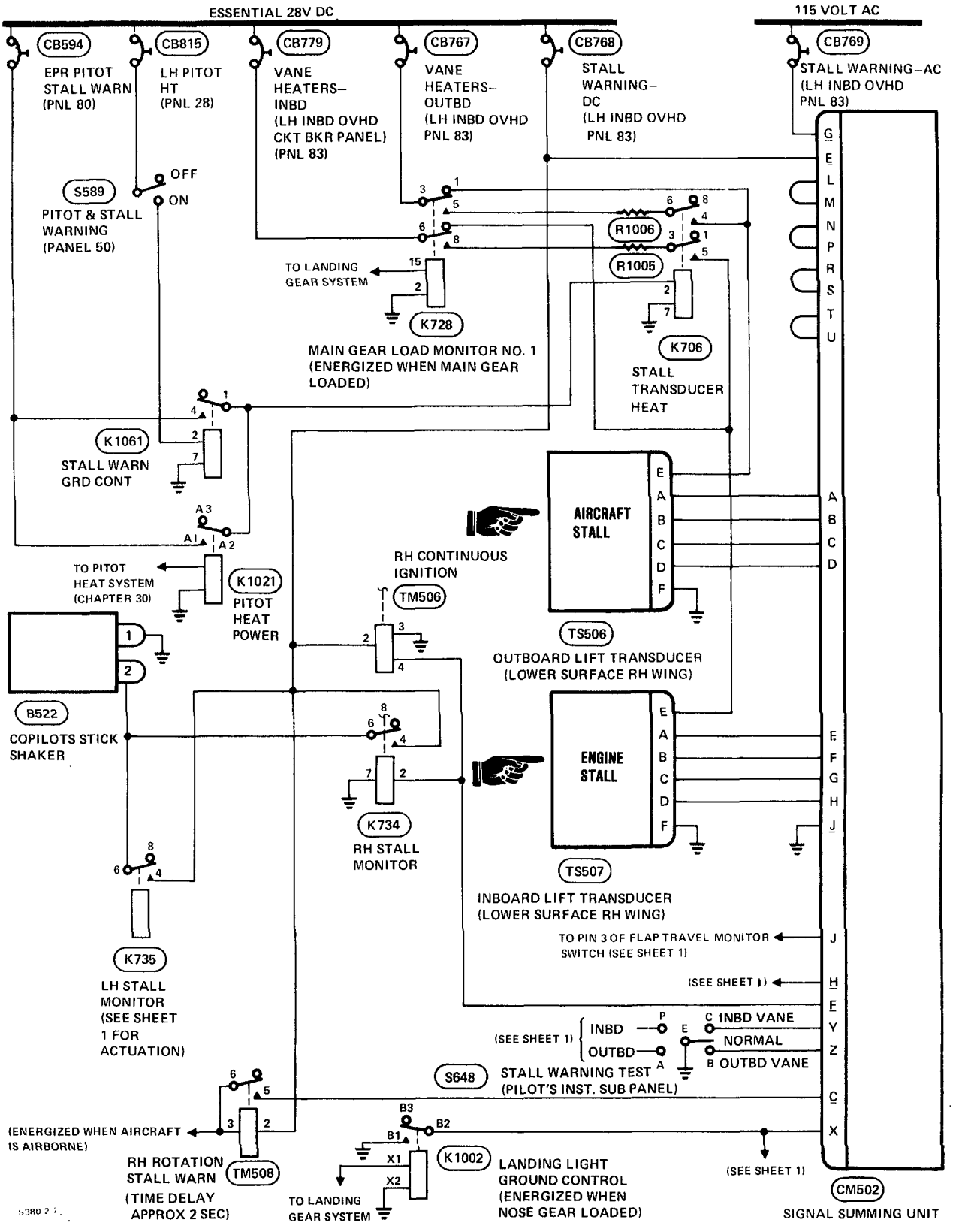

MODEL 80

Figure 34-10-3A — Stall Warning System (Sheet 2 of 2)

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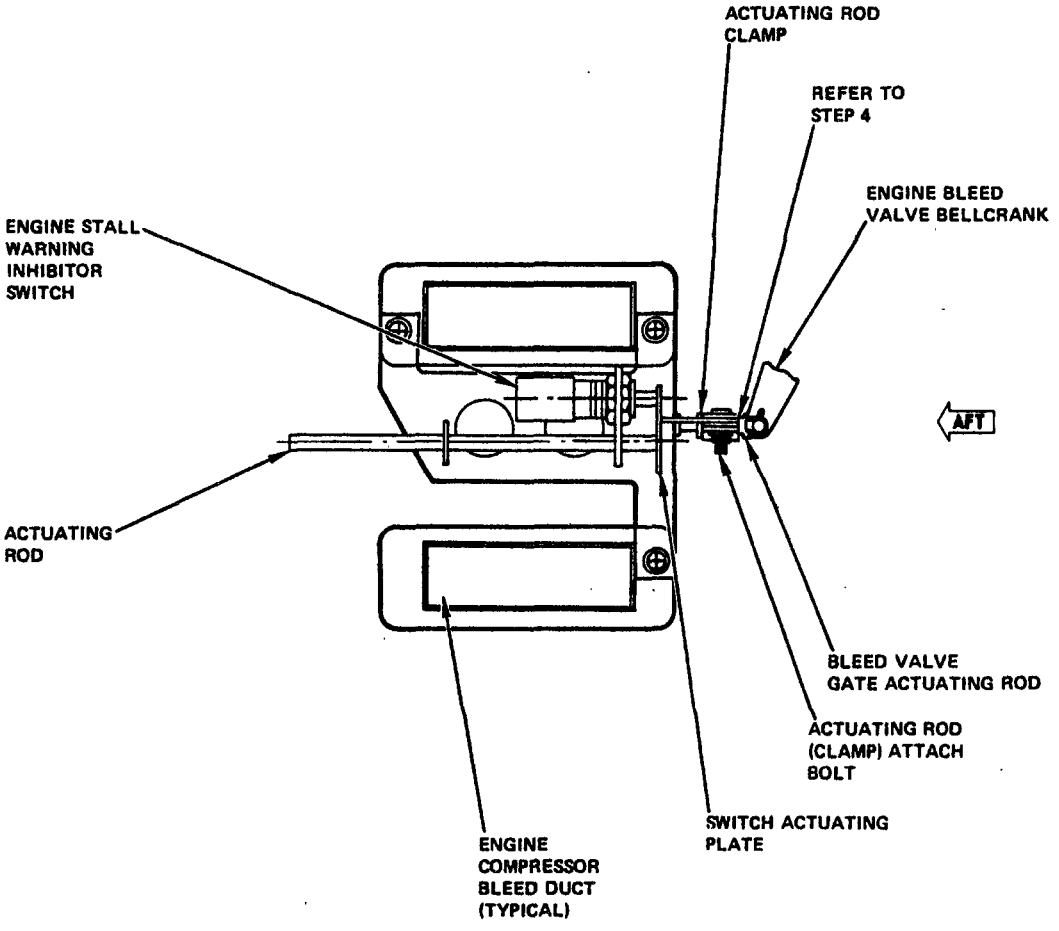

NOTE Rigging procedures are identical for RH and LH engines.

1. Open engine cowling.
2. Loosen switch lock nuts and move switch aft (as far as lock nut adjustment will allow).
3. Push engine bleed valve bellcrank aft until bellcrank is bottomed.
4. Insure that actuating rod clamp is bottomed against the shoulder of the bleed valve gate actuating rod.
5. Move engine bleed valve bellcrank forward 0.16 inch.
6. Adjust switch so that plunger just touches the switch actuating plate.
7. Tighten switch lock nuts.
8. Repeat steps 5, 6 to verify proper switch adjustment.
9. Check system for applicable safeties and security.
10. Close engine cowling.
11. Start engine(s). (Refer to Chapter 71.)
12. Advance applicable engine rpm above 85%.
13. Actuate stall warning test switch and check that stall warning light illuminates (flashing) and stall warning horn sounds.
14. Reduce engine speed to idle rpm.
15. Shutdown engine(s). (Refer to Chapter 71.)

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Figure 34-10-4 — Rigging Stall Warning System Engine Inhibitor Switches, S/N 370-1, -7 380-1 thru -38 Not Changed by SOK 9

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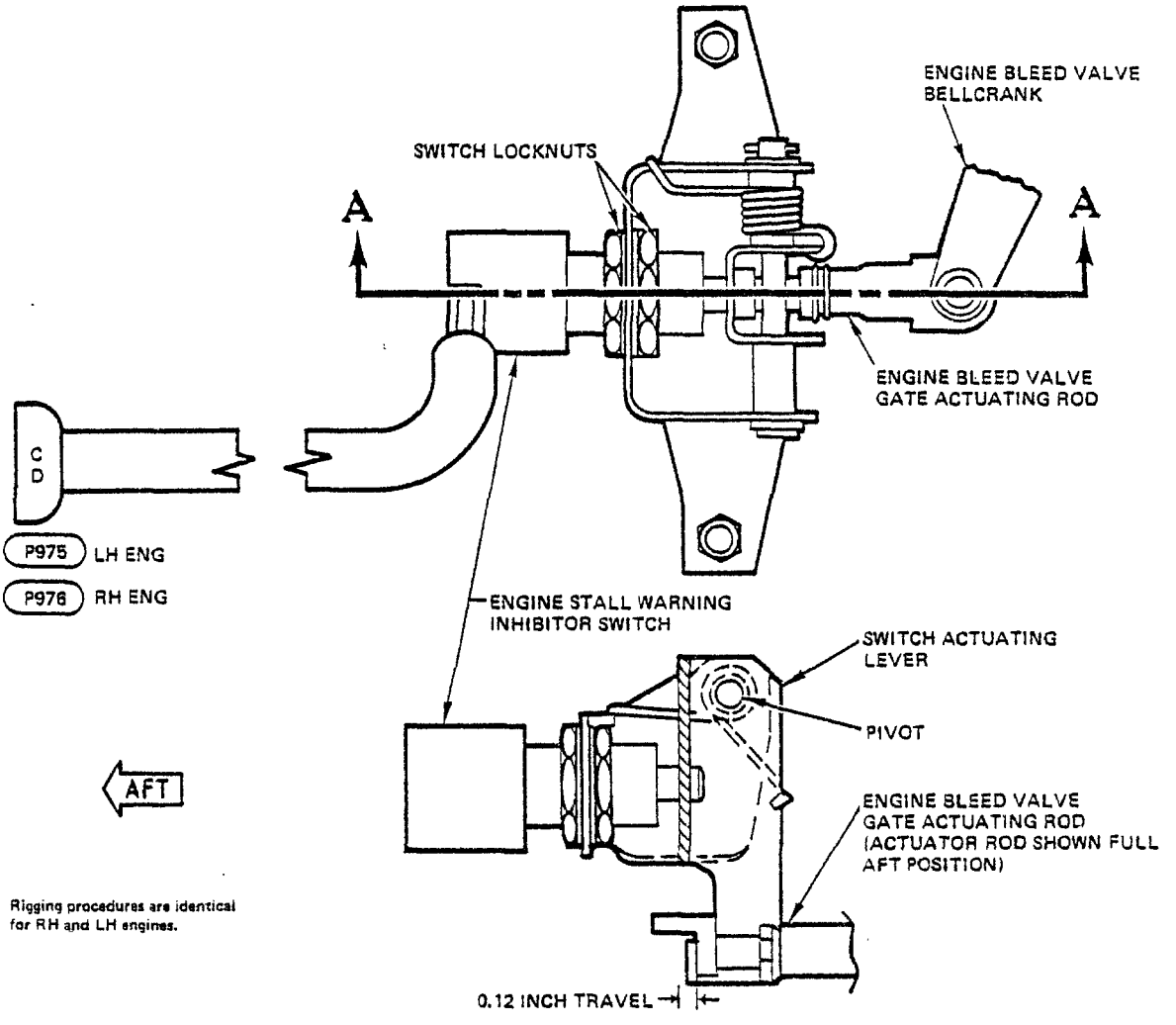

NOTE Rigging procedures are identical for RH and LH engines.

1. Open engine cowling.
2. Loosen switch locknuts and move switch aft (as far as locknut adjustment will allow).
3. Push engine bleed valve bellcrank aft until bleed valve gate actuating rod is bottomed.
4. Move engine bleed valve gate actuating rod forward .12 inch (switch actuating lever should also move .12 inch).
5. Disconnect electrical plug P975 (P976 for RH engine) and connect continuity tester leads to pins "C" and "D."
6. Adjust switch forward (using switch locknuts) until continuity is broken (open circuit).
7. Tighten switch locknuts.
8. Repeat steps 3 and 4 to verify proper switch adjustment.
9. Lockwire switch locknuts as applicable. Connect electrical plug. Check complete installation for security.
10. Close engine cowling.
11. Start engines. (Refer to Chapter 71.)
12. Advance applicable engine rpm above 85%.
13. Actuate stall warning test switch and check that stall warning light illuminates (flashing) and stall warning horn sounds.
14. Reduce engine speed to idle rpm.
15. Shutdown engine(s). (Refer to Chapter 71.)

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Rigging Stall Warning System Engine Inhibitor Switches - NA-370-1, -7, and NA-380-1 Thru -38 Changed by Optional Kit No. 9 and NA-380-39 and Later Airplanes
Figure 34-10-5

FLIGHT ENVIRONMENT DATA - TROUBLE SHOOTING

1. Trouble Shooting Pitot and Static Systems

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. INCORRECT OR ERRATIC READINGS.		
(1) Leaky tubing.	Make line leak test.	Inspect tubing and tighten fittings.
(2) Defective indicator.	Make case leak test.	Replace indicator.
(3) Ice or water in lines.		Drain and purge system.
B. NO READING OBTAINABLE.		
(1) Obstruction in pitot-static head or system tubing.	Check pitot-static head or system for obstructions.	Check pitot-static head and/or purge system.
(2) Ice or water in lines.		Drain and purge system.
(3) Obstruction in static ports.	Check static ports on pitot-static head for obstructions.	Clean and purge system.
C. POINTERS VIBRATE.		
(1) Excess vibration of instrument panel.	Check instrument frame shock mounts for looseness or permanent set.	Replace shock mounts if necessary.

FLIGHT ENVIRONMENT DATA

MAINTENANCE PRACTICES

PITOT AND STATIC PRESSURE SYSTEMS

All pressure lines, electrical connectors, and instrument receptacles must be covered when disconnected, to prevent entry of foreign materials, especially metal particles, that might make the instruments inoperative. Pitot and static line leakage tests should be made after any part of the pitot and static systems has been removed or replaced. To ensure a gravity drain of moisture to the drain sumps, reconnect all pitot and static lines, making certain there are no dips or sharp bends.

Draining Static Systems

At specified inspection intervals and after every airplane wash, or when the aircraft is exposed to, or parked outside during inclement weather, check pitot static system for water. Water must be drained prior to flight due to the extreme cold temperatures found at high altitudes, which result in the formation of ice. Ice in the pitot static system lines may cause erratic and unreliable instrument indications or ruptured pitot static system lines. Drain systems by opening drain sumps. Five drian sumps are located as follows: two static drain sumps on each side of the fuselage near the pitot-static tube, and a static drain sump in the stall warning line on the right side of the cabin.

Purging Pitot and Static Lines

1. Loosen fasteners that hold instrument panel in place.
2. Disconnect all instrument lines from their respective terminating points, such as autopilot, servo units, flight instruments, etc.
3. Turn disconnected lines away from instrument fittings and pitot-static operated electrical equipment.

CAUTION

Never apply pressure to static lines when instruments are connected. This may damage instruments beyond repair.

4. Remove drain sump plug, attach a source of pressure, and blow lines clear with dry air.
5. Connect all lines, and check system for tightness of all connections.

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1. Removing Pitot-Static Head

A. Pitot-static heads are on each side of the airplane, directly below the windshield. The fittings and connectors are reached from the cockpit, and the mounting screws are reached from outside. To remove pitot-static head, proceed as follows:

(1) Place all switches at OFF or airplane - parked position and disconnect external electrical power from airplane.

(2) To gain access to pitot-static head fittings and electrical connector, loosen fasteners that hold outboard trim panel in place, and move panel out.

(3) Disconnect pitot and static lines and electrical connector from pitot-static head, and cap all open lines, fittings, and electrical connectors.

(4) From outside the airplane, remove screws securing pitot-static head and remove head from fuselage.

2. Installing Pitot-Static Head

A. To install pitot-static head, proceed as follows:

NOTE: Be sure the word TOP on pitot-static head is up before installing.

(1) From outside the airplane, install screws to secure pitot-static head to fuselage.

(2) Remove protective caps from all lines, fittings, and electrical connectors.

(3) Connect pitot and static lines and electrical connector to pitot-static head.

NOTE: The system must be tested for leaks whenever the pitot-static head has been disconnected from the system. Refer to pitot-static pressure system test procedures.

(4) Reinstall access trim panel.

3. Removing Flap Travel Monitor Switch

A. The flap travel monitor switch is in the flap well outboard of the right main landing gear. To remove the switch, proceed as follows:

(1) Remove RH wheel well cover (dishpan).

(2) Disconnect switch arm and rod assembly.

(3) Cut lockwire and remove switch locknut.

(4) Disconnect electrical plug P1080 (located outbd side of wheel well).

(5) Remove switch assembly from switch support.

4. Installing Flap Travel Monitor Switch

A. Install switch as follows:

(1) Install switch so that the lead wires are down and to the left (looking inboard). Ensure that the two index pins on switch are mated with the two holes in switch support bracket.

(2) Locate the two index pins in holes provided in switch support.

(3) Install switch locknut, and safety with lockwire.

(4) Install switch arm and rod assembly. Do not tighten clamp bolt.

(5) Refer to rigging procedures.

5. Rigging Stall Warning System Engine Inhibitor Switches NA-370-1, -7 NA-380-1 Thru -38 Not Changed by Optional Kit No. 9

See figure 34-10-4.

6. Rigging Stall Warning System Engine Inhibitor Switches NA-370-1, -7 and NA-380-1 Thru -38 Changed by Optional Kit No. 9 and NA-380-39 and Later Airplanes

See figure 34-10-5

ADJUSTMENT/TEST

1. PITOT AND STATIC SYSTEM LEAKAGE TEST - S/N 370-2 THRU -6, -8 AND -9

NOTE

Test equipment used to perform the leakage tests on the pitot and static systems may be one of the following: the VPT-8 or VPT-10 air data test set with adapter kit, or the MB-1A pitot-static system field test set.

A. To check pitot static system for leakage, proceed as follows:
   (1) Apply external power to airplane.

NOTE

Electrical power will be required if an encoder altimeter is used on the pilot's instrument panel.

   (2) Connect test set (MB-1A or VPT-8 or VPT-10 tester) to a suitable adapter and connect adapter to LH pitot head.

CAUTION

To avoid damage to instruments, do not apply suction to pitot heads with ambient pressure on the static system.

   (3) Apply pressure slowly until pilot's airspeed indicator reads about three-fourths (300 knots) of full scale.
   (4) Shut off air flow and allow system to remain under pressure.
   (5) Lightly tap pilot's airspeed indicator and check that indicated airspeed does not decrease more than 5 knots after one minute.
   (6) Release system pressure.
   (7) Repeat steps (2) thru (6) using RH pitot head and checking co-pilot's airspeed indicator.
   (8) Position both static selector valves to normal.
   (9) Remove plug from static fitting on left hand side of fuselage (FS 89 aft plug).
  (10) Connect tester to static fitting.

CAUTION

To avoid damage to instruments, do not apply pressure to static lines. ***During static system checks, reduce the pitot system pressure sufficiently to keep the static and pitot pressure relationship within the design operating range of components. For example, a 400-knot airspeed indicator will read full scale with sea level ambient pressure on the pitot system and a simulated altitude of approximately 8700 feet on the static system: therefore, if the simulated altitude is increased (pressure decreased) the pitot pressure must also be reduced to avoid equipment damage.***

  (11) Apply vacuum to system to cause the altimeter to increase, at a rate not to exceed 5,000 fpm, to indicate 25,000 feet.
  (12) Shut off vacuum source and check that altimeter reading does not decrease by more than 500 feet after one minute.
  (13) Slowly relieve vacuum from system.
  (14) Disconnect tester from static fitting and install plug.
  (15) Connect tester to forward static fitting (FS 81).
  (16) Tape static ports on RH pitot tube.
  (17) Position pilot's static selector to ALERNATE.
  (18) Repeat steps (11) and (12). This step checks the plumbing to the opposite pitot head.
  (19) Slowly relieve vacuum from system.
  (20) Disconnect tester and install plug in static fitting.
  (21) Remove tape from static ports on RH pitot head.
  (22) Position Pilot's static selector to NORMAL.

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ADJUSTMENT/TEST (Continued)

1. PITOT AND STATIC SYSTEM LEAKAGE TEST - S/N 370-2, -6, -8 AND -9 (Cont'd)

(23) Remove plug from static fitting on right hand side of fuselage (FS 89 aft plug).
(24) Connect tester to static fitting.
(25) Apply vacuum to system to cause the altimeter to increase, at a rate not to exceed 5,000 fpm, to indicate 25,000 feet.
(26) Shut off vacuum source and check that altimeter reading does not decrease by more than 500 feet after one minute.
(27) Slowly relieve vacuum from system.
(28) Disconnect tester from static fitting and install plug.
(29) Connect tester to forward static fitting.
(30) Tape (FS 81) static ports on LH pitot tube.
(31) Position co-pilot's static selector to ALTERNATE.
(32) Repeat steps (25) and (26). This step checks the plumbing to the opposite pitot head.
(33) Slowly relieve vacuum from system.
(34) Disconnect tester and install plug in static fitting.
(35) Remove tape from static ports on LH pitot head.
(36) Position co-pilot's static selector to NORMAL.
(37) Ensure that all static fitting plugs are secured and all tape is removed from pitot heads.

2. PITOT AND STATIC SYSTEM LEAKAGE TEST - S/N 370-1, -7 AND 380-1 AND SUBS

A. To perform a pitot and static system leakage test, proceed as follows:
(1) Connect test set to a suitable adapter and connect adapter to LH pitot head.

CAUTION

To avoid damage to instruments, do not apply suction to pitot heads with ambient pressure on the static system.

(2) Apply pressure slowly until airspeed indicator reads about three-fourths (300 knots) of full scale.
(3) Shut off air flow and allow system to remain under pressure.
(4) Lightly tap pilot's airspeed indicator and check that indicated airspeed does not decrease more than 5 knots after one minute.
(5) Release system pressure.
(6) Repeat steps (1) thru (5) using RH pitot head and checking co-pilot's airspeed indicator.
(7) Position pilot's static selector valve to left (L).
(8) Remove plug from static fitting on left hand side of fuselage (FS 89 aft plug).
(9) Connect tester to static fitting.
(10) Tape static holes on left pitot head only.
(11) On airplanes with an encoding altimeter, apply external electrical power and energize inverter system.

CAUTION

To avoid damage to instruments, do not apply pressure to static lines. ***During static system checks, reduce the pitot system pressure sufficiently to keep static and pitot pressure relationship within the design operating range of components, for example, a 400-knot airspeed indicator will read full scale with sea level ambient pressure on the pitot system and a simulated altitude of approximately 8700 feet on the static system: therefore, if the simulated altitude is increased (pressure decreased) the pitot pressure must also be reduced to avoid equipment damage.***

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(12) Apply pitot pressure to read 100 knots. Maintain this 100 knot indication by decreasing pressure as required on the pitot system as the altitude is increased.
(13) Apply vacuum to system to cause the altimeter to increase at a rate not to exceed 5,000 feet per minute, to indicate 25,000 feet.
(14) Shut off vacuum source and check that pilot's altimeter reading does not decrease by more than 500 feet after one minute. This step also checks the pilot's selector valve for internal leakage and the cross ship plumbing.
(15) Slowly relieve vacuum from system.
(16) Disconnect tester from static fitting and install plug.
(17) Remove tape from left hand pitot head static port.
(18) Position pilot's selector to right (R). Tape static ports on right pitot head only.
(19) Remove plug from LH forward static fitting (FS81).
(20) Connect tester to static fitting.
(21) Repeat steps 12, 13 and 14.
(22) Slowly relieve vacuum from system. Remove tester from static fitting and install plug. Position pilot's selector to both. Remove tape from RH pitot tube. Remove electrical power.
(23) Position copilot's static selector valve to left (L).
(24) Remove plug from static fitting on right hand side of fuselage (FS81 Plug).
(25) Connect tester to static fitting.
(26) Tape static holes on left pitot head.
(27) Apply pitot pressure to read 100 knots. Maintain this 100 knot indication by decreasing pressure as required on the pitot system as the altitude is decreased.
(28) Apply vacuum to system to cause the altimeter to increase, at a rate not to exceed 5,000 feet per minute, to indicate 25,000 feet.
(29) Shut off vacuum source and check that copilot's altimeter reading does not decrease by more than 500 feet after one minute. This step also checks the copilot's selector valve for internal leakage and the cross ship plumbing.
(30) Slowly relieve vacuum from system.
(31) Disconnect tester from static fitting and install plug.
(32) Position copilot's selector to right (R). Tape static ports on right pitot head. Remove tape from LH static ports.
(33) Remove plug from rear static fitting (FS89).
(34) Connect tester to static fitting.
(35) Repeat steps 27, 28 and 29.
(36) Slowly relieve vacuum from system. Remove tester from static fitting and install plug. Position copilot's selector to both.
(37) Remove tape from right hand pitot head static port.
(38) Ensure that all static fitting plugs are secured and all tape is removed from the pitot heads.

3. Testing Mach and Airspeed Warning System

A. Test Mach and airspeed warning systems as follows:

NOTE: Because the tolerance for the sensor switch is less than that for the flight instruments, the sensor switch may have to be removed from the airplane and checked separately if the switch does not check within the limits required in the testing procedures. Do not reject switch unless it is found to be defective after a separate test. (Refer to paragraph 3.B.)

(1) Connect pitot-static tester to airplane.

(2) Connect a headset to pilot's or copilot's headset jack.

(3) Apply 28-volt dc power to airplane.

(4) Push in "MACH WARN" circuit breaker on left overhead forward circuit-breaker panel.

(5) Depress "MACH-A/S WARNING TEST" push button on pilot's main outboard instrument subpanel. An audible tone should be heard from the pilot's warning horn and in the pilot's and copilot's headsets.

(6) Apply pitot and static pressure to the system as required to produce the altitude, airspeed, and Mach number settings necessary for operation of the airspeed sensor switch. The switch should operate and the warning horn should blow at the following settings:

ALTIMETER (FEET)	INDICATED AIRSPEED	INDICATED MACH NUMBER
0	350.0 to 356.0	
10,000	355.4 to 361.4	
21,000	364.6 to 370.6	
22,000		0.800 to 0.810
30,000		0.800 to 0.810
40,000		0.800 to 0.810

B. Test Mach and airspeed sensor switch as follows:

(1) Remove Mach and airspeed sensor switch from airplane.

(2) Connect pitot-static tester to pressure and static fitting on sensor switch.

(3) Connect a 28-volt dc power source to pin "5" of the sensor switch receptacle and connect a ground wire to pin "6." Using a multimeter, connect one probe to pin "3" and the other probe to pin "4." Set multimeter to read continuity.

(4) Check that multimeter reads continuity between pins "3" and "4" at the altitude, airspeed, and Mach number settings noted in step (6) of paragraph 3.A.

4. Testing Stall Warning System - Aircraft NA-370-2 Through -6, -8 and -9

NOTE: Test equipment required to perform this test is as follows: an MB-1A pitot-static system field test set; a multimeter; and a lift transducer calibrator, Safe Flight Instrument Corp, Part No. 989-21, or force applicator, Part No. 990, with 0.6 to 6.0 gram gage, Part No. 34263-2.

A. Prepare for test as follows:

(1) Pull both transducer vane heater circuit breakers on left overhead circuit-breaker panel.

(2) Place airplane on jacks to extend main and nose gear struts. An alternate method is to open the nose gear load switch and close the right and left main gear load switches to simulate a flight condition. The airplane must be level within ±2 degrees along the pitch axis.

(3) Connect a 28-volt dc external power supply to airplane.

B. Test stall warning system as follows:

(1) Place "INVERTER" switch at NO. 1.

(2) Push in both transducer vane heater circuit breakers. Note that inboard and outboard lift transducer vanes become hot to the touch within 2 minutes.

CAUTION: TURN OFF HEATER POWER WITHIN 2 MINUTES, TO AVOID EXCESSIVE HEATING OF THE VANES IN STILL AIR.

(3) With flaps up and vanes held in aft position, position "STALL WARNING TEST" switch to OUTBD VANE and INBD VANE. Note that control stick shaker operates in both positions.

(4) Lower flaps to full down position.

(5) Push in "STALL WARNING - DC" circuit breaker and "STALL WARNING AC" circuit breaker. Pull out both inboard and outboard vane heater circuit breakers.

(6) Adjust test calibrator gage so that it touches inboard lift transducer vane within 1/32 of tip. To steady calibrator, rest screws of calibrator plate against surface of wing. Force applicator, Part No. 990, with 0.6- to 6.0-gram gage should be used if available.

(7) Apply a 2.5 (±0.5) gram force to inboard lift transducer vane tip in forward direction. The shaker should operate. A slight (1/4-gram) reduction force should cause the shaker to stop.

(8) Raise flaps to full up position.

(9) Apply a 1.5 (±0.5) gram force to inboard lift transducer vane tip in forward direction. The shaker should operate.

(10) Connect MB-1A pitot-static tester to static system. Simulate a pressure altitude above 45,000 feet.

(11) If shaker is operating, apply force to outboard lift transducer in aft direction to stop shaker. Then slowly reduce force until shaker starts to operate. The shaker should start operating with a 1.5 (±1.0) gram force applied in the aft direction.

(12) Note that any force applied to inboard lift transducer has no effect on shaker operation.

(13) Simulate a pressure altitude of 15,000 feet or below.

(14) Note that any force applied to outboard lift transducer has no effect on shaker operation

(15) Push in "LH ENGINE - MASTER" and "RH ENGINE - MASTER" circuit breakers, and left and right "IGN & BOOST CONT" circuit breakers. Place "ENGINE MASTER" switch at ON and "IGNITION" selector switch at NORMAL; then advance left and right throttles. Note that left and right ignition exciters do not operate.

(16) Position "STALL WARNING TEST" switch at INBD VANE. Note that both stick shakers operate, left and right ignition exciters are operating, and left and right ignition lights are on.

C. Test main and nose gear load switches as follows:

(1) Remove jacks from airplane (or close nose gear load switch and open both main gear load switches).

(2) With system energized, except for vane heaters, note that control shakers do not operate.

(3) Operate "STALL WARNING TEST" switch in both direction; shakers should operate.

(4) Reset both vane heater circuit breakers. Note that both inboard and outboard vanes become warm to touch. (Transducer heaters operate at reduced voltage when airplane weight is on wheels.)

4A. Testing Stall Warning System - Aircraft NA-370-1, -7, NA-380-1 and Later

NOTE: Test equipment required to perform this test is as follows: an MB-1A pitot-static system field test set and a lift transducer calibrator. Safe Flight Instrument Corp., Part No. 989-21, or force applicator, Part No. 990, with 0.6 to 6.0 gram gage, Part No. 34263-2 and a 3 to 30 gram gage, Part No. 34263-4.

A. Prepare for test as follows:

(1) Check that airplane is level within $\pm 5^{\circ}$ along the pitch axis.

(2) Connect pitot-static test set to static system.

(3) Raise wing flaps.

(4) Open RH and LH stall warning dc power heater circuit breakers.

(5) Disconnect nose and main landing load switches. Position load switches to simulate airborne position.

(6) Open engine access doors.

(7) Connect external electrical power to airplane.

B. Test stall warning system as follows:

(1) Open RH Stall Warning - AC circuit breakers (LH forward overhead circuit breaker panel).

(2) Close (push in) LH Stall Warning - AC circuit breakers.

(3) Hold left hand transducer vanes in aft position. Position STALL WARNING TEST switch at INBD VANE and OUTBD VANE (left hand wing). Shaker should operate in both positions.

(4) Manually close stall warning system engine inhibitor switches.

(5) Adjust test calibrator gage so that it touches the left hand inboard lift transducer vane within 1/32 inch (.03") of tip. To steady calibrator, rest screws of calibrator plate against surface of wing. Force applicator, Part No. 990, with 0.6- to 6.0 or 3 to 30-gram gage should be used if available.

(6) Apply a 2-gram force to left hand inboard lift transducer vane tip in the aft direction. Decrease force slowly and check that shaker operates between 0.0 $\pm$ 0.5 grams. Move vane slightly aft and release. The shaker should stop and remain off. Move vane forward and release. The shaker should operate and remain on.

(7) Apply a 2.5 (±0.5) gram force to left hand inboard lift transducer vane tip in forward direction. The shaker and engine stall warning light should energize.

(8) Lower flaps to PART position. Shaker operation should cease at 60% ±10% of flap position.

(9) Lower flaps to full down position.

(10) Reduce gram force slightly, then reapply force (forward) on vane until shaker operates. Shaker should operate at 2.4 ±0.5 grams.

(11) Apply 4.9 ±0.5 grams force in the forward direction. Check that shaker and engine stall warning light are energized.

(12) Raise flap to full up position. Simulate a pressure altitude above 45,000 feet.

(13) Apply a 17 gram force aft on the LH outboard lift transducer with gage positioned the same as in step (5). Slowly reduce gram force and check that shaker operates at 14.5 ±1.0 grams.

(14) Continue to reduce force and check that engine stall warning light and shakers are energized at 8.5 ±1.0 grams. Remove vacuum source from static system.

(15) Manually push inboard and outboard vanes full aft. Actuate stall test switch in inboard and outboard (LH) and check that engine stall warning and shakers are energized.

(16) Open LH STALL WARNING AC circuit breaker and close RH STALL WARNING AC circuit breaker.

(17) Repeat steps (5) through (15) for right hand transducer system.

CAUTION: TO PREVENT A POSSIBLE ENGINE FIRE, ENSURE THAT ENGINE TAIL PIPE IS FREE OF FUEL BEFORE PERFORMING THE IGNITION TEST.

(18) Close "LH ENGINE - IGN & BOOST and RH ENGINE - IGN & BOOST" circuit breakers. Open the "BOOST PUMP - LH FUEL," "BOOST PUMP - RH FUEL," "LH ENGINE - MASTER" and "RH ENGINE - MASTER" circuit breakers.

(19) Advance right hand power lever and check that ignition exciters do not operate.

(20) Position "STALL WARNING TEST" switches at INBD VANE (right wing). Check that shakers operate and both engine ignition exciters are operating. Ignition ON indicator lights in starter buttons should be illuminated.

(21) Close RH STALL WARNING - AC circuit breakers and open RH STALL WARNING - AC circuit breakers.

(22) Repeat steps (19) and (20) for left hand system.

(23) Place engine power levers at cut-off, remove electrical power from airplane, engage all circuit breakers and position all switches in their normal or OFF position.

(24) Remove pitot-static test set and close engine access doors.

4B. Stall Warning Transducer Vane Heaters - Operational Check

A. To perform an operational check of the transducer vane heaters, proceed as follows:

(1) Close the following circuit breakers:

(a) On airplanes NA-370-1, -7, NA-380-1 and Later, LH STALL WARNING-INBD and OUTBD VANE HEATERS

(b) RH STALL WARNING-INBD and OUTBD VANE HEATERS

(c) LANDING GEAR-POS

(2) Remove pitot and EPR tube covers (if installed).

(3) Connect external electrical power to airplane.

(4) Position ANTI-ICE-PITOT & STALL WARNING switch to ON. Check transducer vanes for presence of "low" heat.

CAUTION: TURN OFF HEATER POWER WITHIN 1 MINUTE, TO AVOID EXCESSIVE HEATING OF THE VANES IN STILL AIR.

(5) Position ANTI-ICE-PITOT & STALL WARNING switch to OFF.

(6) Open LANDING GEAR-POS circuit breaker. Check transducer vanes for presence of "high" heat.

(7) Close LANDING GEAR-POS, RH and LH PITOT HEAT circuit breakers, remove external electrical power from airplane.

MAINTENANCE PRACTICES (Continued)

5. RIGGING FLAP TRAVEL MONITOR SWITCH — S/N 370-2 thru -6

A. Connect a continuity light or ohmmeter across terminals M7 and M9 of TB593.

B. Operate flaps to full up position. The circuit must be closed.

C. Operate flaps to 13.0° ±30′ degrees travel. The circuit must be open.

D. Adjust position of arm on switch, if necessary, to obtain circuit operation within required limits.

E. Operate flaps full down. The circuit must remain open from the 20 (±1) degree position to full down position.

F. Operate flaps to full up position. The circuit must close at the 20 (±1) degree position and remain closed for remainder of flap travel to full up position.

6. RIGGING FLAP TRAVEL MONITOR SWITCH — (S/N 370-1, -7 thru -9, 380-1 and Subs)

CAUTION

To prevent damage to switch, ensure that switch arm clamping bolt at switch shaft is loose.

A. Remove RH wheel well cover (dishpan) to gain access to P1080.

B. Mount inclinometer on top of flap.

C. Apply electrical power to airplane and position flaps at 11°±30′ from full up (0°+15′, -00′) position. Turn off electrical power to airplane.

D. Turn switch shaft counterclockwise until internal stop is reached.

E. Connect ohmeter between pin D of electrical plug P1080 (located outbd side of RH wheel well) and ground.

F. Slowly turn switch shaft clockwise until continuity is indicated. Hold switch shaft in this position and tighten shaft and clamping bolt. Reconnect P1080.

G. Apply electrical power to airplane. Operate flaps to full up position. Disconnect P1080 and check for continuity between electrical plug (P1080) pins with the flap in the following positions:

NOTES

- ***The flap travel monitor switch has two separate sets of contacts which are actuated by a double ramped cam. The stall warning switch (terminals 1, 2 and 3) actuates first at 11°±30′. The landing gear warning (terminals 4, 5 and 6) actuates with the flaps below the part position.***
- ***Electrical plug P1080 must be connected prior to flap operation.***

FLAP POSITION	CONTINUITY BETWEEN PINS OF P1080
(1) "UP"	"C" & ground and "H" & "E"
(2) Move to "PART"	"D" & ground and "H" & "E"
(3) Move to "DOWN"	"D" & ground and "H" & "G"
(4) Move to "PART"	"D" & ground and "H" & "E"
(5) Move to "UP"	"C" & ground and "H" & "E"

H. Remove electrical power from airplane.

I. Connect and safety electrical plug P1080.

J. Check complete switch installation for proper safeties and security.

K. Install RH wheel well cover (dishpan).

L. Perform an operational check of flap system.

7. TESTING FREE AIR TEMPERATURE INDICATING SYSTEM

Bench Check the Free Air Temperature Indicator

A. Remove the indicator from the airplane.

B. The indicator shall be tested by applying power of 28.5 VDC and the resistance values, normally produced by the resistance bulb for the test points, applied to the resistance bulb terminals of the indicator by menas of a precision resistance box.

CAUTION

Refer to wiring diagram manual to ensure that power is applied to proper terminal on the indicator.

C. The tolerance shall be as listed when the temperature indicators are tested at the test points specified. The indicator shall be tapped lightly before each reading and the difference in reading after tapping shall be considered as the friction error.

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MAINTENANCE PRACTICES (Continued)

Free Air Temperature Indicator Table

Test Points	Test Resistance Ohms	Indicator 162BL21 MS28008-1 AF21295, G-7		Tolerance (±°C) 260119 260120		Bulb Tolerance (± ohms)
		Scale	Friction	Scale	Friction	
-70	68.27			3	3	0.5
-40	77.39			3	3	0.5
-30	80.56	5	3			0.4
0	90.38	5	3	3	3	0.4
+30	100.91	5	3			0.4
+40	104.60			3	3	0.4
+80	120.36			3	3	0.5

Checking the Free Air Temperature Bulb (while installed)

The bulb resistance shall be 97 to 101 ohms at room temperatures from 20°C to 30°C (68°F to 86°F). Specific temperatures vs resistance checks may be made in accordance with the free air temperature indicator table.

8. FUNCTIONAL TEST ALTIMETERS
   A. Remove altimeters from instrument panel.
   B. Record altimeter part and serial numbers.
   C. Calibrate and certify altimeters in accordance with Federal Aviation Regulation (FAR) Part 43. Appendix E.
   D. Install altimeters in instrument panel.
   E. Leak check system.

9. EMERGENCY GYRO POWER SUPPLY PS-800 or PS-823 — TEST

   Inspect and performance test emergency gyro power supply as per Jet Electronics and Technology, Inc., Service Letter 8()(PS823 power supply) or Service Letter No. 4 (PS-800 power supply).

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ATTITUDE AND DIRECTION INSTRUMENTATION - DESCRIPTION

1. Stand-by Compass

A conventional magnetic compass, suspended from the center windshield divider, is furnished for navigation in case of instrument or electrical system failure. The stand-by compass light is controlled by the indirect portion of the pedestal and overhead light rheostat on the left console circuit-breaker panel. (Refer to Chapter 33.) Compass cards are on the center windshield divider below the compass.

2. Remote Magnetic Flux Detectors and Compensators - Airplanes NA-380-1 Through -40

A remote magnetic flux detector is installed in each wing tip for the No. 1 and No. 2 magnetic compass systems. The flux detectors are factory-installed units to be used in the customer-installed magnetic compass systems. The flux detectors are the direction-sensing components for the magnetic compass systems or for other navigation systems requiring a magnetic heading reference. The detecting elements of the flux detectors sense the horizontal component of the earth's magnetic field, establishing a magnetic heading reference for the airplane. A compensator is mounted on top of each detector to allow for error adjustment.

ATTITUDE AND DIRECTION INSTRUMENTATON

MAINTENANCE PRACTICES

STAND-BY COMPASS SWINGING AND ADJUSTING

The stand-by compass is swung by positioning the airplane on a compass rose to face actual headings north, east, south, and west. At each of these headings, the stand-by compass reading is recorded. If a compass rose is not available, the sighting compass method may be used. If the sighting compass method is used, the area should be reasonably level. The compass should be within 5 degrees of horizontal during swinging or adjusting.

General Preswinging Precautions

1. Position airplane at least 100 yards from automobiles, tractors, steel buildings, or other airplanes.
2. Use nonmagnetic tools when adjusting compass.
3. Secure, in position occupied during normal flight, all airplane equipment which has any magnetic effect on compass.

Swinging Stand-by Compass

1. Position airplane on compass rose.

NOTE

If compass rose is not available, use sighting compass method to swing compass.

2. Loosen screws on adjustment screw cover and swing cover to expose adjustment screws.
3. Align the NS and EW adjusting screws until dots on screws mate with dots on case. This zeroes the compensating magnets.
4. During compass swinging, operate all systems which would normally be on in flight, with the following exceptions:
   - Windshield anti-ice
   - Electronic bay ground blower
   - Indicator, console, pedestal overhead, engine, and radio indirect lights
5. With airplane facing north on compass rose, turn NS adjustment screw so that all error is removed and compass reads north (0).
6. Swing airplane to face east. Turn EW adjustment screw so that all error is removed and compass reads east (090).
7. Swing airplane to face south. If compass does not read south (180), turn adjustment screw so that only half the error is removed. For example, if stand-by compass reads 178 on the south heading, turn NS adjustment screw so that compass reads 179.
8. Swing airplane to face west. If compass does not read west (270), turn EW adjsutment screw so that only half the error is removed. For example, if stand-by compass reads 274 on the west heading, turn EW adjustment screw so that compass reads 272.
9. To perform a residual compass swing, reposition airplane to north heading on compass rose. Swing airplane through cardinal headings (north, east, south, and west) and note compass readings. The maximum spread between maximum plus deviation and maximum minus deviation should not be more than ±10 degrees of the aircraft heading.

NOTE

The compass swing should be made with the DC generators turned on and off at each 15-degree heading, and the readings should be entered on two compass correction cards, one marked DC GEN ON and the other marked DC GEN OFF.

10. Shut down all equipment turned on in step 4.
11. Fill in steer column on compass card by making a swing through all 24 positions listed.
12. Place steer card in stand-by compass card holder.

REMOTE MAGNETIC COMPASS SWINGING AND ADJUSTMENT - S/N 380-1 THRU -40

NOTE

The swinging and adjusting procedure is applicable to both the pilot's (No. 1) and the co-pilot's (No. 2) systems.

Precompensation Steps

1. Position airplane on compass rose.

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MAINTENANCE PRACTICES (Continued)

Precompensation Steps (Cont'd)

NOTE

The clearances between the compass compensator and wing tip structure are very close. Extreme care should be taken when removing or installing the wing tips. Flexing the wing tips up and down during installation or removal may result in damage to the compensator.

2. Remove left and/or right wing tip (left tip for No. 1 system, right tip for No. 2 system).
3. Align white dots on both NS and EW compensating screws on each flux detector.
4. Loosen detectors on mounting screws, and rotate detector until zero-degree line is aligned with reference slot on mounting bracket; then tighten screws.

Swinging and Compensation

1. Connect external electrical power to airplane.
2. Push in system circuit breakers.
3. Turn on AC and DC power to system.
4. Allow 30 seconds for warm-up; then place NO. 1 COMP and/or NO. 2 COMP switch at AUTO. The horizontal situation indicator should rapidly turn to the approximate airplane heading, and the compass slave indicator should be centered within ±1 pointer width.

NOTE

Hereafter, when the airplane is rotated to a new heading, the compass system must be allowed to slave to the new heading. The system is slaved when slave indicator is centered within ±1 pointer width.

5. Rotate airplane to a north heading as indicated by marker on left landing gear, aligning with 270-degree radial line painted on the rose. The marker must always line up with the radial line 90 degrees counterclockwise from the desired heading.
6. Record heading of horizontal situation indicator.
7. Rotate airplane to an east heading. Record the heading of the indicator.
8. Repeat step 7 for a south and west heading.
9. Compute the deviation as follows:

NOTE

If the indicator reading is greater than the heading it is being subtracted from, the deviation will be negative.

A. For north deviation, take the difference between north indicator reading and either zero or 360 degrees, depending on which is applicable.

B. For east deviation, take the difference between east indicator reading and 90 degrees.

C. For south deviation, take the difference between south indication reading and 180 degrees.

D. For west deviation, take the difference between west indicator reading and 270 degrees.

10. Compute quantity coefficients A, B, and C by algebraically adding and subtracting deviations in accordance with the following equations:

$$\text{Coefficient B} = \frac{\text{East Deviation - West Deviation}}{2}$$

$$\text{Coefficient C} = \frac{\text{North Deviation - South Deviation}}{2}$$

$$\text{Coefficient A} = \frac{\text{North D. + East D. + South D. + West D.}}{4}$$

Rules for algebraic addition and subtraction are as follows:

ADDITION - To add two numbers with the same sign, add their absolute [1] values and prefix the common sign.

(+6) + (+3) = 9
(-6) + (-3) = 9
(-6) + (-3) + (-2) + (-4) = -15

To add two numbers with opposite signs, take the difference between their absolute values and prefix the sign of the number with the largest absolute value.

(+6) + (-3) = +3
(-6) + (+3) = -3
(-6) + (+3) + (+2) + (-4) = -5

SUBTRACTION - The process of subtraction is opposite to the process of addition. To subtract a quantity from another, change the sign of the quantity to be subtracted; then add the quantities.

(-6) - (-3) = (-6) + (+3) = -3

(-3) - (-6) = (-3) + (+6) = +3

(+6) - (-3) = (+6) + (+3) = +9

(+6) - (+3) = (+6) + (-3) = +3

[1] Magnitude only without regard to the sign.

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MAINTENANCE PRACTICES (Continued)

Swinging and Compensation (Cont'd)

11. Turn airplane to a north heading; then adjust NS compensating screw of flux detector with a brass screwdriver until indicator is corrected by the amount of coefficient C. Observe sign as well as magnitude of coefficient.
12. Turn airplane to an east heading. Adjust EW compensating screw of flux detector, with a screwdriver, until indicator reading is corrected by the amount of coefficient B. Observe sign as well as mangitude of coefficient.
13. Loosen flux detector mounting screws and rotate detector until indicator is corrected by the amount of coefficient A. Observe sign as well as magnitude of coefficient. Retighten flux detector mounting screws.

NOTE

The following are sample calculations showing the method of determining coefficients C, B, and A as outlined in steps 9 and 10, and the resultant flux detector adjustment.

MANGETIC HEADING	INDICATOR READING	DEVIATION
000	006-1/2	-6-1/2
050	090	0
180	175-1/2	+4-1/2
270	276	-6

C = 1/2 [-6-1/2 - (+4-1/2)] = -5-1/2 deg

B =1/2 [0 - (-6)] = +3 deg

A = 1/4 [-6-1/2 + 0 + (+4-1/2) + (-6)] = -2 deg

A. With airplane on a north heading, adjust NS compensating screw, as required, using coefficient C until indicator reads 001 degree.

B. With airplane on an east heading, adjust EW compensating screw, as required, using coefficient B until indicator reads 093 degrees.

C. With airplane on an east heading, rotate flux detector, as required, using coefficient A until indicator reads 091 degrees.

NOTE

The clearances between the compass compensator and wing tip structure are very close. Extreme care should be taken when removing or installing the wing tips. Flexing the wing tips up and down during installation or removal may result in damage to the compensator.

14. Replace left wing tip.
15. Fill out compass correction card.
16. Turn off compass system and AC and DC power.
17. Disconnect external DC power and install wing tips.

REPLACEMENT OF REMOTE MAGNETIC FLUX DETECTORS - S/N 380-1 THRU -40

NOTE

A magnetic flux detector and detector compensator unit is mounted as a bracket in each wing tip.

Removal:

1. Place all switches in OFF or airplane-parked position and disconnect external electrical power.

NOTE

The clearances between the compass compensator and wing tip structure are very close. Extreme care should be taken when removing or installing the wing tips. Flexing the wing tips up and down during installation or removal may result in damage to the compensator.

2. Remove applicable wing tip (left wing tip for pilot's system and right wing tip for co-pilot's system).
3. Disconnect flux detector electrical leads.
4. Remove the three mounting screws securing detector to bracket and remove detector unit.

Installation:

1. Place all switches in OFF or airplane-parked position.
2. Disconnect external electrical power.

NOTE

The clearances between the compass compensator and wing tip structure are very close. Extreme care should be taken when removing or installing the wing tips. Flexing the wing tips up and down during installation or removal may result in damage to the compensator.

3. Remove wing tip, if installed.
4. Place detector unit on bracket and partially install three mounting screws.
5. Align white dots on both the NS and EW compensating screws on detector.
6. Rotate flux detector to align zero-degree line on detector with reference slot on mounting bracket; then tighten screws.

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MAINTENANCE PRACTICES (Continued)

Installation: (Cont'd)

7. Connect flux detector electrical leads.
8. Perform compass swing.

NOTE

The clearances between the compass compensator and wing tip structure are very close. Extreme care should be taken when removing or installing the wing tips. Flexing the wing tips up and down during installation or removal may result in damage to the compensator.

9. Install wing tip.

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CHAPTER 35

OXYGEN

CHAPTER 35

OXYGEN

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CHAPTER 35

OXYGEN

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*The asterisk indicates pages revised, added, or deleted by the current revision.

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OXYGEN

DESCRIPTION AND OPERATION

GENERAL

The breathing oxygen system is a high-pressure (1800 psi) gaseous type. A single high-pressure oxygen storage cylinder of 74-cubic-foot capacity is stored in the aft lower left side of the passenger compartment. A frangible, disk-type, high-pressure relief valve, in the manual shutoff and coupling valve, will rupture if pressure in the oxygen bottle should exceed 2700 to 2900 psi. A green, frangible disk indicator, on the left side of the fuselage, provides a visual indication that the relief valve has ruptured, venting the oxygen overboard. An electrical high-pressure sensing transmitter is mounted next to the oxygen storage cylinder and is plumbed into a high-pressure line between the oxygen storage cylinder and the oxygen pressure regulator. The transmitter transmits the total cylinder pressure to a gauge on the oxygen system panel in the cockpit.

Oxygen flows from the storage cylinder to the oxygen system pressure regulator. This regulator reduces the pressure to 65-95 psi and routes part of the oxygen flow to the pilot's and copilot's automatic diluter-demand pressure-breathing oxygen regulators. Oxygen flow to the passengers' oxygen masks is controlled by a solenoid valve and an aneroid valve, both within the oxygen regulator.

The solenoid valve is controlled by two switches; one a switch on the oxygen system panel, and the other an aneroid-type altitude switch that is vented to cabin atmosphere. The altitude switch closes and completes a circuit to the solenoid when pressure within the cabin reaches a 13,500 (±500) foot altitude. The switch on the oxygen system panel has three positions: AUTO, OPEN, and CLOSED. The switch is normally positioned at AUTO, allowing the altitude switch to open the solenoid valve when cabin pressure decreases sufficiently. Positioning the switch to OPEN bypasses the altitude switch and energizes the solenoid, opening the valve. Positioning the switch to CLOSED deactivates the circuit, keeping the solenoid valve closed.

When oxygen flow is past the solenoid valve, it is then controlled by an aneroid-type flow regulator that is vented to cabin atmosphere. This valve allows no flow at a sea-level altitude, but flow starts and increases as cabin altitude increases, the maximum flow to the passengers' portion of the oxygen system being about 36 psi.

A bypass valve is installed on the oxygen system panel. This mechanically operated valve is part of a plumbing loop that is installed to bypass the solenoid valve within the oxygen regulator. When this valve is positioned at NORMAL, the valve is closed and there is no effect on the oxygen flow; when the valve is positioned at EMER OPEN, the valve is open and 65 to 95 psi flow from the pilot's side of the system bypasses the solenoid valve and flows directly to the aneroid-type regulator.

Passengers' oxygen masks are located in five compartments located in the center line of cabin ceiling. A hinged door holds the masks within the compartment. The latch is released by oxygen pressure, and the latched doors under spring tension, opens, thus making the oxygen masks available. A small pin, attached to the mask, keeps a valve closed (no oxygen flow) until the mask is pulled down for use. A plastic bag, part of the mask assembly, immediately inflates, and the system is ready for use.

The electrical portion of the oxygen system is powered by a 28VDC primary circuit. When an aneroid-type altitude switch, part of the cabin pressurization system, closes at 10,000 feet cabin altitude, the FASTEN SEAT BELT light above the passengers' seats goes on and CABIN PRESS FAIL caution light (on caution and warning panel) illuminates. When the oxygen system aneroid-type altitude switch closes at 13,500 (±500) feet cabin altitude and the passengers' oxygen control switch is at AUTO, the WEAR OXYGEN MASK and NO SMOKING lights above the passengers' seats illuminate, the oxygen warning horn sounds, and the solenoid valve within the oxygen regulator opens. Positioning the passengers' oxygen control switch on the oxygen system panel to OPEN will also light the lights, sound the horn, and open the solenoid valve.

OXYGEN REGULATORS

ARO 14950-33 automatic diluter-demand pressure-breathing oxygen regulators are located on the pilot's left console and the co-pilot's right console to provide a controlled oxygen breathing supply for each of these positions.

The regulator automatically supplies a proper mixture of oxygen and air to the pilot or copilot (upon demand) at all cabin altitudes to 27,000 feet, at which altitude 100 percent oxygen is supplied (pressure-fed). It incorporates a diluter lever, and emergency toggle lever, a pressure gauge, a flow indicator, and an oxygen supply lever. The diluter lever is used to select either normal oxygen under all normal flight conditions or 100 percent oxygen during emergencies, such as when there is smoke or fuel fumes in the cockpit. The three-position emergency toggle is used when symptoms of hypoxia appear. The two-position oxygen supply lever controls the starting and stopping of the oxygen flow.

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DESCRIPTION AND OPERATION (Continued)

OXYGEN REGULATORS (Cont'd)

When it is necessary to use the emergency toggle lever, the lever is pushed from the center or neutral position to EMERGENCY. It will remain in this position until it is manually changed. The operation of this EMERGENCY position can be checked by pulling (and holding) the lever to the TEST MASK position.

OXYGEN HIGH-PRESSURE GAUGE

A sensing transmitter, immediately downstream from the oxygen storage cylinder outlet, senses the cylinder pressure and transmits the information to a high-pressure gauge on the passengers' oxygen control panel. Both of these units are powered by the 28 VAC bus. This gauge indicates to the pilot the oxygen cylinder oxygen pressure in psi (0 to 1800) before the oxygen is reduced to using pressure and inform him of the amount of oxygen remaining for use.

OXYGEN PRESSURE REGULATOR

The oxygen pressure regulator is installed forward of the oxygen storage cylinder, in the aft lower left side of the passenger compartment, between fuselage stations 289 and 324. It is part of an assembly of units encased in a metal container. The container has an oxygen flow inlet (1800 psi), two outlets (flightcrew and passengers' oxygen flow), and an electrical connection for the solenoid-actuated valve. The units within the container are a pressure regulator, an electrically actuated solenoid shutoff valve, and an aneroid-type flow regulator. The pressure regulator regulates the pressure from 1800 psi to 65-95 psi. The solenoid-actuated shutoff valve is powered by the 28 VDC primary bus, through a circuit controlled by the passengers' oxygen control switch and the actuation of an aneroid-type altitude switch. The aneroid-type flow regulator is vented to cabin atmosphere and controls the flow of oxygen to the passengers' oxygen masks.

MANUAL SHUTOFF VALVE

The manual shutoff valve, attached to the oxygen bottle, contains a high-pressure gauge and a high-pressure relief valve. The relief valve is a frangible disc-type valve that will rupture under a pressure of 2700 to 2900 psi. If the relief valve ruptures, the oxygen will be vented overboard at station 302 on the left side of the fuselage. A green disk in the fuselage skin will rupture at 22 to 55 psi, providing a visual indication that the relief valve has ruptured.

SURGE VALVE

The surge valve is located forward of the oxygen pressure control regulator. The surge valve is connected in series with the solenoid bypass inlet port and passenger distribution port of the oxygen pressure control regulator. The surge valve provides a flow of oxygen pressure (65 to 95 psi) to the passengers oxygen system. This initial pressure is used to actuate the oxygen mask compartment door locks. The valve will automatically reset after the initial pressure surge.

BLEED ORIFICE

A bleed orifice, installed in the passengers oxygen supply line between the surge valve and the oxygen masks, is provided to relieve the surge pressure in the oxygen supply line which opens the passengers oxygen compartment doors. The bleed orifice will relieve pressure at the rate of between 1cc to 10cc per minute. When the surge pressure has been relieved, the oxygen compartment doors can be closed and latched.

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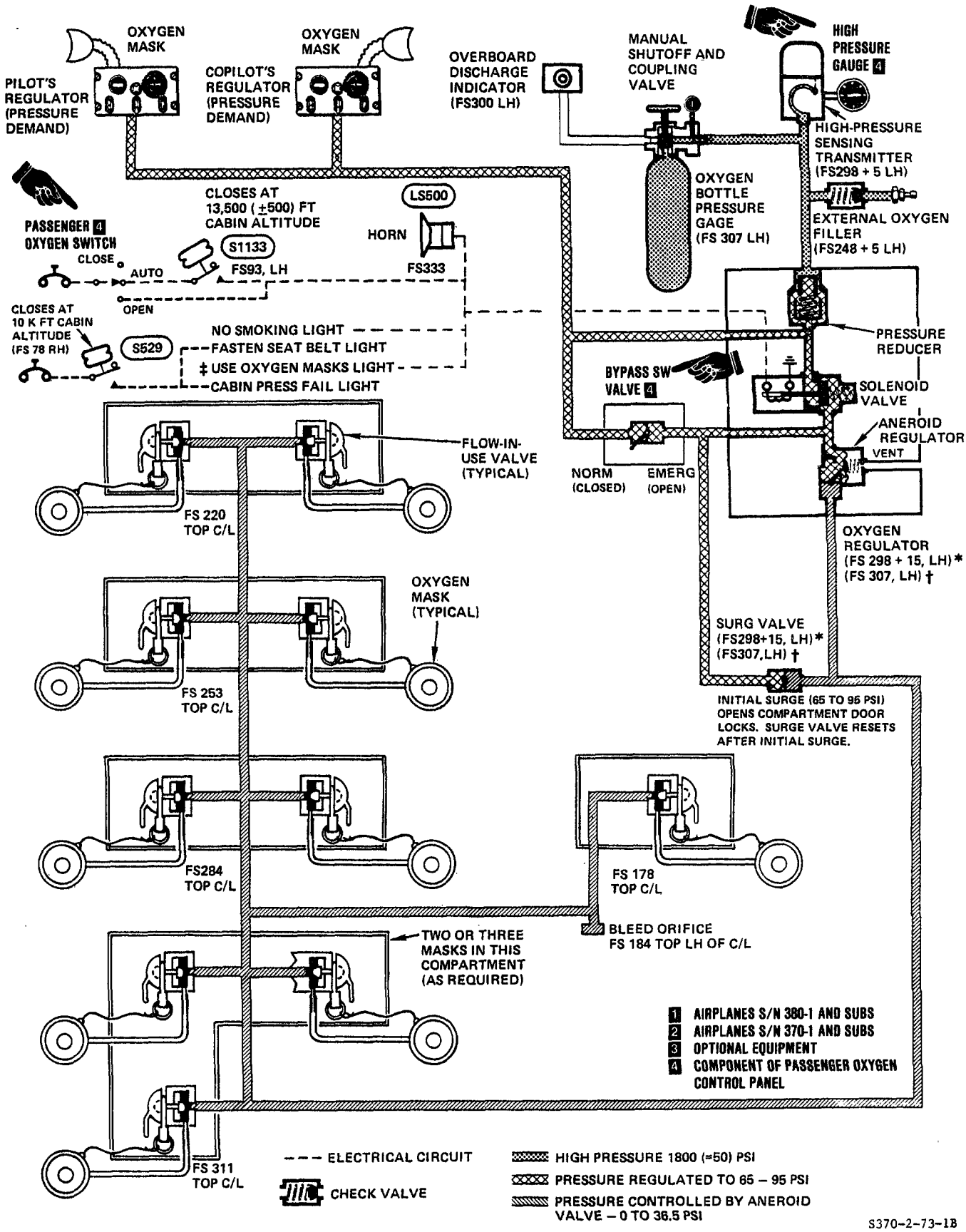
OXYGEN MASK
OXYGEN MASK
PILOT'S REGULATOR (PRESSURE DEMAND)
COPILOT'S REGULATOR (PRESSURE DEMAND)
OVERBOARD DISCHARGE INDICATOR (FS300 LH)
MANUAL SHUTOFF AND COUPLING VALVE
HIGH PRESSURE GAUGE 4
HIGH-PRESSURE SENSING TRANSMITTER (FS298 + 5 LH)
OXYGEN BOTTLE PRESSURE GAGE (FS 307 LH)
EXTERNAL OXYGEN FILLER (FS248 + 5 LH)
PASSENGER 4 OXYGEN SWITCH
CLOSE
AUTO
OPEN
CLOSES AT 13,500 (±500) FT CABIN ALTITUDE
S1133
FS93, LH
LS500
HORN
FS333
CLOSES AT 10 K FT CABIN ALTITUDE (FS 78 RH)
S529
NO SMOKING LIGHT
FASTEN SEAT BELT LIGHT
‡ USE OXYGEN MASKS LIGHT
CABIN PRESS FAIL LIGHT
PRESSURE REDUCER
BYPASS SW VALVE 4
SOLENOID VALVE
ANEROID REGULATOR
VENT
NORM (CLOSED)
EMERG (OPEN)
FLOW-IN-USE VALVE (TYPICAL)
FS 220 TOP C/L
OXYGEN REGULATOR (FS 298 + 15, LH)* (FS 307, LH) †
OXYGEN MASK (TYPICAL)
SURG VALVE (FS298+15, LH)* (FS307,LH) †
FS 253 TOP C/L
INITIAL SURGE (65 TO 95 PSI) OPENS COMPARTMENT DOOR LOCKS. SURGE VALVE RESETS AFTER INITIAL SURGE.
FS284 TOP C/L
FS 178 TOP C/L
BLEED ORIFICE FS 184 TOP LH OF C/L
TWO OR THREE MASKS IN THIS COMPARTMENT (AS REQUIRED)
1 AIRPLANES S/N 380-1 AND SUBS
2 AIRPLANES S/N 370-1 AND SUBS
3 OPTIONAL EQUIPMENT
4 COMPONENT OF PASSENGER OXYGEN CONTROL PANEL
FS 311 TOP C/L
ELECTRICAL CIRCUIT
CHECK VALVE
HIGH PRESSURE 1800 (=50) PSI
PRESSURE REGULATED TO 65 – 95 PSI
PRESSURE CONTROLLED BY ANEROID VALVE – 0 TO 36.5 PSI

S370-2-73-1B

Figure 35-00-1 — Oxygen System Flow Schematic

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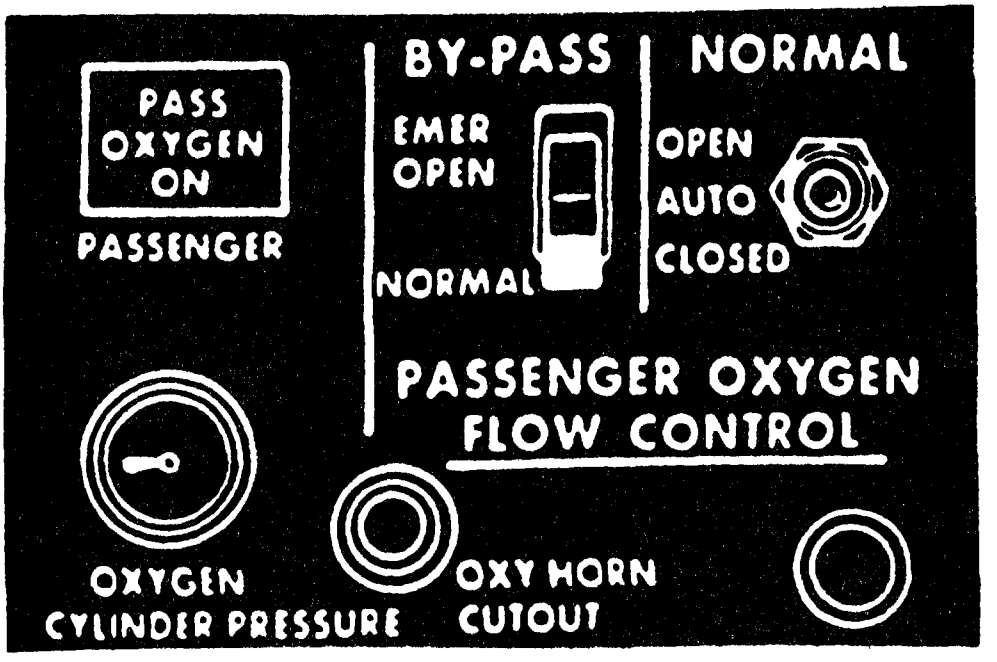
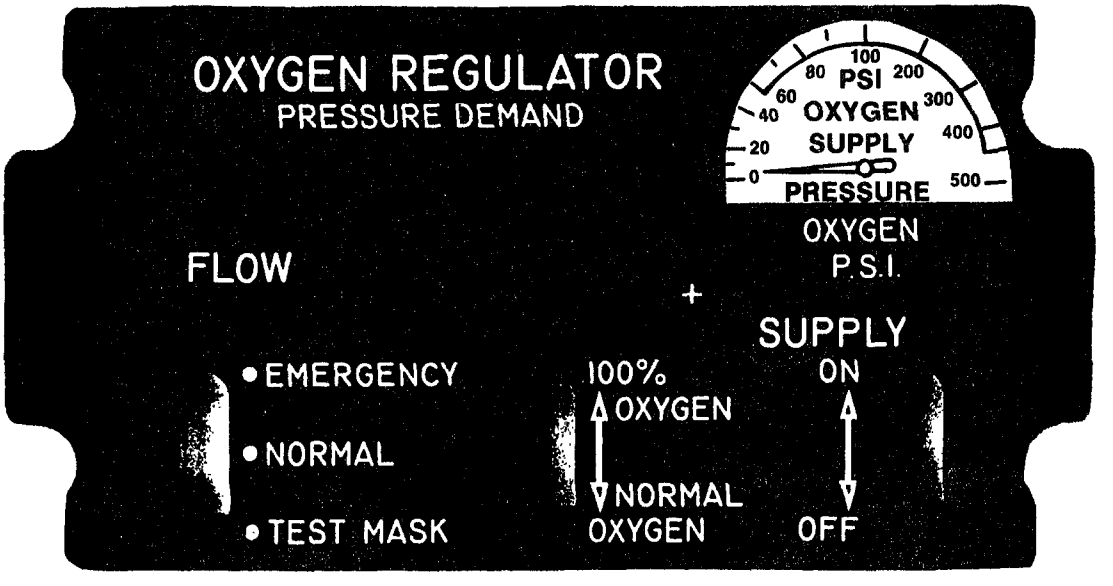


P/N 282-733022-11, -31



P/N 14950-33
All Airplanes Except S/N 370-2 and EROS Sys. Mod.

Figure 35-00-2. — Oxygen Control Panels

NEW

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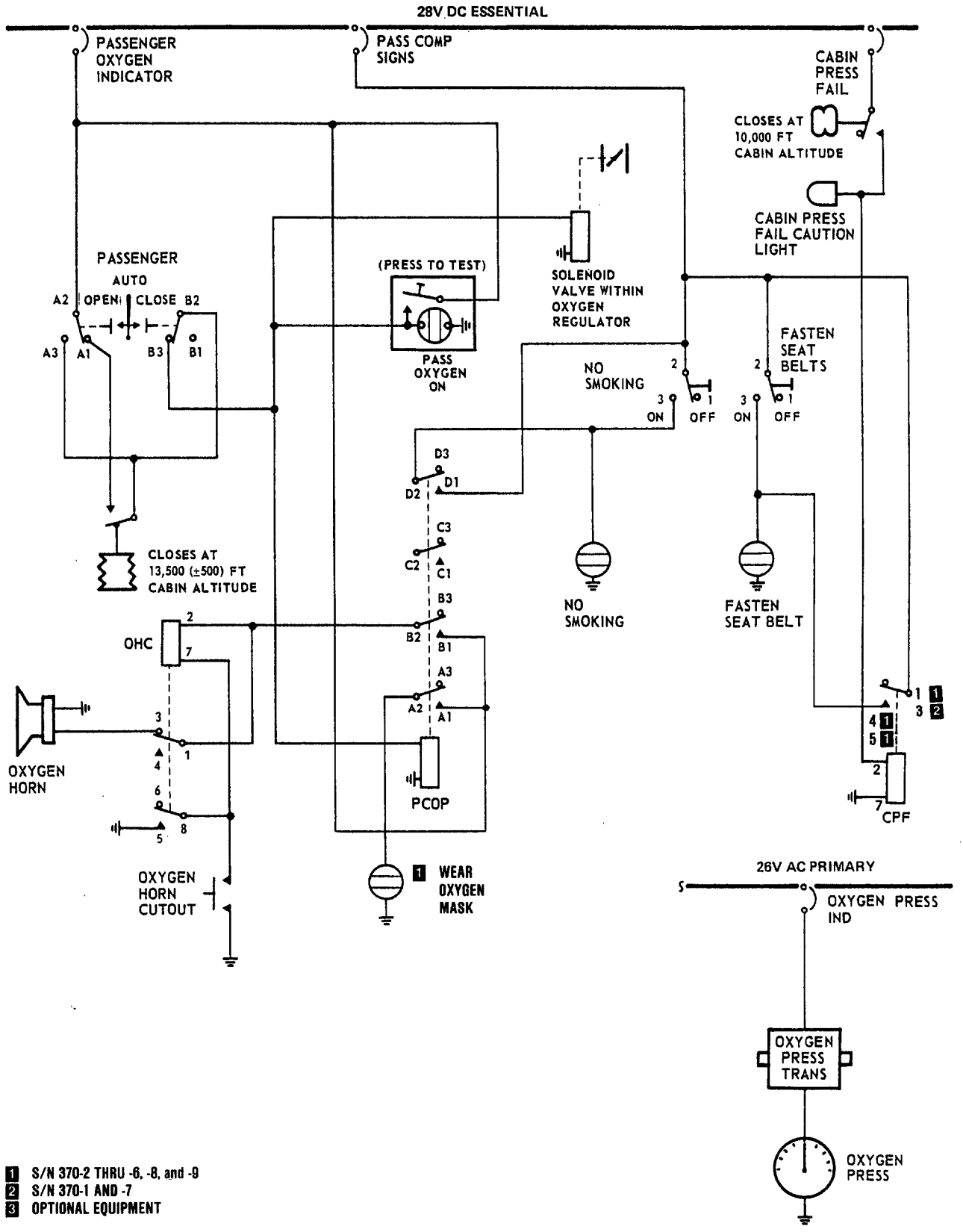

Figure 35-00-3 — Oxygen System Electrical Schematic — S/N 370-1 and Subs

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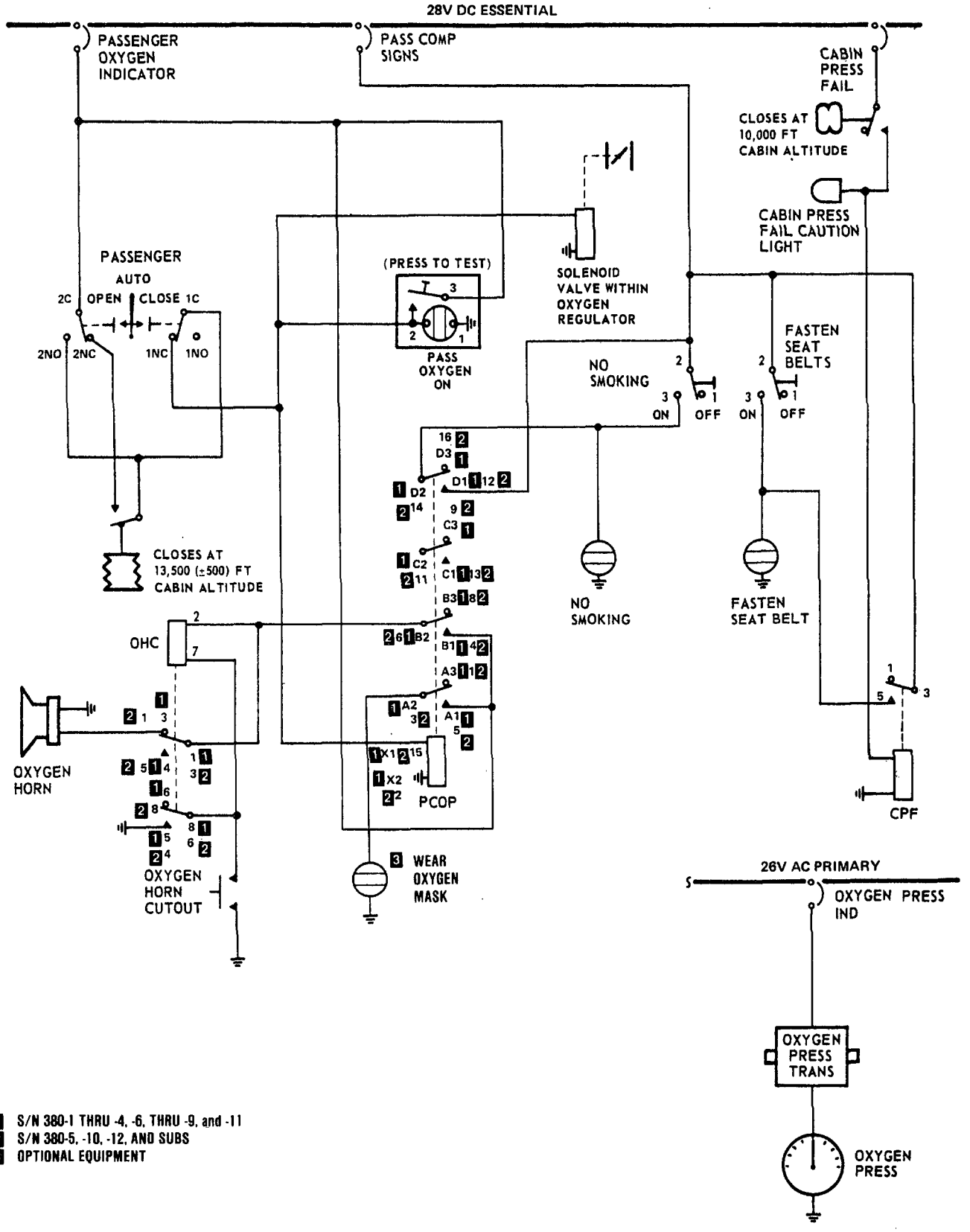

Figure 35-00-4 — Oxygen System Electrical Schematic — S/N 380-1 and Subs

S380-2-72-4A

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OXYGEN - TROUBLE SHOOTING

1. Trouble Shooting Oxygen System

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. OXYGEN PRESSURE DROP ON GROUND.		
(1) After filling, a normal loss of about 50 pounds pressure because of system cooling.	Mark position of indicator on glass of indicator with grease pencil. Check gage in one hour. If there has been no further drop, initial drop was due to system cooling after filling.	Refill system to 50 pounds in excess of desired pressure.
(2) Leak in oxygen system.	Check system for leaks.	Repair or replace defective units as necessary.
(3) Emergency toggle lever in pilot's or co-pilot's oxygen regulator in EMERGENCY or TEST MASK position.	Check end of mask-to-regulator hose for oxygen flow.	Place lever at NORMAL and refill system. Replace regulator if toggle lever sticks in TEST MASK position.
B. EXCESSIVE OXYGEN PRESSURE DROP DURING FLIGHT.		
(1) Ruptured line.	Inspect system visually and also soap-test.	Replace defective line.
(2) Ruptured ARO 14950-(), regulator diaphragm.	Check regulator for leak.	Replace regulator.
(3) Ill-fitting oxygen mask.	Check with pilot or copilot; then test masks.	Replace masks.
C. ARO 14950-33 REGULATOR FAILS TO DELIVER OXYGEN.		
(1) Shipping plugs not removed from regulator before installation.	Remove regulator and examine openings for presence of plugs.	Remove plugs.
(2) Puncture in demand diaphragm of regulator.	Blow gently into open end of regulator-to-mask hose. Any indication of free passage of air through the regulator indicates a damaged diaphragm.	Replace regulator.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
D. FLOW INDICATOR DOES NOT INDICATE OXYGEN FLOW.		
(1) Defective ARO 14950-() flow indicator.	Place diluter lever at 100% OXYGEN and oxygen emergency lever at TEST MASK. Check end of mask-to-regulator hose for oxygen flow. If oxygen flows, flow indicator is defective.	Replace regulator.
E. IN-USE VALVE DOES NOT CUT OFF OXYGEN FLOW.		
(1) Compression spring inside in-use valve broken.	Compress valve. If there is no spring-back, spring is defective.	Replace valve.

OXYGEN - MAINTENANCE PRACTICES

1. Servicing Oxygen System

If the equipment is available, the oxygen cylinder may be refilled while installed in the airplane. If equipment is not available, the oxygen cylinder must be removed from the airplane to be refilled.

MAINTENANCE PRACTICES (Continued)

REPLACEMENT OF OXYGEN REGULATOR

Removal:

1. Make sure manual override lever is at CLOSED and cabin oxygen pressure shutoff valve is at OFF.
2. Close manually operated on-off valve on oxygen cylinder. This will shut off oxygen flow from cylinder.
3. Place oxygen supply regulator lever at ON, and emergency lever at EMERGENCY. Allow oxygen to flow until portion of system forward of oxygen control valve is empty.
4. Remove side panels as required.

NOTE

When lines are removed, immediately cap or plug all "B" nuts and fittings. Do not use masking tape.

5. When removing copilot's regulator, disconnect outlet line at regulator outlet. When removing pilot's regulator, disconnect "B" nut from fitting on bottom, inboard side of regulator.
6. Follow electrical leads to terminal block. Cut spot ties; then disconnect wires from terminal block. When removing copilot's regulator, disconnect hose positioning clamp for accessibility to terminal block.
7. Unfasten the four Airloc fasteners on regulator panel.
8. Lift regulator straight up from console. When removing copilot's regulator, disconnect "B" nut from inlet fitting on bottom, outboard side of regulator. When removing pilot's regulator, disconnect outlet line from regulator outlet.
9. Remove regulator assembly.

Installation:

1. Remove caps or coverings from regulator inlet and outlet fittings.
2. Remove plug from regulator inlet hose when copilot's regulator is being installed; then connect hose to fitting on bottom outboard side of regulator. Remove covering from regulator outlet hose when pilot's regulator is being installed; then connect outlet hose to regulator outlet.
3. Position regulator in place and fasten the four Airloc fasteners.
4. Connect electrical leads to terminal block. (Refer to Wiring Diagram Manual.) Spot-tie electrical leads. When copilot's regulator is being installed, install hose clamp removed to gain access to terminal block.
5. When copilot's regulator is being installed, remove covering from regulator outlet hose; then connect outlet hose to outlet fitting. When pilot's regulator is being installed, remove plug from regulator inlet hose; then connect inlet hose to inlet fitting. Torque "B" nut.
6. If oxygen system supply has been depleted longer than 2 hours, purge system with breathing oxygen.
7. Test regulator for leaks according to procedures for the oxygen regulator operational check.

REPLACEMENT OF OXYGEN CONTROL VALVE

Removal:

1. Remove all power from airplane.
2. Close manually operated on-off valve to shut off oxygen pressure to control valve.
3. Remove console side panel.
4. Remove pilot's regulator. (Refer to procedures for removing pilot's and copilot's oxygen regulators.)

NOTE

When lines are removed, immediately cap or plug all "B" nuts and fittings. Do not use masking tape.

5. Disconnect "B" nut from aft fitting, on bottom of passenger's pressure shutoff valve.
6. Unfasten the four Airloc fasteners on panel.
7. Pull assembly aft until wrench can be put on "B" nut oxygen control valve lower, forward 90-degree fitting.
8. Disconnect "B" nut from top fitting on oxygen control valve lower, forward 90-degree fitting.
9. Remove assembly. Remove line from shutoff valve to control valve. Identify line to ensure proper reinstallation.

Installation:

1. Remove caps or coverings from panel assembly.
2. Connect flex line to passenger compartment shutoff valve "IN" fitting and to passengers' oxygen control valve 90-degree upper fitting. Torque "B" nuts.
3. Remove plug from oxygen pressure line leading to oxygen control valve 90-degree lower fitting.
4. Place panel assembly in console and attach oxygen pressure line "B" nut to 90-degree lower fitting. Torque "B" nut.
5. Position panel in console, and tighten the four Airloc fasteners.

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MAINTENANCE PRACTICES (Continued)

Installing Oxygen Control Valve (Cont'd)

6. Remove plug from shutoff valve "out" line. Attach "out" line "B" nut to shutoff valve aft fitting. Torque "B" nut.
7. Install pilot's oxygen regulator. (Refer to procedures for installing oxygen regulator.)
8. If oxygen system supply has been depleted for more than 2 hours, the oxygen system must be purged.
9. Test installed units for leaks. (Refer to procedures for leak-test of oxygen system.)
10. Deleted.

REPLACEMENT OF RUPTURED OXYGEN SYSTEM HIGH-PRESSURE RELIEF INDICATOR DISK

Removal:

The high-pressure relief indicator is on the left side of the fuselage beneath the left engine pylon.

1. Remove snap ring from opening in indicator.
2. Remove all disk fragments which may remain in indicator opening.

Installation:

1. Position disk in opening of indicator.
2. Install snap ring in opening of indicator.

HYDROSTATIC CHECK OF OXYGEN BOTTLES

Standard weight cylinders identified by an ICC or DOT 3AA 1800 or 2100 classification must be hydrostatic tested at the end of each 5-year period. The light weight cylinders identified by an ICC or DOT 3 HT 1850 classification must by hydrostatic tested every 3 years and must be retired from service after 15 years or 4,380 pressurizations whichever occurs first.

REPLACEMENT OF PORTABLE OXYGEN BOTTLE

Removal

1. Remove portable oxygen bottle from holder. Remove bottle from airplane.

WARNING

Keep oil, grease, and hydraulic fluid clear of oxygen system components to prevent serious injury to personnel.

2. Exhaust oxygen from bottle, then remove hose and mask.
3. Check hydrostatic test date on bottle. Hydrostatic test as required.

Installation

1. Install portable oxygen bottle in holder.
2. Install hose and mask to bottle.
3. Service bottle as required (refer to chapter 12).

REPLACEMENT OF OXYGEN SYSTEM BOTTLE

Removal

1. Remove cabin trim as necessary to gain access to oxygen bottle.
2. Close oxygen bottle manual shutoff valve.
3. Exhaust system pressure through pilot's oxygen regulator.
4. Disconnect oxygen lines from oxygen bottle.
5. Remove retaining straps from bottle. Remove bottle from airplane.
6. Exhaust oxygen from bottle.
7. Check hydrostatic test date on bottle. Hydrostatic test as required.

Installation

1. Install oxygen bottle in bracket and connect oxygen lines.
2. Tighten oxygen bottle retaining clamps.
3. Open oxygen bottle manual shutoff valve.
4. Service oxygen system as required (refer to chapter 12).
5. Perform leak-test of system as required.

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1. Leak-testing Lines, Valves, and Fittings

A. To leak-test valves, lines, or fittings, proceed as follows:

(1) Check for leakage, using a castile soap solution. Carefully wipe parts clean and dry immediately after test has been completed. Leaks in flared tubing connections which require more than 10 seconds to produce bubbles 1/4 inch in diameter, or more than one minute to produce bubbles 1/2 inch in diameter, normally will not require repair, unless there are too many.

(2) Tighten excessively leaking flared fittings to the maximum torque as follows:

	MIN TORQUE	MAX TORQUE
1/4-inch OD tubing	80 in. lb	110 in. lb
5/16-inch OD tubing	110 in. lb	125 in. lb

(3) When stopping leaks, start at minimum torque and tighten fittings 25 inch-pounds at a turn until leak is stopped or maximum torque is reached. If this practice is followed, it is possible in many cases to stop leaks several turns before maximum torque is reached.

(4) Spread soap solution over threaded pipe connections. All leaks in these connections must be stopped. If leakage cannot be stopped without excessive torque, disassemble connection and replace fitting. When leaks are being stopped, threaded pipe connections may be tightened 25 inch-pounds at a time until the leak is stopped, or until the following maximum torque is reached:

	MIN TORQUE	MAX TORQUE
1/8-inch pipe	125 in. lb	175 in. lb

(5) Tighten silver-brazed copper tubing to the following torque value:

	MIN TORQUE	MAX TORQUE
3/16-inch OD tubing	50 in. lb	70 in. lb

CAUTION: PARTS WILL RUST IF NOT DRIED IMMEDIATELY AFTER TESTING.

2. Leak-test of Oxygen System

A. To leak-test oxygen system, proceed as follows:

(1) Fill oxygen bottle to 1800 psi, indicated by gage on bottle.

(2) Record pressure indicated on gage attached to bottle, or mark face of gage.

(3) After 24 hours, recheck pressure gage and record drop in pressure.

NOTE: Less time may be allowed for this test by proportioning the allowable leakage into an hourly leakage pressure drop.

(4) Allowable pressure drop over a 24-hour period should not exceed 118 psi or 5 psi per hour.

NOTE: Corrections for change in temperature should be computed from the following table.

CHANGE IN TEMPERATURE	CHANGE IN PRESSURE
5°F	17 psi
10°F	34 psi
15°F	51 psi
20°F	68 psi
25°F	85 psi
30°F	102 psi
35°F	119 psi
40°F	136 psi
45°F	152 psi

(5) If value obtained in step (4) is within tolerance, the system is sufficiently leak-tight.

(6) If value obtained in step (4) is not within tolerance, determine source of leakage by application of castile soap solution around all fittings and connections. Make necessary repairs and repeat leak-test procedure.

NOTE: Do not deplete the system to less than 50 psi unless it is necessary to replace a part.

MAINTENANCE PRACTICES (Continued)

GROUND OPERATIONAL CHECK OF PILOT'S AND COPILOT'S OXYGEN REGULATORS

1. Fill oxygen storage cylinder to 1800 (±50) psi.
2. Position pasengers' oxygen control switch at AUTO and BY-PASS valve at NORMAL.
3. Engage PASS OXY IND and OXYGEN PRESS IND circuit breakers.
4. Energize 28-VDC essential bus and 26-VAC bus.
5. Check OXYGEN CYLINDER PRESSURE gauge on PASSENGER OXYGEN FLOW CONTROL panel for an indication of 1800 (±50) psi.

NOTE

The following checks are given for the pilot's regulator only. Repeat procedure for the copilot's regulator. Disconnect masks for the following tests to prevent mask contamination.

6. Check for 65 to 95 psi oxygen pressure on OXYGEN REGULATOR. Pressure to remain within this range for the following checks:
7. Position "SUPPLY" lever to ON and diluter lever to NORMAL.
8. Position emergency lever to TEST MASK. Oxygen should flow from oxygen mask hose. Release emergency lever. The emergency lever should return to NORMAL position and oxygen flow should stop.
9. Position emergency lever to EMERGENCY. The emergency lever should remain in the EMERGENCY position and oxygen should flow from oxygen mask hose.
10. Position emergency lever to NORMAL. Oxygen flow from oxygen mask hose should stop.
11. Position diluter lever to 100% OXYGEN. Inhale normally a minimum of three times from end of oxygen mask hose. FLOW indicator should alternate from black to white during each inhalation.
12. Inhale and hold breath. Indicator should go to and remain black. A white indicator indicates a leak.
13. Repeat steps 11 and 12 with diluter lever in NORMAL OXYGEN position.
14. Position SUPPLY lever to OFF and emergency lever at TEST MASK. Hold TEST MASK lever until oxygen stops flowing from oxygen mask hose. If oxygen flow does not stop, the supply valve of the oxygen regulator is defective.
15. Position SUPPLY lever to ON, and emergency lever to EMERGENCY. Place thumb over open end of oxygen mask hose. The FLOW indicator should not blink. If FLOW indicator blinks, check for system leakage and/or a defective regulator.
16. DELETED
17. Connect oxygen masks to supply hoses.

GROUND CHECK OF PASSENGERS' OXYGEN AND WARNING SYSTEM

1. Fill oxygen storage cylinder to 1800 (±50) psi. Check indicating gauge on pilot's oxygen control panel for indication of 1800 (±50) psi.
2. Push in the PASS OXY IND and PASS COMP SIGNS circuit breakers.
3. Energize the 28-VDC essential buss and 26-VAC buss.
4. Position passengers' oxygen NORMAL control switch to OPEN and check for the following conditions:

   Check that oxygen warning sounds.

   Depress "OXY HORN CUTOUT" button on PASSENGER OXYGEN FLOW CONTROL panel. Warning horn should cease to sound.

   Check that "WEAR OXYGEN MASK" (if installed) and "NO SMOKING" lights are on.

   Check that "PASS OXYGEN ON" light on PASSENGER OXYGEN FLOW CONTROL panel is ON.

   Check that passengers' oxygen masks have fallen out of their respective compartments.

   Remove flow-in-use valve pin from one oxygen mask position, and check for oxygen flow. Replace flow-in-use valve pin.
5. Position passengers' oxygen NORMAL control switch to AUTO.
6. Check that the WEAR OXYGEN MASK, NO SMOKING and PASS OXYGEN ON lights go out.
7. After approximately two minutes, replace flow-in-use valve pin. The two minute time period allows the system pressure to bleed off.
8. Replace oxygen masks in one compartment only and close compartment door.
9. Allow a minimum of 5 minutes after step 5; then position BY-PASS control lever on PASSENGER OXYGEN FLOW CONTROL panel to EMER OPEN.
10. Oxygen masks packed in step 8 should fall out.

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MAINTENANCE PRACTICES (Continued)

GROUND CHECK OF PASSENGERS' OXYGEN AND WARNING SYSTEM (Cont'd)

11. Position BY-PASS control lever on PASSENGER OXYGEN FLOW CONTROL panel to NORMAL.
12. Replace oxygen masks in their respective compartments, after two minutes.
13. Remove electrical power from airplane and service oxygen system as necessary.

GROUND OPERATIONAL CHECK OF OXYGEN REGULATORS (S/N 370-1, -7, 380-1 and Subs)

1. Check oxygen storage cylinder for 1500 to 1800 psi and service as required.
2. Apply 28VDC power to the airplane.
3. Push in the PASS OXY IND and OXYGEN PRESS IND circuit breakers.
4. Place PASSENGER switch to AUTO.
5. Check OXYGEN CYLINDER PRESSURE gauge on CABIN OXYGEN FLOW CONTROL panel for indication of 1500 to 1800 psi.

NOTES

- ***Steps 6 thru 18 are for pilots regulator only. Repeat these steps for copilots regulator.***
- ***Disconnect masks for the following tests to prevent mask contamination.***

6. Check OXYGEN REGULATOR pressure for 65 to 95 psi.

NOTE

Pressure must remain between 65 and 95 psi for the following checks.

7. Position the following controls:
   A. SUPPLY lever ..... ON
   B. Diluter lever ..... NORMAL OXYGEN
   C. EMERGENCY lever ..... TEST MASK
8. Check for oxygen flow at mask hose
9. Release EMERGENCY lever and check that lever returns to NORMAL and oxygen flow stops.
10. Position EMERGENCY lever to EMERGENCY and check that lever stays in EMERGENCY and oxygen flows from mask hose.
11. Return EMERGENCY lever to NORMAL and check that oxygen flow has stopped.
12. Position diluter lever to 100% OXYGEN and inhale normally a minimum of three times from end of mask hose.
13. Flow indicator should blink from black to white with each breath.
14. Inhale and hold breath. Check that indicator blinks to black and stays black (a white indicator indicates a leak).
15. Repeat steps 12 thru 14 with diluter lever in NORMAL OXYGEN position.
16. Position SUPPLY LEVER to OFF and hold emergency lever to TEST MASK until oxygen flow stops. If oxygen flow does not stop, the supply valve in the oxygen regulator is defective.
17. Position SUPPLY lever ON and emergency lever to EMERGENCY while holding your thumb over open end of mask hose.
18. Check that flow indicator does not blink. If indicator blinks check for system leak or defective regulator.
19. Position PASSENGER oxygen switch to OPEN and check for the following:
    A. Oxygen warning horn sounds.
    B. Depress OXY HORN CUTOUT button and horn should silence.
    C. Check NO SMOKING, FASTEN SEAT BELT, OXYGEN ON lights are illuminated.
    D. Check that all masks have dropped out of their storage boxes.
20. Return PASSENGER oxygen switch to CLOSE.
21. Replace oxygen masks in their respective compartments.
22. Check complete system for security and service as required.

CHECKING OXYGEN SYSTEM DURING FLIGHT

The oxygen storage cylinder must be serviced to 1800 psi for the performance of an oxygen check during flight. The check is to be conducted at 20,000 feet airplane altitude. The flight crew should wear oxygen masks, and the person performing the inspection in the cabin should be on oxygen, using a walk-around bottle. To check oxygen system during flight, proceed as follows:

1. Make sure passengers' oxygen control switch is positioned at AUTO, NO SMOKING and FASTEN SEAT BELT switches are at OFF, and cabin reading lights are off.
2. Position pressurization selector switch at OFF.

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MAINTENANCE PRACTICES (Continued)

CHECKING OXYGEN SYSTEM DURING FLIGHT (Cont'd)

3. Check the following items when cabin altitude increases to 10,000 (±400) feet:
   A. FASTEN SEAT BELT placard should light in cabin.
   B. CABIN PRESSURE FAIL and MASTER CAUTION lights should light on instrument panel.
4. Check the following items when cabin altitude increases to 13,500 (±500) feet:
   A. Hinged door in each passenger's oxygen mask compartment should be open.
   B. WEAR OXYGEN MASK (if installed) and NO SMOKING placards in cabin should light.
   C. Oxygen horn should sound.
   D. PASSENGER OXYGEN ON light should go on.
   E. Cabin reading lights or indirect lighting should go on.
5. Depress oxygen horn cutout button. Horn should cease operation.
6. Check for oxygen flow at each passenger's mask as follows:
   A. Hold mask in one hand; then pull in-use valve pin from valve.
   B. Check for oxygen flow.
   C. Reinsert in-use valve pin. Oxygen flow should stop.
7. Have pilot position passengers' oxygen control switch at CLOSE, then to OPEN; check for following items:
   A. WEAR OXYGEN MASK and NO SMOKING placards should remain on.
   B. Oxygen horn should sound. Depress horn cutout button.
   C. Pull in-use valve pin at one place only. Check for oxygen flow; then replace pin.
8. Have pilot position passengers' oxygen control switch at CLOSED. Check for the following items:
   A. WEAR OXYGEN MASK and NO SMOKING placards should go out.
   B. PASSENGER OXYGEN ON light should go out.
   C. Cabin reading lights should go out.
   D. Passengers' oxygen flow should stop.
9. Have pilot position passengers' oxygen control switch at AUTO. Check for the following items:
   A. WEAR OXYGEN MASK and NO SMOKING placards should go on.
   B. PASSENGER OXYGEN ON lights should go on.
   C. Reading lights should go on.
   D. Oxygen horn should sound. Depress cutout button.
10. Position environmental pressurization switch at BOTH ENG. As cabin pressurizes and cabin altitude decreases to less than 13,500 feet, check for the following items.
    A. Pull in-use valve pin at one mask position and check for no oxygen flow. Replace in-use pin.
    B. WEAR OXYGEN MASK and NO SMOKING placards, PASSENGER OXYGEN ON light, and cabin reading lights should go out.
11. After airplane lands, proceed as follows:
    A. Stow passengers' oxygen masks.
    B. Reservice oxygen system.

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1. Disassembly, Assembly and Cleaning of Passengers' Oxygen Masks

A. To disassemble oxygen masks, see figure 35-00-701 and proceed as follows:

(1) Disconnect mask from oxygen in-use-valve.

(2) Remove band (head strap) from retaining ring.

(3) Remove the three screws from retaining ring.

(4) Remove retaining ring from housing by holding retaining ring with fingers and pressing thumbs against housing.

(5) Remove retaining ring from bag; then, remove face cushion.

(6) Remove two screws securing bag to housing.

(7) Fold bag lengthwise and carefully pull through hole from inside housing.

B. To clean oxygen masks, proceed as follows:

(1) Using a clean container, add 1/2 oz. of detergent to one gallon of lukewarm water.

(2) Using a soft brush or towel, clean face cushion until all contamination is removed. Wipe dry.

NOTE: To accelerate drying time, use a low pressure oil-free compressed air or dry nitrogen.

(3) Spray internal surface of face cushion with antiseptic spray (Noxal Solution No. 2 or equivalent). Noxal Solution No. 2 may be procured from Noxal Products, Co., 618 West Garvey, Monterey Park, California.

(4) Apply talcum powder to external surface of face cushion.

(5) Clean retaining ring and housing in solution. Wipe and/or air dry as necessary.

CAUTION: TO PREVENT DAMAGE TO DIAPHRAGMS IN HOUSING, EXTREME CARE SHOULD BE USED WHEN CLEANING.

(6) Spray internal surface of housing with Noxal Solution No. 2 or equivalent.

(7) Clean bag assembly by applying a low pressure oil-free compressed dry nitrogen to inside of bag and hose. Clean inhalation valve and retainer (part of bag assembly) must be cleaned in solution, air dried, and sprayed with antiseptic spray.

C. To assemble masks, proceed as follows:

(1) Fold bag lengthwise (four or five times) along hose and hold in position.

(2) Feed bag through hole in housing from the back side and through retaining ring at the same time, until retainer on bag is seated and aligned with mounting holes in housing.

(3) Install and tighten two screws securing bag retainer to housing.

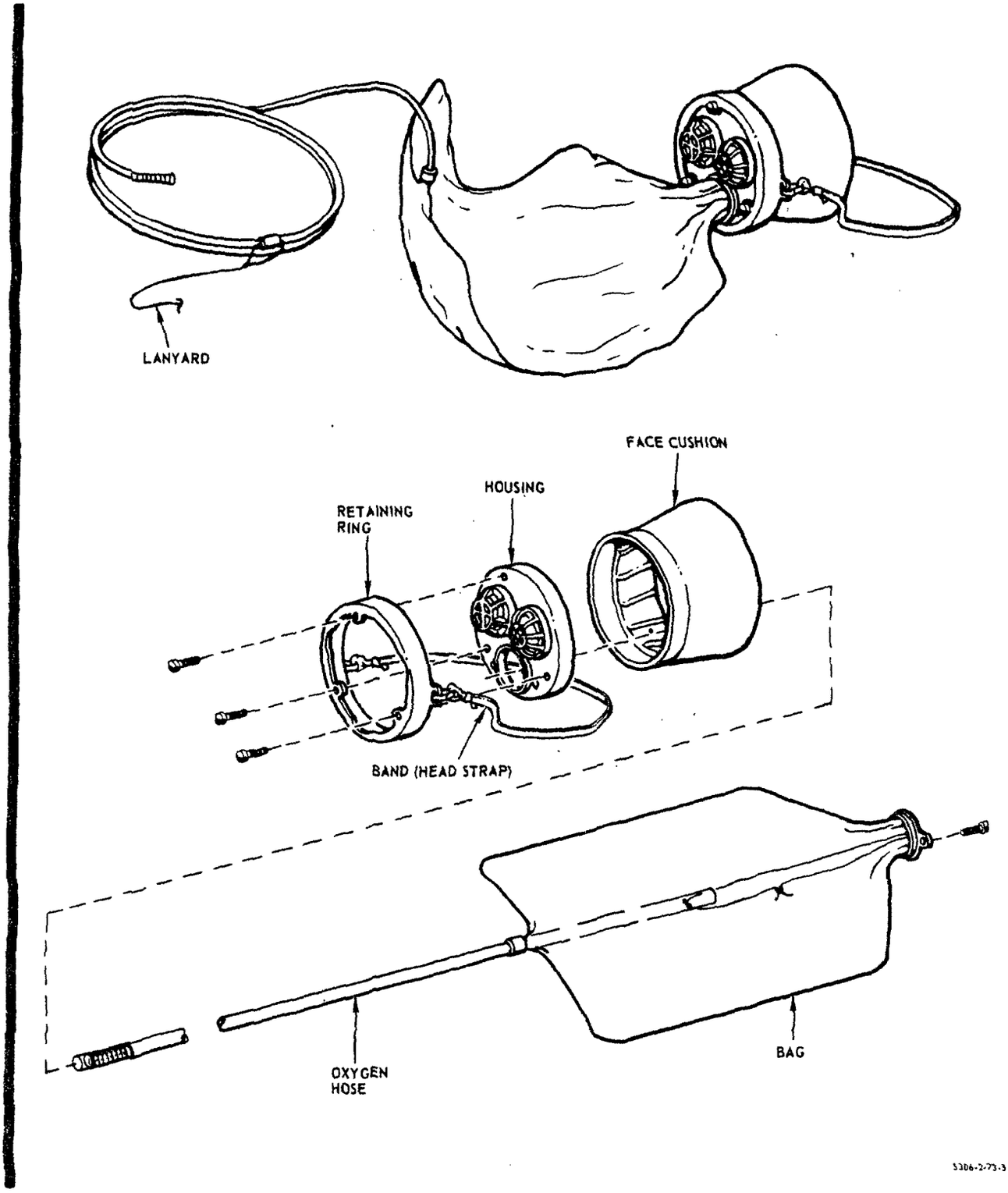
(4) Place small opening of face cushion around housing. Edge of face cushion should be even with front edge of housing.

(5) Align mounting holes in retaining ring with holes in housing; then, slide retaining ring over face cushion. Install and tighten screws in retaining ring.

(6) Attach band (head strap) to retaining ring.

2. Stowing Passengers' Oxygen Masks

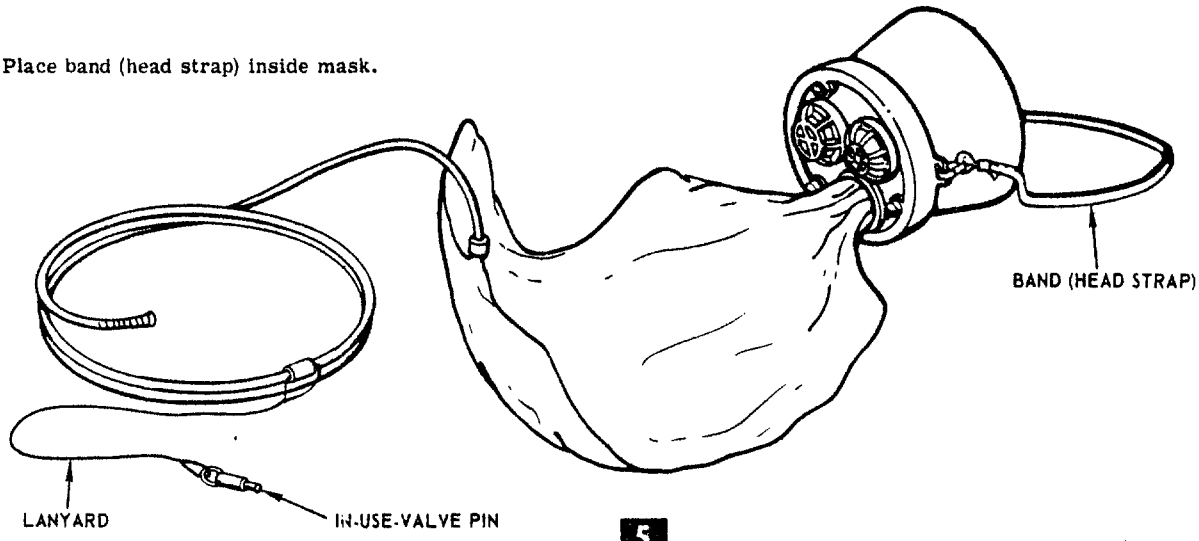
A. See figure 35-00-702.



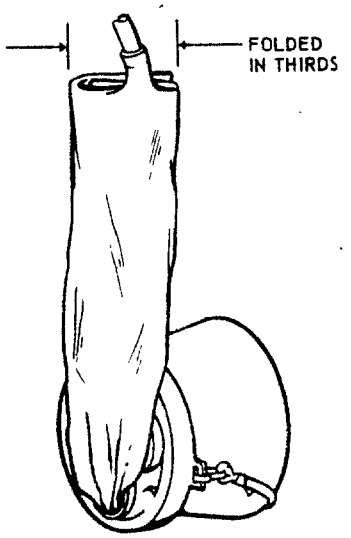
Passenger Oxygen Masks
Figure 35-00-701

After deployment of passengers oxygen mask, stow as follows:

1 Place band (head strap) inside mask.



2 Fold bag into thirds.

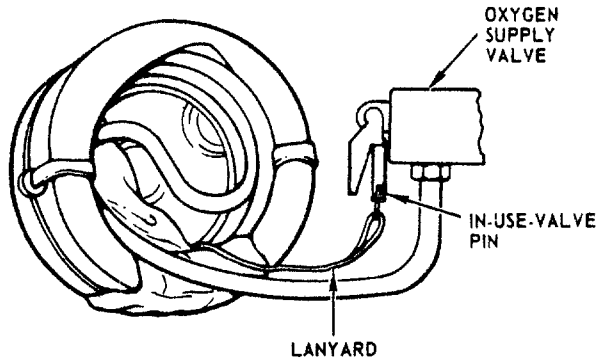


3 Wrap folded bag over nearest edge of mask; then nest into mask.

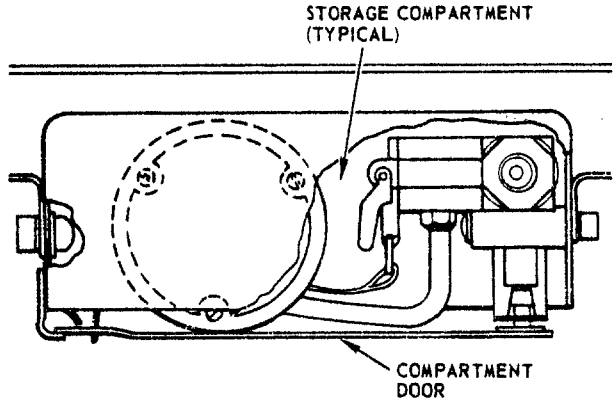
NOTE Keep in-use-valve pin and lanyard free of bag and coiled tube by draping over side of mask.

4 Coil tube into mask using nonlapping coils and avoiding kinks. A crossover coil may be used at approximately half-coil to reduce tendency of hose to twist or kink.

5 Insert in-use-valve pin in oxygen supply valve.



6 Position mask on edge inside oxygen mask compartment with both the last run of tube and the exposed bag toward lid of compartment.



7 Close compartment door.

5306-2-73-1

Stowing Passengers' Oxygen Masks
Figure 35-00-702

CHAPTER 36

PNEUMATIC

CHAPTER 36

PNEUMATIC

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The pneumatic system consists of the deicer boot system and the landing gear wheel seal system. For information and maintenance instructions, refer to chapters 30 and 32, respectively.

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CHAPTER

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VACUUM

CHAPTER 37

VACUUM

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The vacuum system is part of the deicer boot system. Refer to chapter 30 for information and maintenance instructions.

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WATER/ WASTE

CHAPTER 38

WATER/WASTE

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Information for this chapter to be supplied when equipment is installed. Reference vender data for specific equipment installed.

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CHAPTER

49

AIRBORNE AUXILIARY POWER

CHAPTER 49

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	27	Oct 31/74
	28	Oct 31/74
	29	Oct 31/74
	30	Oct 31/74
	31	Oct 31/74

*The asterisk indicates pages revised, added, or deleted by the current revision.

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AUXILIARY POWER UNIT - DESCRIPTION

1. General

NOTE: This chapter provides maintenance instructions for the aircraft peculiar components only. For major maintenance instructions peculiar to the APU, refer to Solar Publication SD2-2049.

A gas turbine auxiliary power unit (APU) is installed in the aft fuselage equipment compartment, aft of fuselage station 357. (See figure 49-00-1). The APU is restricted to ground operation only. It may be operated with both engines inoperative or with the left engine operating only. The APU supplies the necessary bleed air to operate the aircraft air conditioning system. A limited amount of electrical power is also supplied to the aircraft electrical system. The electrical power is supplied by a starter - generator mounted on the right angle reduction drive pad of the APU. The starter-generator is also utilized for starting the APU. Additional major components added to the aricraft by the APU installation are the relay and power junction boxes, generator exhaust and cooling air duct, fire extinguishing and detector systems, fuel supply system, air intake and exhaust outlet, and a control panel mounted in the cockpit on the right hand console. (See figure 49-00-1). The APU control panel consists of an ampmeter, tachometer, and an exhaust gas temperature indicator, and control switches and indicator lights necessary to monitor and control the APU during operation. (See figure 49-00-1).

Provisions are incorporated in the APU system for automatic shutdown whenever the right hand engine is started or in the event of a turbine overspeed or underspeed, excessive exhaust gas temperature, low oil pressure, turbine overheat, fire condition, starter-generator overheat, and a fire detector system power failure. Normal APU shutdown is accomplished by utilizing the turbine START-RUN-OFF switch located on the APU control panel.

The APU is supported within the aft fuselage equipment compartment at five points. Three of the tube supports are attached to the aft end of the APU and the remaining two tube supports are attached to center of the APU. The opposite ends of the tube supports are attached to the aircraft structure. To obtain proper alignment of the APU inlet and exhaust ducts, four of the tube supports are provided with adjustable clevis fittings at one end.

The APU bleed air system consists of ducting from the APU to the aircraft heat exchanger. A bleed air valve is installed in the ducting near the heat exchanger junction. The bleed air valve is controlled by the "BLEED AIR" switch, and on NA-265-80-1, -2, -3, -4, -6, -7, -8, -9 and -11 airplanes, the "EGT" switch on the cockpit control panel. A bleed air control unit is also installed in the system. Electrical signals from the APU thermocouples to the bleed air control unit also provide automatic control of the bleed air valve.

2. APU Drain Lines

The drain lines from the APU components are routed to a central drain manifold attached to the lower side of the fuselage at station 371.

The six drain lines terminating at the drain manifold, from forward to aft in order, are as follows:

A. Reduction Gear Box Vent

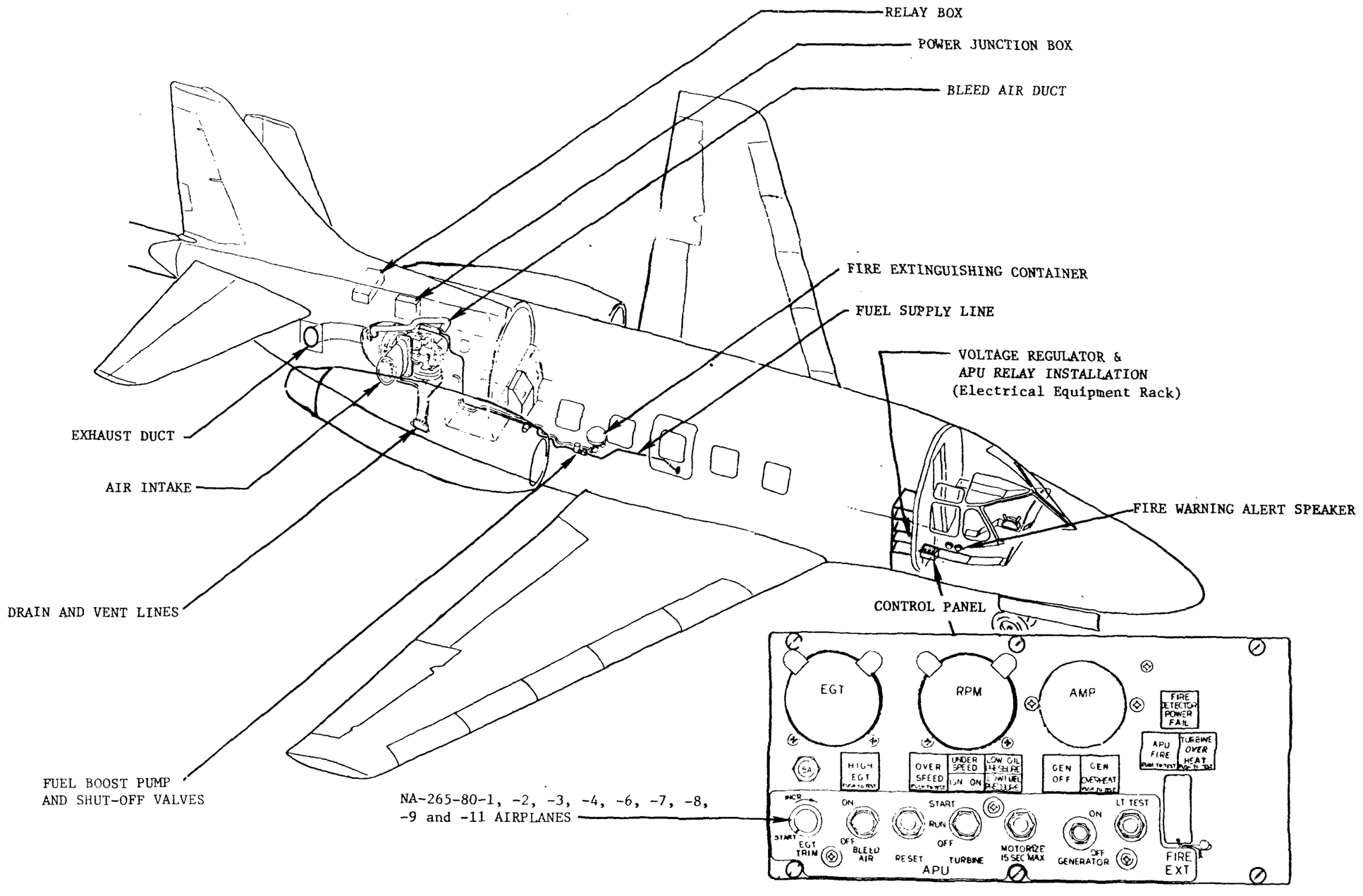
B. Oil Sump Pan Drain

C. Fuel Box Drain

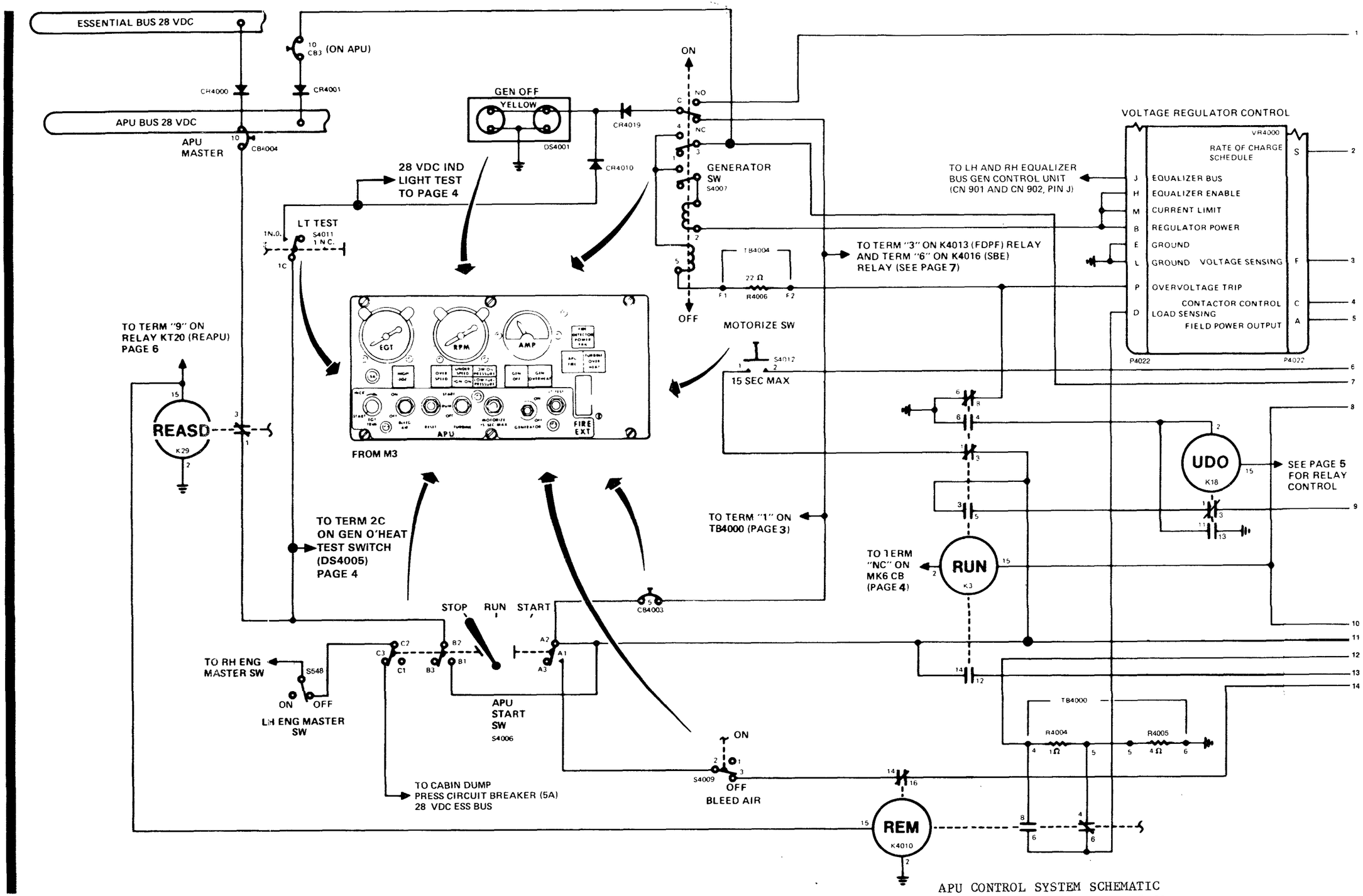
D. Fuel Containment Enclosure Drain

E. Combustor Shroud Drain

F. Start Fuel Nozzle Purge Vent



APU System Component Location
Figure 49-00-1



APU CONTROL SYSTEM SCHEMATIC
Figure 49-00-2 (Sheet 1 of 8)

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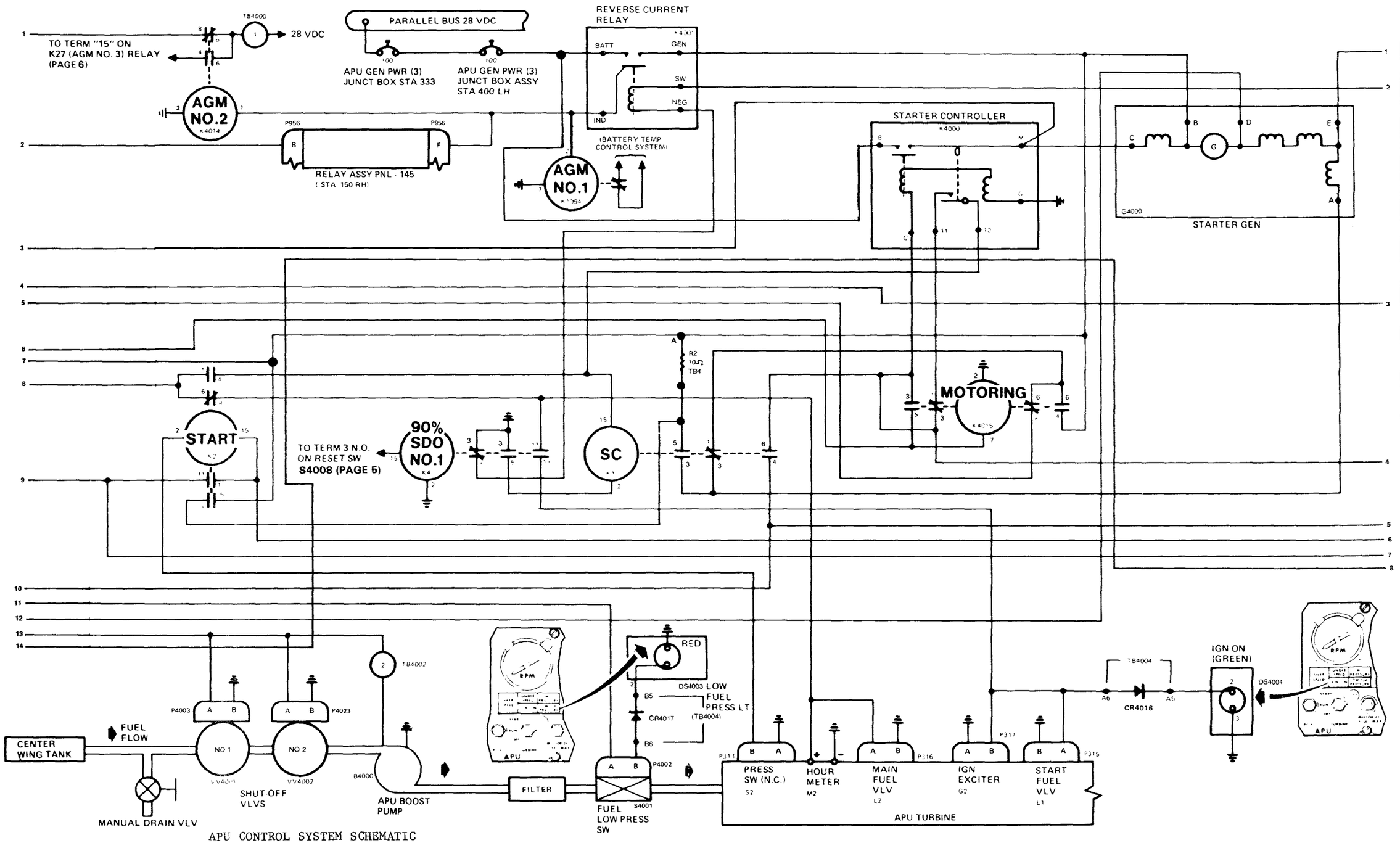

APU CONTROL SYSTEM SCHEMATIC
Figure 49-00-2 (Sheet 2 of 8)

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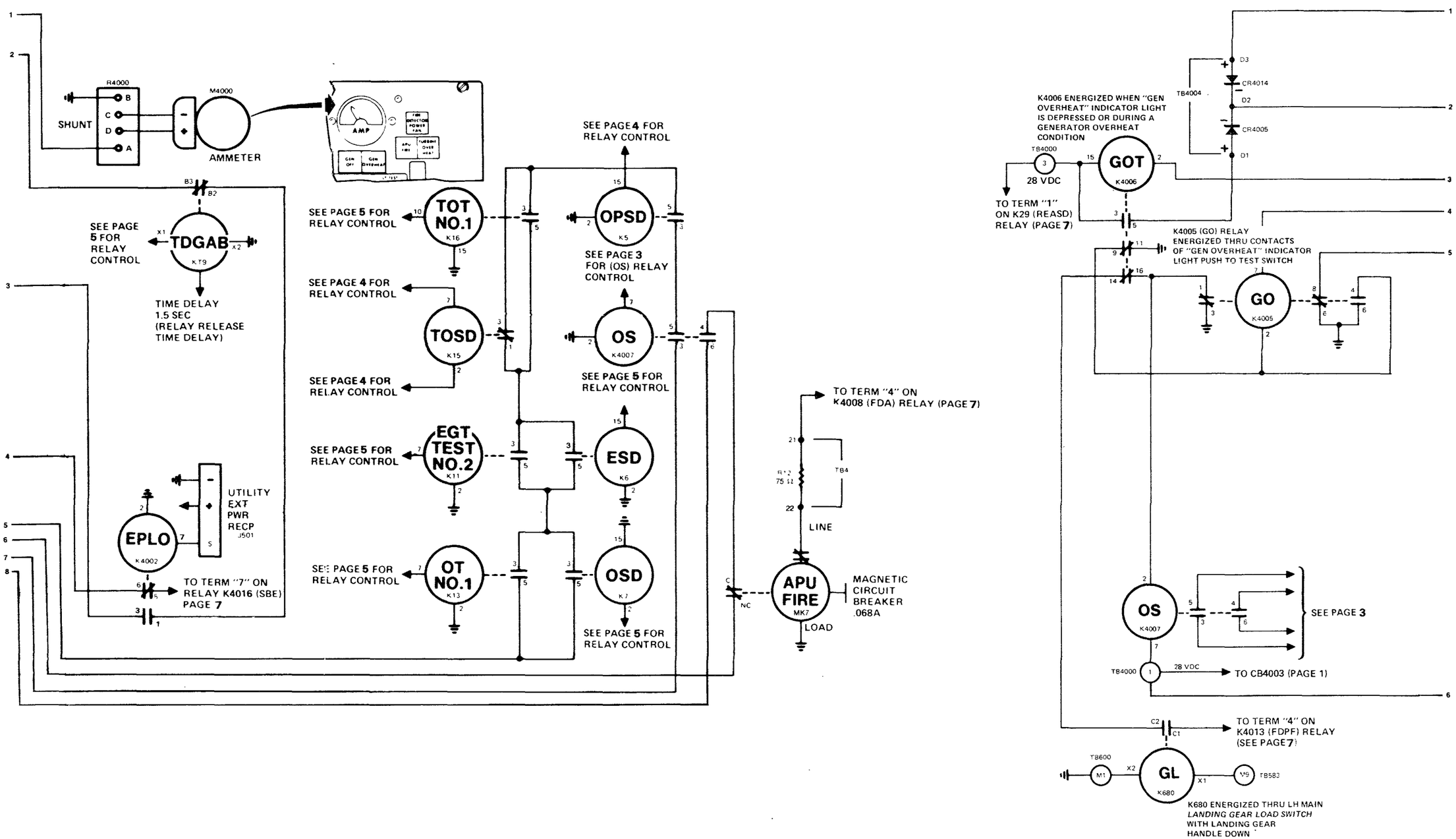

APU CONTROL SYSTEM SCHEMATIC
Figure 49-00-2 (Sheet 3 of 8)

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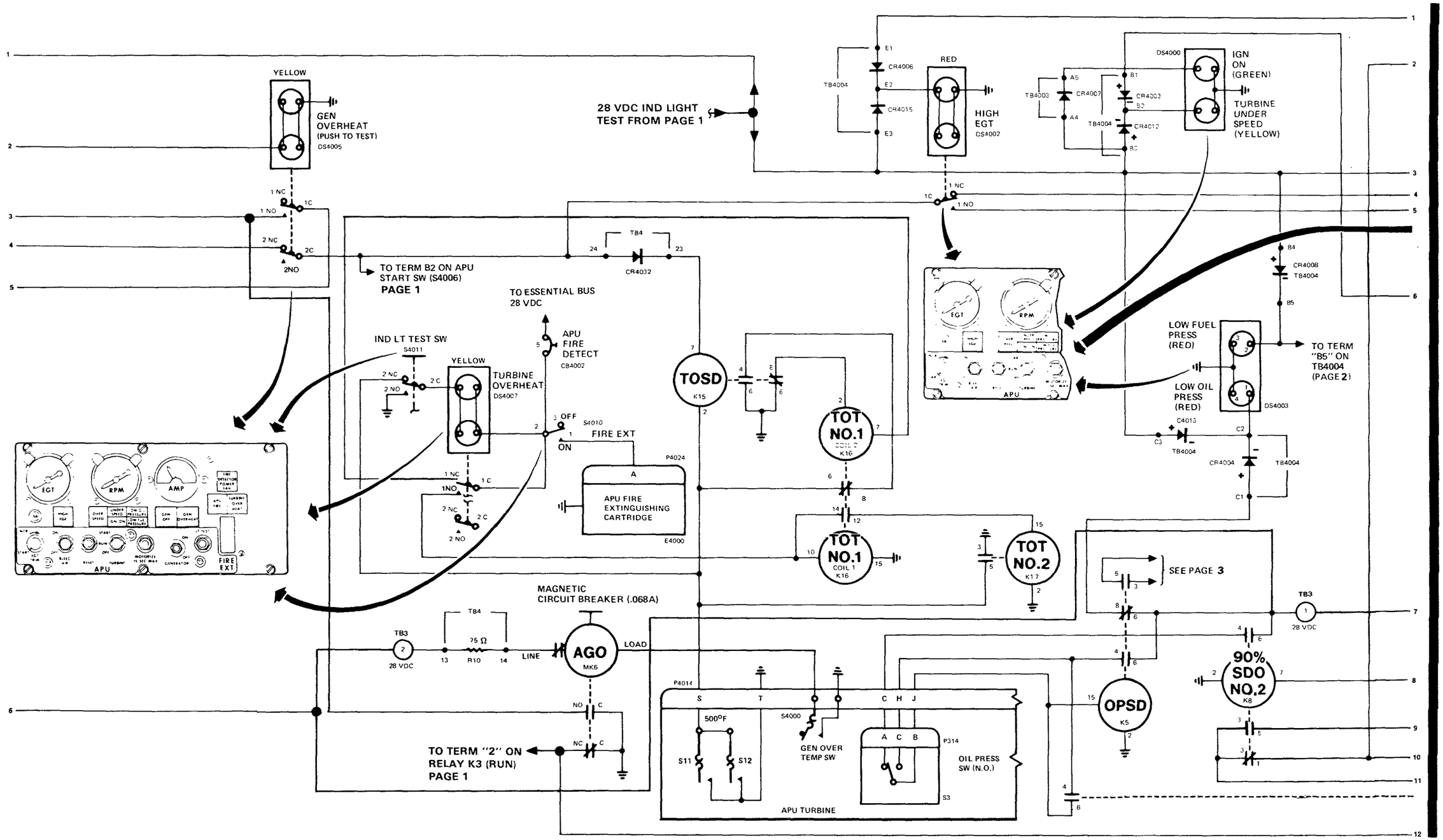

APU CONTROL SYSTEM SCHEMATIC
Figure 49-00-2 (Sheet 4 of 8)

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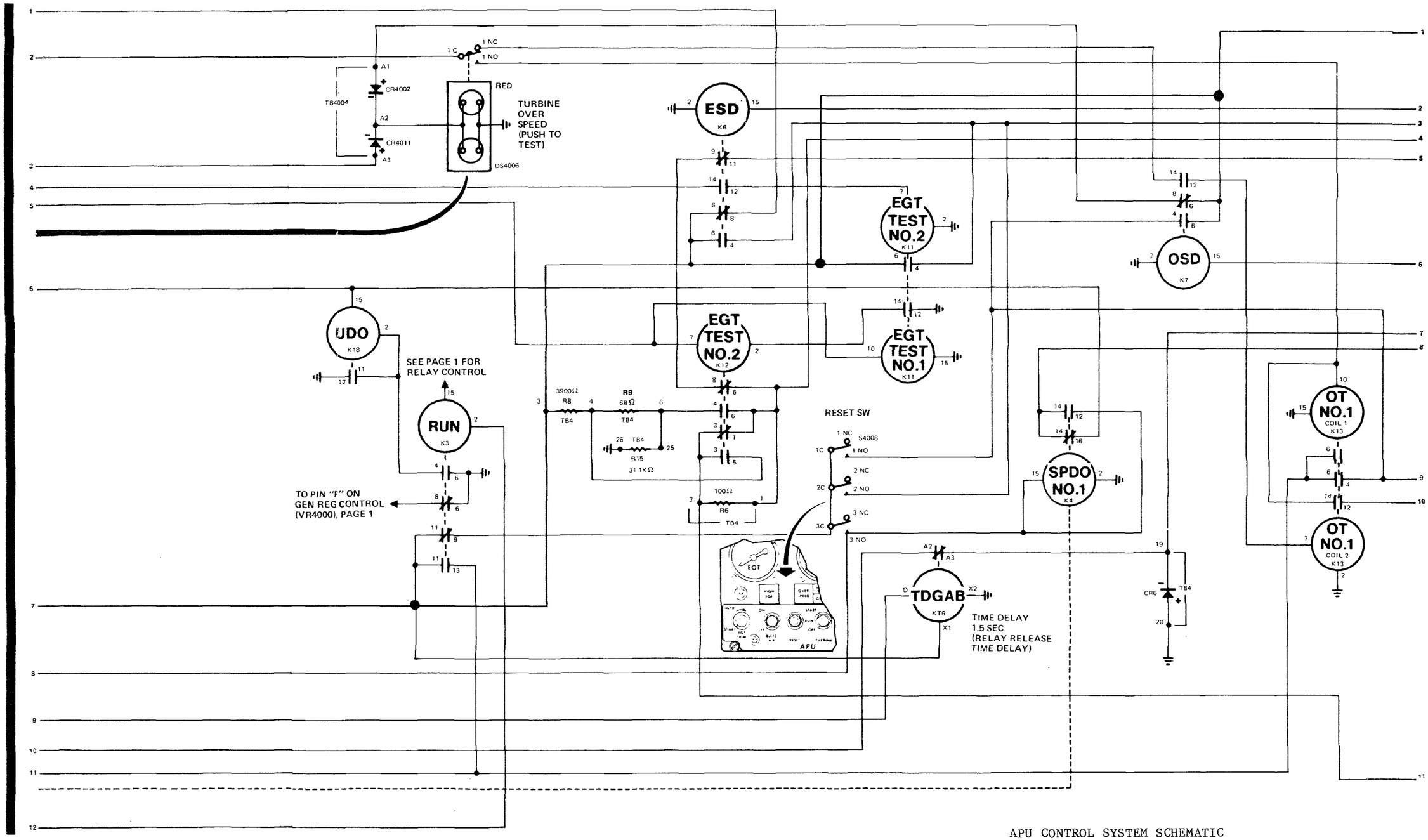

APU CONTROL SYSTEM SCHEMATIC
Figure 49-00-2 (Sheet 5 of 8)

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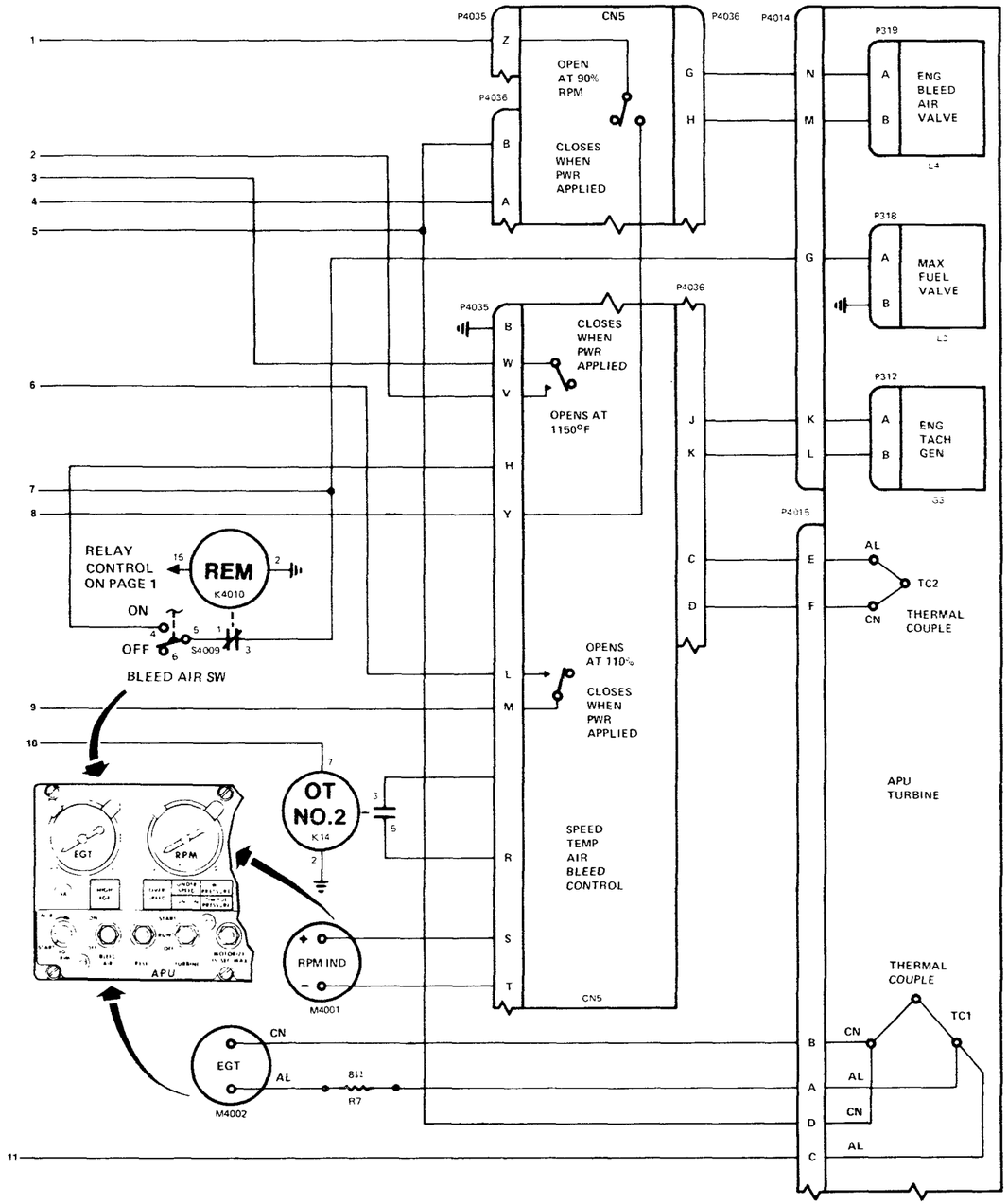

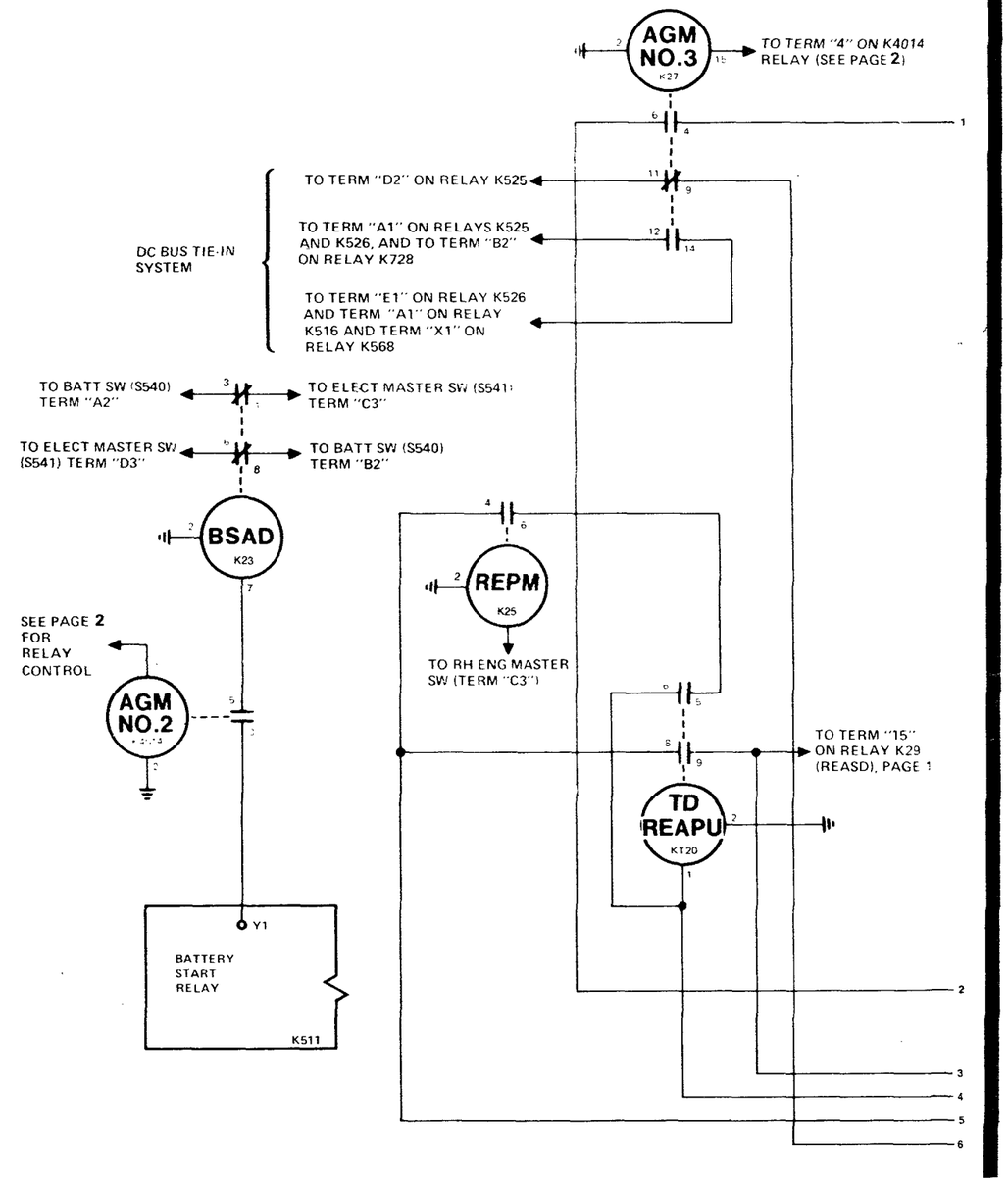
AGM NO.3
K27
TO TERM "4" ON K4014 RELAY (SEE PAGE 2)
DC BUS TIE-IN SYSTEM
TO TERM "D2" ON RELAY K525
TO TERM "A1" ON RELAYS K525 AND K526, AND TO TERM "B2" ON RELAY K728
TO TERM "E1" ON RELAY K526 AND TERM "A1" ON RELAY K516 AND TERM "X1" ON RELAY K568
TO BATT SW (S540) TERM "A2"
TO ELECT MASTER SW (S541) TERM "C3"
TO ELECT MASTER SW (S541) TERM "D3"
TO BATT SW (S540) TERM "B2"
BSAD
K23
SEE PAGE 2 FOR RELAY CONTROL
AGM NO.2
BATTERY START RELAY
K511
REPM
K25
TO RH ENG MASTER SW (TERM "C3")
TO TERM "15" ON RELAY K29 (REASD), PAGE 1
TD REAPU
KT20

APU CONTROL SYSTEM SCHEMATIC
Figure 49-00-2 (Sheet 6 of 8)

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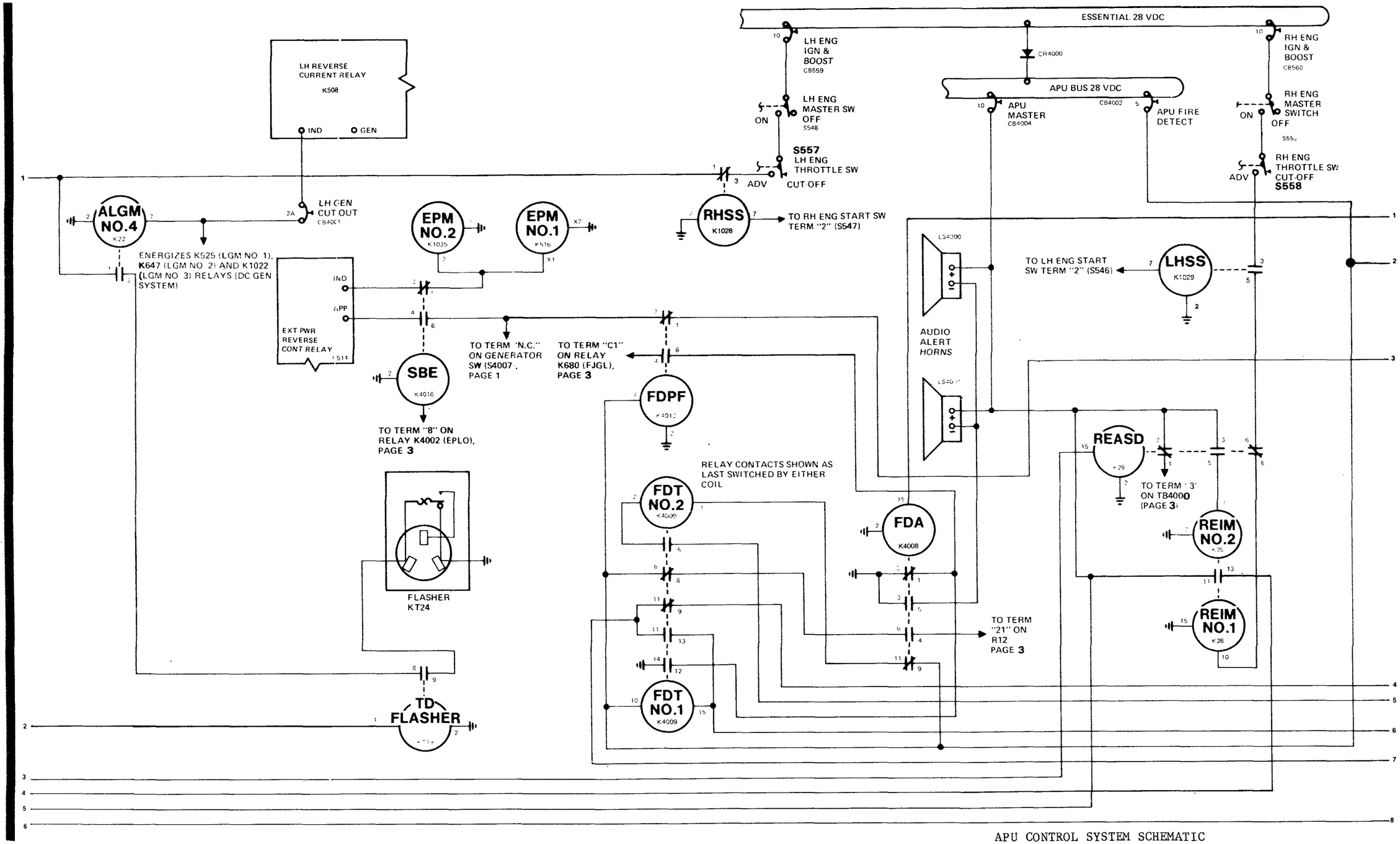

APU CONTROL SYSTEM SCHEMATIC
Figure 49-00-2 (Sheet 7 of 8)

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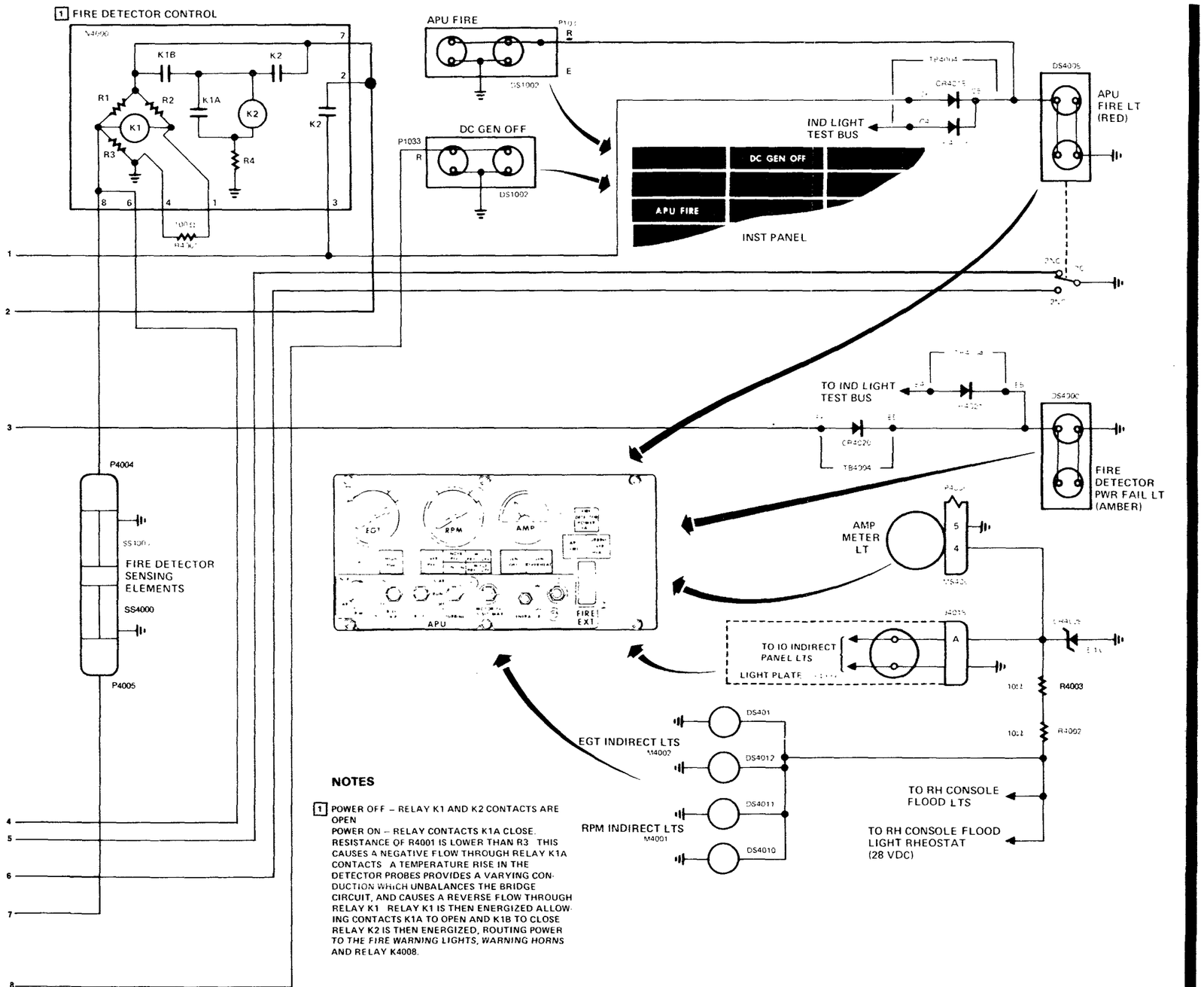

NOTES

1 POWER OFF – RELAY K1 AND K2 CONTACTS ARE OPEN
POWER ON – RELAY CONTACTS K1A CLOSE. RESISTANCE OF R4001 IS LOWER THAN R3 THIS CAUSES A NEGATIVE FLOW THROUGH RELAY K1A CONTACTS A TEMPERATURE RISE IN THE DETECTOR PROBES PROVIDES A VARYING CONDUCTION WHICH UNBALANCES THE BRIDGE CIRCUIT, AND CAUSES A REVERSE FLOW THROUGH RELAY K1 RELAY K1 IS THEN ENERGIZED ALLOWING CONTACTS K1A TO OPEN AND K1B TO CLOSE RELAY K2 IS THEN ENERGIZED, ROUTING POWER TO THE FIRE WARNING LIGHTS, WARNING HORNS AND RELAY K4008.

APU CONTROL SYSTEM SCHEMATIC
Figure 49-00-2 (Sheet 8 of 8)

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AUXILIARY POWER UNIT GENERAL - MAINTENANCE PRACTICES

1. Prevention of Foreign-Object Damage

To prevent foreign-object damage to the APU, suitable inlet and exhaust covers should be installed at all times during maintenance periods. The inlet duct must be inspected for foreign objects, loose fasteners, etc., after maintenance has been performed on the aircraft or APU. Under no circumstances should the inlet or exhaust ducts be used as a resting place for tools, parts, nuts and bolts, etc., during maintenance operations. An inventory of all tools, parts, and equipment must be conducted after maintenance operation is completed.

1. Removal of APU

(1) Remove all electrical power from airplane.

(2) Disconnect fire detector cables at connector.

(3) Remove fire detector cable from APU.

CAUTION: TO PREVENT DAMAGE TO DETECTOR CABLE, USE EXTREME CARE WHEN REMOVING FIRE DETECTOR CABLE FROM APU.

(4) Remove shroud from exhaust pipe outlet. Then, remove clamps from exhaust pipe and remove exhaust pipe outlet from turbine.

(5) Remove clamps from inlet duct parting surface seal. Then, slide seal outboard.

(6) Remove starter-generator. (Refer to 49-10.)

(7) Remove clamp from bleed air outlet duct.

(8) Disconnect drain and vent lines (6).

NOTE: The drain shroud around the fuel inlet fitting must be removed prior to disconnecting fuel line.

(9) Loosen clamp on oil sump drain pan. Then, remove drain pan.

(10) Disconnect electrical plugs from turbine.

(11) Inspect complete assembly to ensure all items have been disconnected or removed.

(12) While an assistant supports the APU assembly, disconnect support tubes.

(13) Remove APU from airplane.

2. Installation of APU

(1) Lift APU into position. While assistance supports assembly, connect the support tubes to turbine housing. Tighten support tube retaining nuts.

NOTE: Minor adjustments to the support tubes may be required to align the air inlet and exhaust outlet to the aircraft ducts.

(2) Install exhaust pipe and secure with 2 clamps.

(3) Install shroud assembly over exhaust pipe.

(4) Install air inlet seal and secure with two clamps.

(5) Connect electrical plugs to turbine.

(6) Install starter-generator. (Refer to 49-10.)

(7) Connect bleed air duct to turbine outlet.

(8) Install oil sump drain pan and secure with clamp.

(9) Install drain shroud around inlet fuel fitting. Connect drain and vent lines.

CAUTION: TO PREVENT DAMAGE TO DETECTOR CABLE, USE EXTREME CARE WHEN INSTALLING CABLE.

(10) Install fire detector cable assembly in support clamps.

(11) Inspect complete installation to ensure that all components are properly installed and secured.

(12) Perform operational check-out of APU.

1. APU Starting Procedures

A. Preoperational Check of APU

Before the initial start of a replacement (new or overhauled) APU or starter-generator, a preliminary cold run of the APU is required. Perform a preoperational check of APU as follows:

(1) Service APU oil system.

(2) Remove protective covers from inlet and exhaust outlets if installed. Check both openings for foreign material.

(3) Install chocks at aircraft wheels.

(4) Connect external power source to external power start receptacle.

NOTE: Ensure that the external power source is capable of providing 24 volt dc and 400 amperes minimum during starting.

(5) Check that all applicable circuit breakers are engaged (pushed in).

(6) Disconnect electrical plug (P317) at ignition exciter box. Cap both end of disconnected cable.

WARNING: TO PREVENT POSSIBLE SERIOUS INJURY TO PERSONNEL DO NOT TOUCH OR HOLD APU IGNITER LEADS OR CONNECTORS.

(7) Move "ELECTRIC MASTER" switch to ON.

(8) Move "BATTERY" switch to ON. Check that "LOW FUEL PRESS" and "GEN OFF" indicator lights come on.

(9) Position "APU TURBINE" switch at RUN.

(10) Ensure that "GENERATOR" and "BLEED AIR" switches are OFF.

(11) On NA-265-80-1, -2, -3, -4, -6, -7, -8, -9 and -11 airplanes move "EGT TRIM" switch to START.

(12) Depress and hold the "LT TEST" switch. Check that all indicator lights (11) come on. Release switch.

(13) Depress "HIGH EGT", "GEN OVERHEAT", "APU FIRE" and "TURBINE OVERHEAT" indicator lights and check that each light comes on while depressed.

(14) Momentarily depress "APU RESET" switch.

(15) Momentarily depress "MOTORIZE" switch. Check for proper rotation of APU (clockwise as viewed from rear) and for any unusual noises. Release the "MOTORIZE" switch.

NOTE: If any rubbing of the compressor rotor-turbine unit is detected investigate cause before proceeding with check.

(16) Depress "MOTORIZE" switch and hold until engine speed stabilizes. Monitor and record rpm and length of time, in seconds, required to reach the stabilized rpm. Ensure that starter operation does not exceed 120 seconds in any 20 minute period.

CAUTION: IF 30% RPM IS NOT REACHED, THE APU IS NOT SAFE TO START.

(17) To determine the starting characteristics of the APU when using fully charged aircraft batteries, proceed as follows:

(a) Disconnect external power source connected to aircraft.

(b) Repeat steps (7) thru (16).

(18) Move "APU TURBINE" switch to OFF.

(19) Move "BATTERY" and "ELECTRIC MASTER" switch to OFF.

(20) Connect electrical plug (P317) to ignition exciter box.

B. Operational Check of APU

An operational check of the APU is mandatory following the required pre-operational check. The following procedures are applicable to a newly installed APU, postflight checks or scheduled APU maintenance. To start APU, proceed as follows:

(1) Check APU inlet area (including inlet screen) for any foreign objects which could be drawn into inlet duct. Also check oil level of lubricating system. Service if necessary.

(2) Connect external power source to external power start receptacle.

NOTE: Ensure that the external power source is capable of providing 24 volt dc and 400 amperes minimum during starting.

(3) Move "BLEED AIR" and "GENERATOR" switches to OFF.

(4) On NA-265-80-1, -2, -3, -4, -6, -7, -8, -9 and -11 airplanes, move "EGT TRIM" control to START.

(5) Check that "APU TURBINE" switch is OFF.

(6) All applicable circuit breakers - IN.

(7) Position "CABIN TEMP" switch of the Crew Compartment Environmental and Pressurization Control Panel to MANUAL COLD and hold for a minimum of 20 seconds. The CABIN AIR switch must be at BOTH ENG prior to positioning of CABIN TEMP switch.

(8) Move aircraft "BATTERY" and "ELECTRIC MASTER" to ON.

NOTE: The "BATTERY" and "ELECTRIC MASTER" switches must be left on during APU operation to provide fire detection and extinguishing system with power.

(9) Depress "LT TEST" switch. Check that all indicator lights come on.

NOTE: During APU operation, the APU and related systems should be monitored.

(10) Move "APU TURBINE" switch to RUN.

(11) Momentarily depress "APU RESET" button.

(12) Depress PUSH TO TEST indicator lights. All lights will come on except "OVER SPEED" indicator light. This light will be checked after APU start.

(13) Move "APU TURBINE" switch to START and release. Switch will return to RUN position.

(14) Monitor EGT indicator. If lightoff does not occur within 5 to 10 seconds, move "APU TURBINE" switch to OFF, and proceed as follows:

(a) If APU start was aborted, reposition "APU TURBINE" switch to "RUN" and depress and hold "MOTORIZE" switch to motor APU for 15 seconds to remove as much fuel as possible. Wait at least 30 seconds for combustor to drain before attempting another start.

(b) Remove any puddles of fuel which may have accumulated in exhaust outlet.

(c) The turbine switch when positioned to START energizes the starter, arms the ignition circuit and APU fuel control valves, opens the two fuel shutoff valves, energizes the electrically operated fuel line fuel boost pump which supplies fuel under pressure to the APU fuel pump system.

(d) If APU fails to start after three successive attempts, purge the fuel system.

(e) Note that GEN OFF light is illuminated.

(f) Note that IGN ON light comes on at about 5% RPM and goes out at about 90% RPM.

(15) If a successful lightoff occurs, the starter should continue to operate assisting the engine to accelerate until the starter controller relay opens which in turn deenergizing the starter. If starter disengages before 80% rpm is reached, a hung start may occur.

(16) Monitor "EGT" and "RPM" indicator during acceleration. Exhaust gas temperature should not exceed 1100°F and rpm should not exceed 103%. If either or both of these conditions occur, abort start and refer to APU Exhaust Gas Temperature Limitation.

WARNING: IF AN APU FIRE OR A TURBINE OVERHEAT CONDITION OCCURS, FOLLOW PROCEDURES GIVEN FOR FIRE IN APU COMPARTMENT OR OVERHEAT OF APU.

(17) Check that "LOW FUEL PRESS" and "IGN ON" indicator lights go out. If "LOW OIL PRESSURE" indicator light comes on and APU does not automatically shut down, place "APU TURBINE" switch at OFF and investigate problem.

(18) Note that start fuel solenoid valve operates (closes) satisfactorily as indicated by cessation of overboard fuel drainage.

NOTE: If the "IGN ON" indicator light remains on after steady state speed is attained, shutdown APU and investigate cause.

(19) Depress "OVER SPEED" indicator light. Ensure that "OVERSPEED" indicator light comes on when depressed and go out when released. This will verify circuit integrity.

(20) If turbine rpm is not within 100 to 103% rpm, shut down APU and refer to Solar Maintenance Instruction Manual SD2-2049 for adjustment procedures.

(21) Perform dc generator and/or bleed air supply system check as required. If no further check is required, shut down APU.

2. APU Shutdown

A. To perform APU shutdown, proceed as follows:

(1) Place "GENERATOR" and "BLEED AIR" switches to OFF. Note that GEN OFF light comes on.

(2) On NA-265-80-1, -2, -3, -4, -6, -7, -8, -9 and -11 airplanes, position EGT trim rotary switch at start.

(3) Place "ELECTRIC MASTER" and "BATTERY" switches at ON.

(4) Place CABIN TEMP switch to MAN COLD, CABIN TEMP rheostat to COLD and CABIN AIR switch to BOTH ENG.

(5) Move "TURBINE START-RUN-OFF" switch to OFF.

(6) Note that GEN OFF light goes out.

(7) During APU shutdown, check that fuel combustor drain valve opens as indicated by fuel draining through the valve overboard drain line.

(8) Observe "coast down" for any unusual noises (scraping or grinding) at low speeds and especially during final rotations. Coast down should not be less than 12 seconds (rated speed to 0 rpm) or should not exceed 35 seconds.

WARNING: IF AN APU FIRE OCCURS DURING OR AFTER SHUTDOWN, FOLLOW PROCEDURES GIVEN FOR FIRE IN APU COMPARTMENT.

(9) Move "ELECTRIC MASTER" and "BATTERY" switches to OFF.

(10) Disconnect external electrical power.

(11) If APU is a new installation, proceed as follows:

(a) Check APU and aft fuselage compartment for fuel oil and exhaust gas leaks.

(b) Replace APU fuel supply line filter. (Refer to Fuel System).

(c) Replace APU oil filter. (Refer to Lubrication System).

(d) Service lubrication system as required.

3. Fire in APU Compartment

A. If fire occurs in APU compartment, proceed as follows:

(1) Ensure that "ELECTRICAL MASTER" and "BATTERY" switches are at ON.

NOTE: Automatic shutdown of the APU is initiated whenever the APU FIRE warning light (red) illuminates. In addition to the fire light indicator, a warbling audio alarm sounds.

(2) Move "APU TURBINE" switch to OFF.

(3) Check that "GENERATOR" and "BLEED AIR" switches are at OFF.

NOTE: If the left hand engine is operating, it should be shutdown as soon as possible.

(4) Lift "FIRE EXT" switch guard and place switch to ON. This will discharge the extinguishing agent into APU compartment.

(5) Move "FIRE EXT" switch to OFF.

(6) If fire persists, open aft fuselage equipment access doors and direct fire extinguishing agent into compartment until fire is extinguished.

4. Turbine Overheat Condition

A. If a turbine overheat condition occurs, as indicated by the illumination of the "TURBINE OVER HEAT" indicator light, proceed as follows:

(1) Ensure that "ELECTRICAL MASTER" and "BATTERY" switches are at ON.

(2) Move "APU TURBINE" switch to OFF.

(3) Check that "GENERATOR" and "BLEED AIR" switches are at OFF and that "EGT TRIM" switch is at START.

(4) If "TURBINE OVER HEAT" indicator light does go out when turbine decelerates to 5%, move "APU TURBINE" switch to RUN then, depress and hold "MOTORIZE" switch for 15 seconds.

(5) If overheat condition persists, direct fire extinguishing agent into the APU exhaust tailpipe duct.

NOTE: To prevent excessive damage to APU, a dry chemical fire extinguishing agent should be used.

5. Postfire Procedures

A. Normally the fire extinguishing agent (monobromotrifluoromethane) evaporates completely without leaving any deposit. However, the agent may puddle in certain areas and cause corrosion. Therefore, to prevent corrosion, proceed as follows:

(1) Wash affected area with a five percent chromic acid solution followed immediately by a clean water rinse.

(2) Throughly inspect all equipment and/or components in the affected area. Clean as required.

(3) Remove discharged container along with the discharge valve.

(4) Purge system with dry compressed air or nitrogen to remove any residual agent.

NOTE: In order to purge the manual discharge line, it is necessary to disconnect the line at the discharge indicator.

(5) Install a serviceable (charged) container along with its associated discharge valves. (Refer to Fire Protection System.)

(6) Replace the fire bottle discharge indicator disks as required. (Refer to Chapter 26.)

(7) Inspect APU, components, wiring, fire detection system, and aft fuselage compartment for fire damage.

6. APU Exhaust Gas Temperature Limitations

When an overtemperature condition occurs, the following table can be used to determine the type of inspection and inspection procedures required before APU operation is continued.

Exhaust Gas Temperature Limitations				
Type of APU Operation	Temperature °F	Duration of Temperature	Number of Occurrences	Inspection Required
Start and Normal Operation	Up to 1100	No Limit	No Limit	None
Start and Normal Operation	1100 to 1170	Momentary	Each Occurrence	*
Start and Normal Operation	Over 1170	Momentary	Each Occurence	**

* Shutdown the APU. Determine and correct the cause of overtemperature.

**Shutdown the APU immediately. Perform the inspection and unit operational checks outlined below under Overtemperature Inspection procedures whenever observed APU exhaust gas temperature exceeds 1170°F. This inspection is also required immediately after the occurrences of a "hung" start or an aborted start during which residual fuel was burned within the APU. If the inspection indicates the APU can be considered satisfactory, continue the APU in service. If any abnormal condition is evident, an APU teardown and inspection is required.

Overtemperature Inspection Procedures

(1) Inspect inlet and exhaust ducts for foreign particles and evidence of rubbing.

(2) Inspect the compressor wheel for free rotation with no audible evidence of circumferential contact and for nicks, dents, or distortion.

(3) Remove and disassemble the combustor. Refer to Installation, Operation, Maintenance, and Overhaul Instructions Manual, Solar SD2-2049.

(4) Inspect the combustor housing, liner, and shield for cracks, buckling, warping, and burn-away.

(5) Inspect the turbine wheel and vanes for evidence of rubbing, burns, nicks, dents, distortion, cracks and damage.

(6) Check the clearance between the turbine wheel vane tips and turbine nozzle. If the clearance is 0.14 inch or less, remove APU for overhaul.

(7) If inspection reveals satisfactory condition of the combustor housing and liner, compressor and turbine wheels, turbine nozzle, etc., reassemble and install the combustor.

(8) Following reinstallation and connection of all lines, fittings, ducts, etc., motor engine. (Refer to Preoperational Check of APU.)

(9) Start the APU. (Refer to APU Starting Procedures.) Allow the APU to accelerate to 50% RPM at which speed the "APU TURBINE" switch should be positioned to OFF shutting down the APU.

(10) If the results of the investigations and functional checks accomplished above indicated normal APU operation, restart the APU. (Refer to APU Starting Procedures.) Allow the APU to accelerate and stabilize. Position the "GENERATOR" switch to ON and "BLEED AIR" switch to ON. On NA-265-80-1, -2, -3, -4, -6, -7, -8, -9 and -11 airplanes if APU operation appears normal, slowly rotate the "EGT TRIM" switch clockwise to increase exhaust gas temperature to 1100°F if not already at this value. Allow the exhaust gas temperature to stabilize.

(11) If all conditons appear normal, the APU can be considered acceptable for service.

(12) Shutdown APU. (Refer to APU Shutdown.)

7. APU Overspeed Limitations

When an overspeed condition occurs, the following table can be used to determine the type of inspection and inspection procedures required before APU operation is continued.

APU Over-speed Limitations

Type of APU Operation	Speed % RPM	Duration of Speed	Number of Occurrences	Inspection Required
Start and Normal Operation	up to 103	No Limit	No Limit	None
Start and Normal Operation	103 to 109	Momentary	Each Occurrence	*
Start and Normal Operation	109 to 115	Momentary	Each Occurrence	**
Start and Normal Operation	Over 115	Momentary	Each Occurrence	***
Start and Normal Operation	Below 100	Momentary	Each Occurrence	*

* Shutdown the APU. Determine and correct the cause of the overspeed or underspeed condition.

** Shutdown the APU immediately. Perform a Visual Inspection whenever observed APU speed exceeds 109% (61798) RPM but does not exceed 115% (65199) RPM. If the inspection indicates the APU can be considered satisfactory, continue

MAINTENANCE PRACTICES (Continued)

APU Overspeed Limitations (Cont'd)

the APU in service. If any abnormal condition is evident, a complete APU teardown and inspection is required.

***When observed APU speed exceeds 115% (65199) RPM, shutdown the APU immediately and return APU to overhaul.

Visual Inspection Procedures

(1) Inspect the inlet and exhaust ducts for foreign particles, evidence of rubbing or damaged compressor or turbine vanes.

(2) Inspect compressor-turbine wheels for free rotation with no audible evidence of circumferential contact and for nicks, dents, etc.

(3) Inspect accessory gear box for evidence of oil leakage, security and external damage.

Inspection of APU

NOTES

- ***The following inspections and procedures are in no way intended as a substitute for the maintenance practices found in the APU Service/Maintenance Manual, T-62 T-39. Contact Sundstrand Power Systems, 4400 Ruffin Rd., P.O. Box 85757, San Diego, CA 92138-5757. Tel (619) 569-4400.***
- ***The following inspections will be performed at the specified intervals.***

Daily/Preflight

Observe maximum EGT during starting, no-load, and steady-state operation. Observe governed no-load and full load speed (when meters are installed). Listen for unusual noise and vibration during start, operation and shutdown.

Weekly

Check oil level at sight gauge for proper quantity. Replenish as necessary.

50 Hours

Check complete APU externally for fuel and oil leakage. Check for obstructions in the air inlet, exhaust, and drains. Visually inspect housing covers and brackets. Visually inspect combustor section for signs of corrosion, cracks, deformation or other conditions that may affect strength or performance. Visually inspect for condition and security of installation.

12 Months

1. Visually inspect APU harness assembly for security and insulation breakdown, rubbing, chafing, and general condition. Check for tightness of electrical connectors and terminal block connections.
2. Visually check the bleed air system for leaks at clamped joints and ducting. Check bleed control valve for cracks, dents, deformation, and other conditions that may affect performance of valve.
3. Drain oil sump and replenish with approximately 3 U.S. quarts (2.8 liters) of lubricating oil to correct oil level.
4. Remove magnetic plug. Check for unusual magnetic debris. Clean and reinstall.
5. Visually check air inlet and compressor wheel for cleanliness and signs of erosion or rubbing (installation permitting). As much of the air path as can be observed should be checked. If severe scratches or effects of erosion are found, return the APU for repair.

NOTE

Some compressor wheels are titanium based. Refer to handling caution in the APU manual.

6. Check attaching bolts, screws and lockwires for condition and security. Replace or verify proper torques as required.
7. Remove, clean, and inspect the combustor fuel drain valve.
8. Check igniter plug for proper operation. Remove and replace as required.
9. Check restrictor orifice for proper operation. Remove, clean, and flow check as required.
10. Visually check fuel tube bundle and fuel lines for chafing, signs of leakage, and general condition.

NOTE

The hour intervals for the following inspections are APU operating hours, not airplane hours, and are taken from the APU hour meter.

1000 Hours or 3 Years, Whichever Occurs First

1. Remove and replace fuel inlet filter.
2. Remove and replace oil filter.

1500 Hours or 5 Years, Whichever Occurs First

Remove fuel pump internal filter. Visually inspect for contamination. Clean or replace filter as required.

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APU ELECTRICAL SYSTEM - DESCRIPTION AND OPERATION

1. General

During APU operation, dc electrical power is supplied to the aircraft dc essential bus by a 30 volt, 200 ampere starter-generator mounted on the APU drive gear box. The starter-generator is also used to start the APU. The dc electrical system consists of a start controller, voltage regulator, reverse current relay, and the necessary relays, circuit breakers, control units, switches and indicators, to control and monitor the system. On NA-265-80-1, -2, -3, -4, -6, -8, -9 and -11 airplanes the control units and relays are installed in two relay boxes in the APU compartment at fuselage station 402 and 405. On NA-265-80-5, -10, -12 and subsequent airplanes the control units and relays are installed in the relay boxes in the aft fuselage and also in the electrical equipment rack at fuselage station 150. See figure (49-00-1).

2. Start Controller

The start controller (contactor) is mounted in the APU relay box located at FS 401. The start controller is used during the starting cycle only. The controller is energized through operation of the turbine control or motorize switch. When the controller is energized, electrical power from the aircraft parallel bus is routed to the series field in the starter-generator, thereby initiating a start cycle. When the APU rpm reaches approximately 90% as sensed by the rpm sensor the start controller is automatically de-energized through relays K1 and K4, thereby disconnecting the starter-generator series field from the aircraft essential bus.

3. Voltage Regulator Control

The solid state voltage regulator control (VR4000) is mounted in the electrical equipment rack at fuselage station 150 RH. (See figure 49-00-1.) The regulator control provides the necessary voltage regulation, current limiting, overvoltage protection and overcurrent protection for the starter-generator shunt field. The generating system is energized by actuation of the generator switch located on the APU control panel. This switch also functions as a circuit breaker. During operation of the APU generator system, if a overvoltage or a generator field overcurrent occurs, the voltage regulator control senses this and trips the generator control switch to off, thereby disconnecting the generator from the aircraft bus. (See figure 49-10-1.)

4. Reverse Current Relay

The reverse-current relay (K4001) is mounted in the APU relay box located at FS 401. When the APU is operating, the reverse-current relay connects the output of the generator to the aircraft essential bus. (See figure 49-10-1.) The reverse-current relay is energized by the voltage regulator control when the generator control switch is on. Generator output power is removed from the essential bus whenever the right engine master switch is turned on, or if external power is applied to the aircraft.

5. Generator and Turbine Overheat Systems

The generator overheat system consists of a overtemp switch located on the generator and a special type circuit breaker.

In the event of a generator overheat condition, the overtemp switch closes, allowing the circuit breaker to trip which in turn removes the ground circuit from relay K3. Relay K3 is then deenergized, thus removing power from the fuel boost pump and fuel shut-off valves resulting in an automatic APU shutdown. (See figure 49-10-0.)

The turbine overheat system consists of two overheat switches located on the turbine shroud, either switch will actuate when the temperature reaches 500°F. When either switch is actuated, a ground circuit is provided for relay K15. Relay K15 is then deenergized, which in turn removes power from relay K3. Relay K3 is then deenergized, thus removing power from the fuel boost pump and fuel shut-off valves resulting in an automatic APU shutdown.

6. Overspeed and Underspeed Systems

The APU overspeed and underspeed system consists of a tachometer generator mounted on the APU and a rpm sensor (CN1) mounted in the relay box at fuselage station 405. The rpm sensor receives electrical signals from the tachometer generator. During an underspeed condition, a set of contacts in the rpm sensor closes, which in turn energizes relay K18. When relay K18 is energized, power is removed from relay K3. Relay K3 is then deenergized, thus removing power from the fuel boost pump and fuel shut-off valves resulting in an automatic APU shutdown.

During an overspeed condition, a second set of contacts in the rpm sensor opens, which in turn removes power from relay K7. Relay K7 is then deenergized, removing power from relay K3. Relay K3 is then deenergized, thus removing power from the fuel boost pump and fuel shut-off valves resulting in an automatic APU shutdown.

7. EGT Systems

The EGT system consists of thermocouples (TC1 and TC2), EGT indicator and sensor and on NA-265-80-1, -2, -3, -4, -6, -7, -8, -9 and -11 airplanes, a EGT trim switch. If the temperature of the APU turbine exceeds 1150°F, as sensed by one set of thermocouples (TC1) a set of contacts in the EGT sensor (CN2) opens, removing power from relay K6. Relay K6 is then deenergized, removing power from relay K3. Relay K3 is then deenergized, thus removing power from the fuel boost pump and fuel shut-off valves resulting in an automatic APU shutdown. The EGT trim switch is used in the bleed air control system.

8. Bleed Air Control System

The bleed air control system consists of a electrical controlled bleed valve, air bleed controller, air bleed valve control switch, and on NA-265-80-1, -2, -3, -4, -6, -7, -8, -9 and -11 airplanes a EGT trim switch. When the air bleed switch is on, the air bleed controller is energized, which in turn

automatically controls the air bleed valve to maintain a constant air bleed supply. The EGT trim switch is used to provide a maximum air bleed supply during operation of the air conditioning system. The switch is used to trim the EGT temperature to a maximum temperature of 1100°F.

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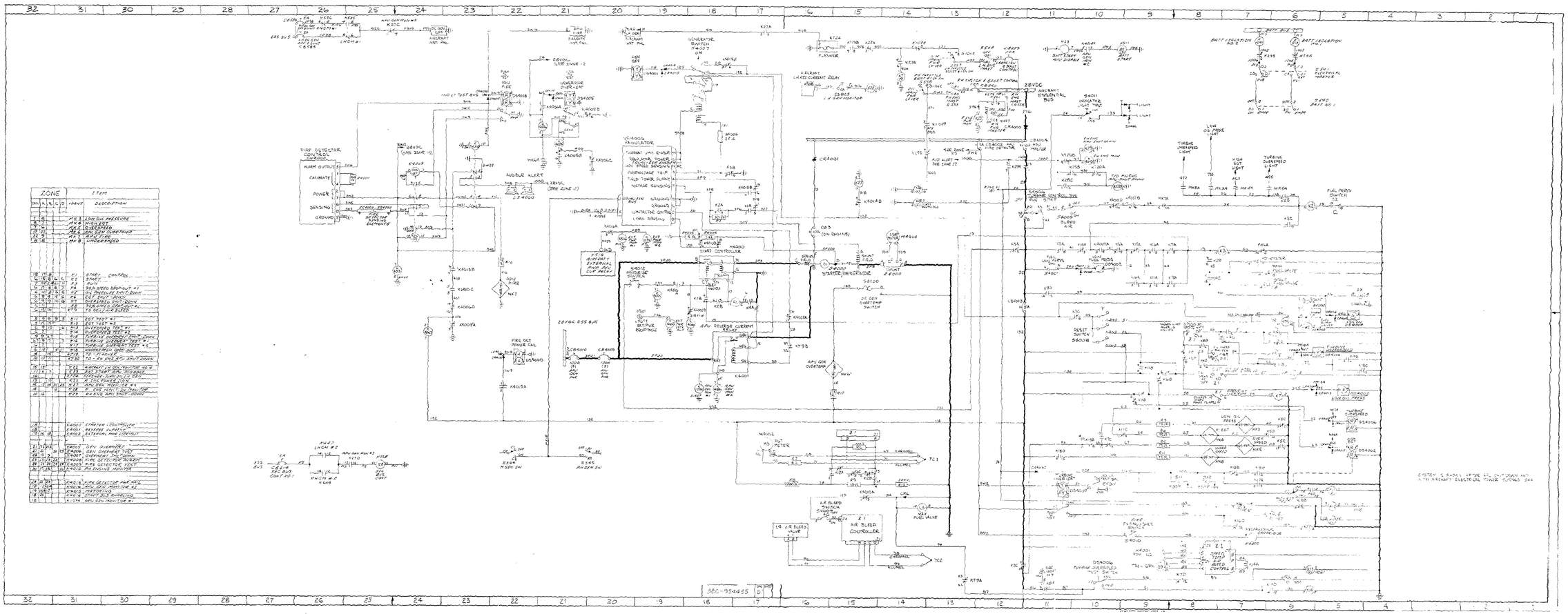

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DC GENERATION AND BLEED AIR - TROUBLE SHOOTING

CAUTION: WHEN TROUBLE SHOOTING AN ELECTRICAL CIRCUIT, ENSURE THAT ELECTRICAL POWER IS TURNED OFF BEFORE REMOVING ANY ELECTRICAL WIRES. DO NOT APPLY POWER TO CIRCUIT UNTIL AFTER IT HAS BEEN DETERMINED THAT NO DAMAGE WILL OCCUR TO AIRPLANE EQUIPMENT OR WIRING. AFTER TROUBLE SHOOTING SYSTEM, ENSURE THAT ANY TEMPORARILY DISCONNECTED CIRCUIT WIRING IS RECONNECTED AT COMPLETION OF PROCEDURE.

1. Trouble Shooting Generator System

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. NO OUTPUT VOLTAGE FROM GENERATOR		
(1) Internal short in generator	Perform voltage and continuity check of generator	Replace Generator
(2) Short or open circuit in airplane wiring	Perform continuity check of system	Repair short or open circuit
B. LOW-VOLTAGE OUTPUT FROM GENERATOR		
(1) Loose internal connection in generator	Perform voltage output check of generator	Replace generator
(2) Loose wiring harness connection	Perform voltage output check of generator	Tighten loose connection
(3) Foreign material on generator commutator	Perform voltage output check of generator, and inspect commutator	Replace generator
C. FLUCTUATING OUTPUT CURRENT OR VOLTAGE FROM GENERATOR		
(1) Loose brush connection	Visually inspect connections	Tighten as necessary
(2) Brushes not seated	Inspect brush wear and seat	Replace generator
(3) Internal generator failure	Perform an operational check-out of system under load	Replace generator

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
D. FALSE TRIPPING		
(1) Faulty voltage regulator control	Perform test of overvoltage system (Refer to Adjusting APU Generator System)	Replace voltage regulator control
E. NO LOADMETER READING		
(1) Loose connection	Inspect connections	Tighten as necessary
(2) Open circuit	Perform power-off continuity check of loadmeter circuit	Repair circuit
(3) Faulty loadmeter	Perform continuity check of loadmeter	Replace loadmeter
F. GENERATOR WILL NOT STAY ON LINE		
(1) Faulty voltage regulator control	Perform test of system (Refer to Adjusting APU Generator System)	Replace voltage regulator control
(2) Faulty generator control switch	Perform continuity check of switch	Replace switch

2. Trouble Shooting Bleed Air System

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. NO BLEED AIR TO HEAT EXCHANGER		
(1) Faulty air bleed vlave (L4)	Perform continuity check of valve	Replace valve
(2) Faulty bleed air switch (S4009)	Perform continuity check of switch	Replace switch
B. PARTICAL BLEED AIR TO HEAT EXCHANGER		
(1) Faulty bleed air controller (CN4)	Perform resistance check of thermocouple system	Replace air bleed controller
(2) Max fuel valve (L3) is faulty	Perform continuity check of valve	Replace valve

APU ELECTRICAL SYSTEM - MAINTENANCE PRACTICES

1. Replacement of Starter-Generator

A. Removing starter-generator

(1) Place all switches at OFF or airplane-parked position and disconnect all external power from airplane.

(2) Open aft fuselage access doors on fuselage lower centerline.

(3) Remove clamp securing air inlet duct to starter-generator shroud.

(4) Loosen clamp securing air inlet duct to heat exchanger inlet duct and rotate inlet duct clear of starter-generator.

(5) Remove clamp securing starter-generator exhaust duct at starter-generator shroud. Loosen clamp at other end of exhaust duct and rotate exhaust duct clear of starter-generator.

(6) Remove clamp securing shroud to starter-generator then remove shroud.

(7) Disconnect and remove electrical wiring from starter-generator.

(8) Remove screw from overheat sensor clamp, then carefully remove clamp and sensor from starter-generator.

CAUTION: SUPPORT STARTER-GENERATOR BEFORE REMOVING LAST FLANGE ATTACHING NUT.

(9) Remove generator flange attaching nuts (6).

(10) Remove starter-generator from gear box.

B. Installing Starter-Generator

(1) Disconnect all external power from airplane and place all switches at OFF or airplane-parked position.

(2) Lubricate starter-generator drive spline with grease (Spec. MIL-G-81322A or equivalent).

(3) Install generator or APU drive gear box.

(4) While supporting weight of generator install the six remaining nuts. Tighten and torque nuts to 160-185 inch-pounds.

(5) Place sensor clamp between terminal block and data plate, then rotate clamp until sensor is directly over the center line of the top field pole retaining screw. Install clamp screw and tighten.

(6) Connect electrical wiring to generator terminals.

(7) Position starter-generator shroud in place and secure with clamp.

(8) Rotate inlet and exhaust ducting back into position and secure in place with clamps. Tighten all clamps.

(9) Perform operational check-out of starter-generator.

2. Replacement of Control Units or Relays

The removal and installation procedures for the control units or relays located within the APU relay boxes, or on the electrical equipment rack shelf are essentially the same.

A. Removing Control Unit or Relay

(1) Place all switches at OFF or airplane-parked position.

(2) Disconnect all external power from airplane.

(3) Gain access to control unit or relay to be removed by removing covers from APU relay boxes in aft fuselage or by sliding electrical equipment rack shelf inboard, as applicable.

(4) Locate relay or control unit (by "K" or "CN" number) on decal adjacent to relay.

(5) Disconnect electrical wiring from control unit or relay.

(6) Remove mounting screws that secure relay or control unit to relay box.

(7) Remove control unit or relay from relay box.

B. Installing Control Unit(s) or Relay(s)

(1) Position control unit or relay in relay box or on electrical equipment rack shelf, as applicable, and secure with mounting screws.

(2) Connect wiring to relay or control unit terminals. (For wire identification information, refer to Wiring Diagram Figure 49-10-3.)

(3) Install relay box cover(s).

(4) Make an operational check-out of the affected system.

MAINTENANCE PRACTICES (Continued)

ADJUSTING APU GENERATOR SYSTEM

Equipment/Tools

- 28 VDC External Electrical Load Bank capable of 50-400 amp load with 10 amp adjustment and a 0-10 amp variable
- 500-Ohm, 25 Watt Wire-Wound Rheostat
- Precision DC Ammeter
- Voltmeter

1. Place all switches at OFF or airplane-parked position.
2. Disconnect all external power from airplane.
3. Connect test equipment to APU generator system as follows:
   A. Gain access to APU compartment and remove relay box (KB4000) cover.
   B. Remove three wires from BAT terminal of reverse current relay (K4001), then insulate and stow wires.
   C. Ground load bank frame to earth ground.
   D. Connect load bank to airplane ground and BAT terminal on reverse current relay (K4001). Adjust load bank to 100 amp load.
   E. Remove wire 3P5P16 from GEN terminal of reverse current relay.
   F. Using 16 gauge (or larger) wire, connect CCW terminal of rheostat to wire 3P5P16 and insulate.
   G. Using 16 gauge (or larger) wire, connect rotor terminal of rheostat to GEN terminal of reverse current relay.
   H. Adjust rheostat full CCW (minimum resistance).
   I. Connect positive lead of ammeter between wire 3D9A22 and loadmeter-shunt. Connect negative lead of ammeter between wire 3D10D22 and loadmeter-shunt (R4000).
   J. Gain access to voltage regulator (VR4000) and connect DC voltmeter to test jacks of voltage regulator. Ensure that polarity is correct.
4. Start APU (refer to 49-00-03).
5. Place GENERATOR switch at ON. Check that GEN OFF indicator light goes out.
6. To check current limiting system of voltage regulator, proced as follows:

CAUTION

To prevent possible generator overheat condition, perform the following test in the shortest possible time.

   A. While monitoring precision ammeter, increase load at load banks until the current limiting system takes control. This is noted by a drop in generator voltage and a constant current flow.
   B. See Figure 49-10-2 for acceptable current limits.

NOTE

The generator may be considered temperature stabilized if the amperage does not decrease more than 5 amps in a 5-minute period.

   C. Decrease load at load bank to indicate 100 amp load on precision ammeter.
7. Check APU loadmeter for 0.5 (±0.4 scale divisions) load.
8. Check precision voltmeter for 28.5 ±0.5 volts. Adjust voltage regulator if required.
9. Adjust load bank to indicate 150 amps on precision ammeter.
10. Check APU loadmeter for 0.75 (±0.4 scale divisions) load.
11. Decrease load at load bank to indicate 100 amp load on precision ammeter.
12. To check overvoltage trip setting, proceed as follows:
   A. While monitoring precision voltmeter, slowly rotate rheostat clockwise. The voltage regulator should trip when 31 to 33 volts is noted on precision voltmeter.
   B. Check that GENERATOR switch has tripped to OFF and that GEN OFF indicator light is on.
13. Shut down APU (refer to 49-00-03).
14. Disconnect all test equipment except precision voltmeter. Connect airplane electrical wiring to the reverse current relay and loadmeter-shunt.
15. Start APU (refer to 49-00-03) then, remove external electrical power from airplane.

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MAINTENANCE PRACTICES (Continued)

ADJUSTING APU GENERATOR SYSTEM (Cont'd)

16. Place GENERATOR switch at ON. Operate generator for 10 to 15 minutes to obtain temperature stabilization.
17. Check loadmeter reading. The indicated reading should not exceed limits of Figure 49-10-2.

NOTE

Voltage difference is dependent upon the battery charge.

18. Check precision voltmeter for approximately 24-29 volts.
19. Place BATTERY switch at OFF. Check precision voltmeter for approximately 28-29 volts.
20. If precision voltmeter readings in step 18 and/or 19 are not as specified, proceed as follows:
    A. With BATTERY switch at OFF, adjust voltage regulator as required to obtain 27.7 volts on precision voltmeter.
    B. Repeat steps 18 and 19.
21. Remove precision voltmeter from voltage regualtor.
22. Perform operational checkout of APU generator and bleed-air system.

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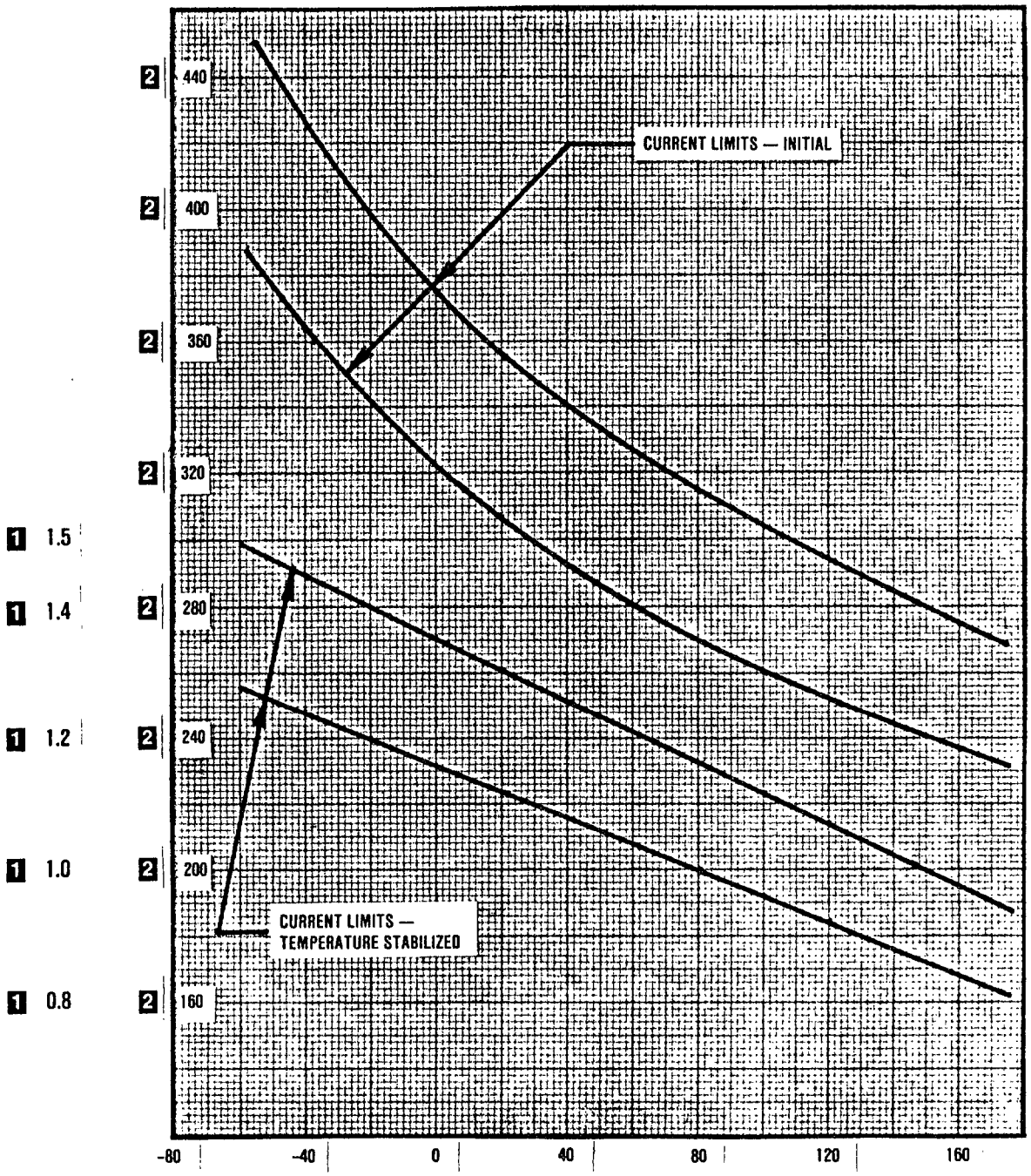

Figure 49-10-2 — Loadmeter Reading Versus Temperature Variation

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1. Operational Check-out of APU Generator and Bleed Air System

(1) Start APU (refer to 49-00-3).

(2) Place "GENERATOR" switch at ON.

(3) Check that "GEN OFF" indicator light goes out. Note loadmeter reading.

(4) Position "CABIN AIR" selector switch to BOTH ENG.

(5) CABIN TEMP selector - MAN COLD. Hold the CABIN TEMP selector switch in the MAN COLD position approximately 20 seconds to ensure the hot air mixing valve is fully closed.

(6) Position "INVERTER" switch to No. 1.

(7) Place "BLEED AIR" switch at ON. Note that APU rpm decreases and EGT increases.

(8) On NA-265-80-1, -2, -3, -4, -6, -7, -8, -9 and -11 airplanes in high ambient temperatures, rotate "EGT TRIM" switch to INCR, until 1100°F is obtained.

NOTE: During high ambient temperatures, the "EGT TRIM" switch is used to increase the APU exhaust gas temperature until 1100°F is obtained. This provides maximum bleed air output from the APU.

(9) Place "GENERATOR" switch to "OFF". Check that "GEN OFF" indicator light comes on.

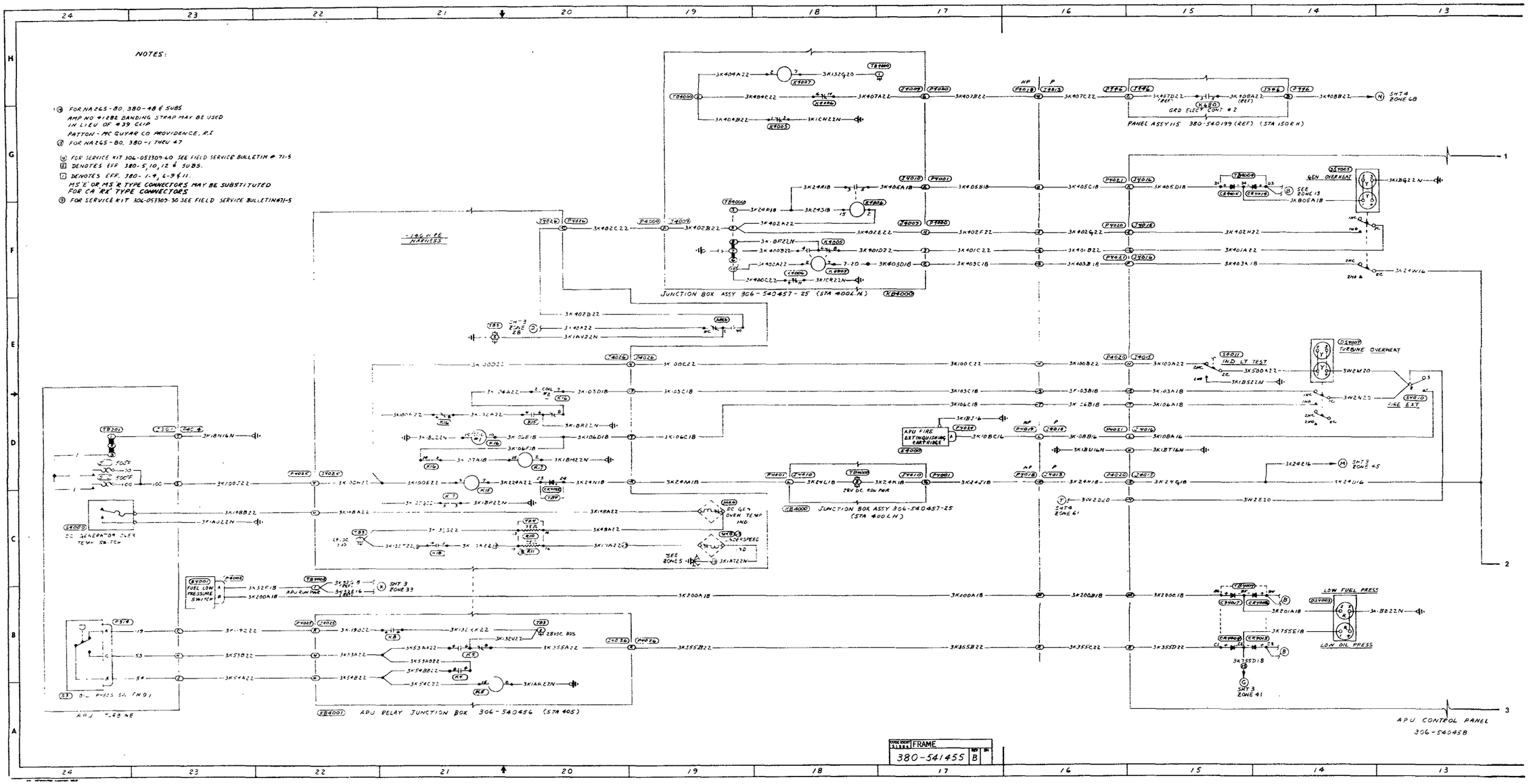
(10) After rpm has stabilized, place "GENERATOR" switch to ON. Check that "GEN OFF" indicator light goes out. Ensure that EGT does not exceed 1100°F, and RPM does not exceed 103%.

(11) Position CABIN TEMP selector to AUTO.

(12) Position cockpit and/or cabin air AUTO CONT as desired.

(13) Position "GENERATOR" and "INVERTER" switches to OFF.

(14) Shutdown APU (refer to 49-00-3).


NOTES:
1 (3) FOR NA265-80, 380-48 & SUBS
AMP NO 91281 BANDING STRAP MAY BE USED IN LIEU OF #39 CLIP
PATTON-MC GUYAR CO. PROVIDENCE, R.I.
(2) FOR NA265-80, 380-1 THRU 47
(6) FOR SERVICE KIT 306-053309-40 SEE FIELD SERVICE BULLETIN # 71-5
(2) DENOTES EFF. 380-5, 10, 12 & SUBS.
(7) DENOTES EFF. 380-1-9, 6-9 & 11.
MS 'E' OR MS 'K' TYPE CONNECTORS MAY BE SUBSTITUTED FOR CA 'RK' TYPE CONNECTORS
(9) FOR SERVICE KIT 306-053309-30 SEE FIELD SERVICE BULLETIN 71-5
-14G #12 E.E. HARNESS
JUNCTION BOX ASSY 306-540457-25 (STA 400L.H.)
PANEL ASSY 115 380-540199 (REF) (STA 150R.H.)
GRD ELECT CONT #2
GEN OVERHEAT
TURBINE OVERHEAT
IND LT TEST
FIRE EXT
APU FIRE EXTINGUISHING CARTRIDGE
28V DC. FOR PWR
LOW FUEL PRESS
LOW OIL PRESS
DC GEN OVER TEMP IND
FUEL LOW PRESSURE SWITCH
APU RUN PWR
28VDC BUS
DC GENERATOR OVER TEMP SW-TCH
OIL PRESS SW. (NO)
APU RELAY JUNCTION BOX 306-540456 (STA 405)
APU CONTROL PANEL 306-540458
SEE ZONE 5
SEE ZONE 15
SHT 3 ZONE 33
SHT 3 ZONE 41
SHT 4 ZONE 68
SHT 3 ZONE 45
SHT 6 ZONE 61
ADJ. -14G #12 E.E.
FRAME 380-541455 REV B

SEE BLOW-UP FICHE NO. ASE0411-ITEM G

APU Wiring Diagrams (306-540456-91, CN5)
Figure 49-10-3 (Sheet 1 of 6)

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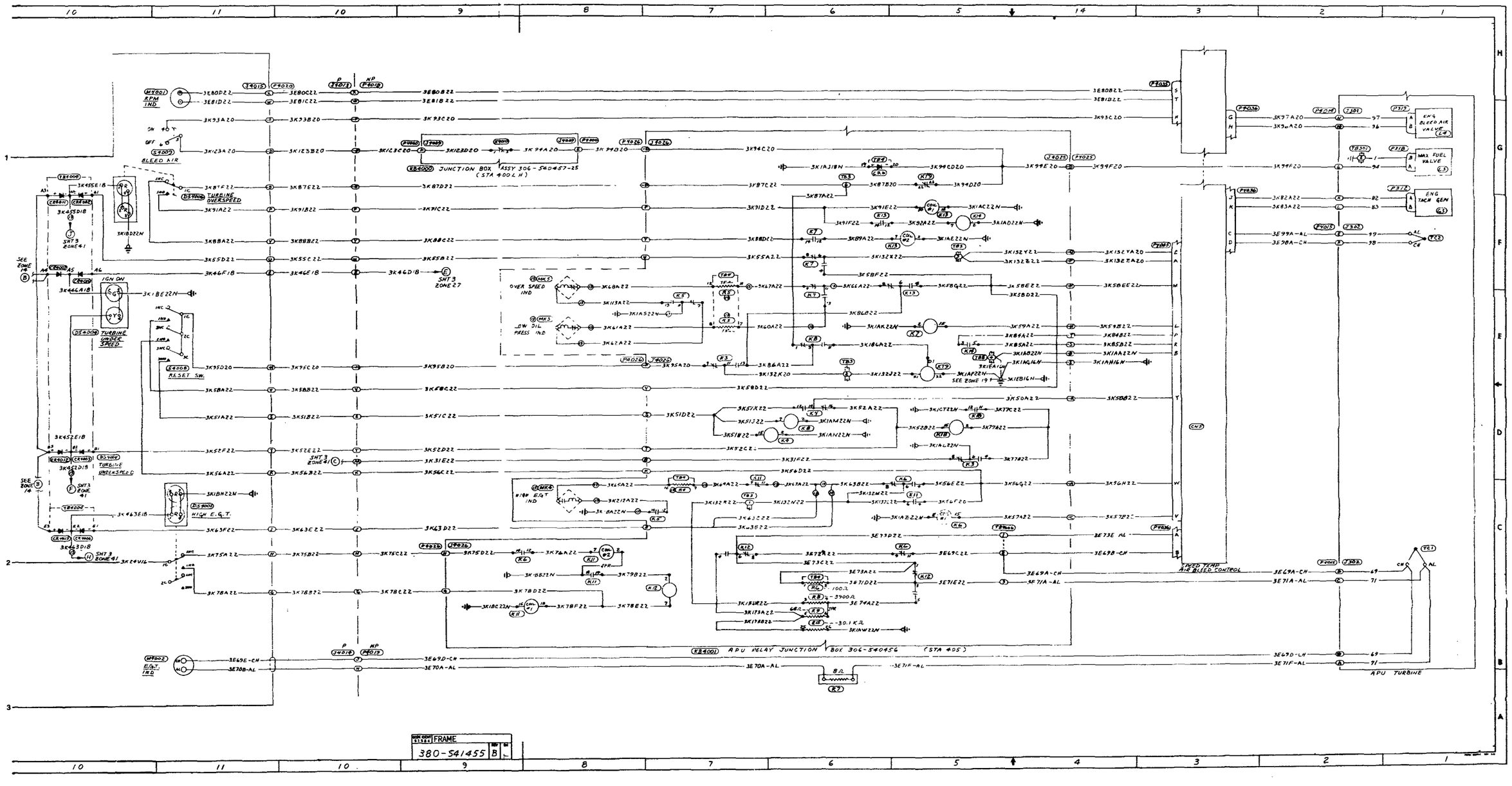

SEE BLOW-UP FICHE NO. ASE0411-ITEM H

APU Wiring Diagrams (306-540456-91, CN5)
Figure 49-10-3 (Sheet 2 of 6)

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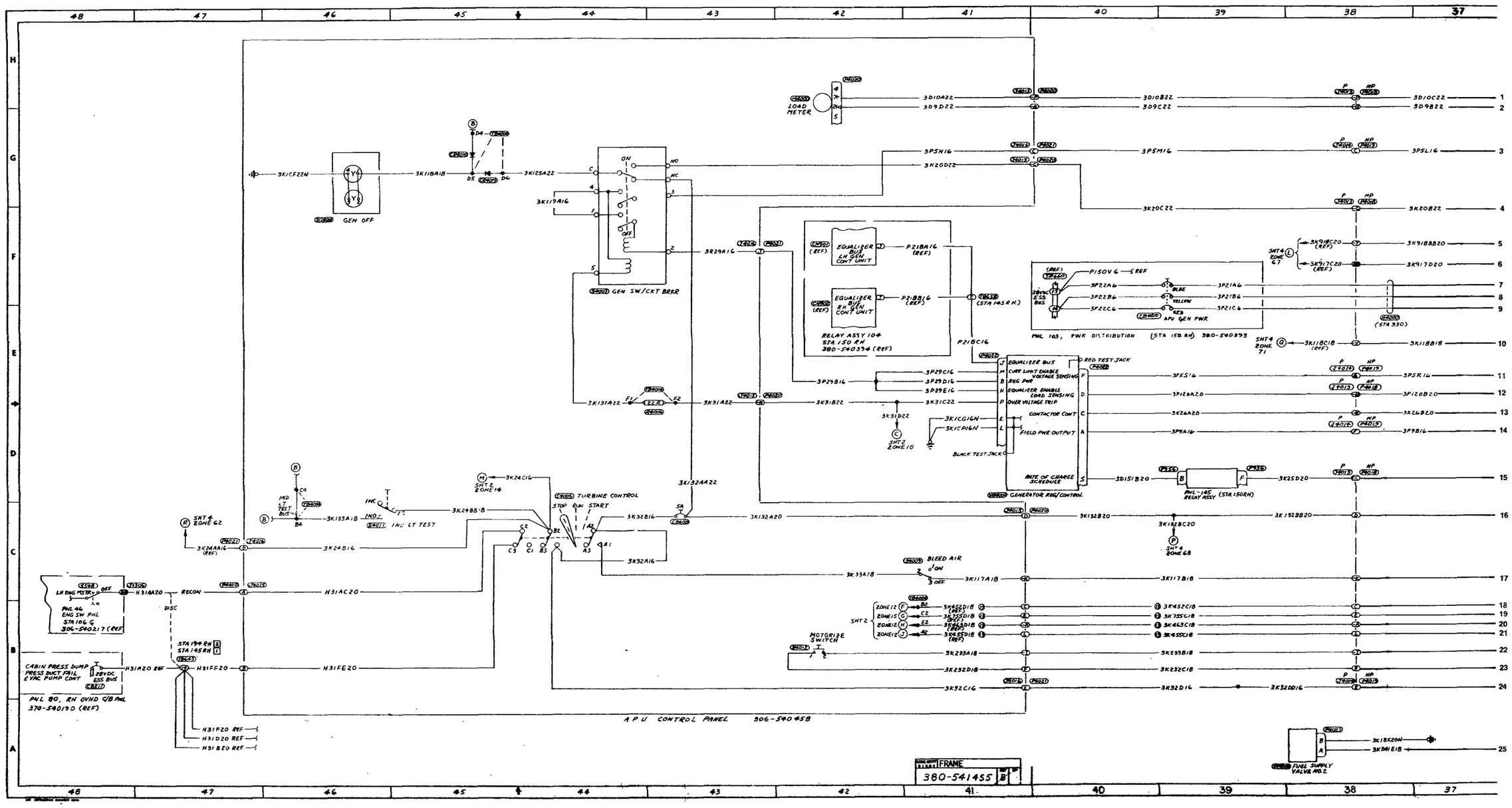
GEN OFF
GEN SW/CKT BRKR
EQUALIZER BUS LH GEN CONT UNIT
EQUALIZER BUS RH GEN CONT UNIT
RELAY ASSY 104 STA 150 RH 380-540394 (REF)
PNL 103, PWR DISTRIBUTION (STA 158 RH) 380-540895
APU GEN PWR
EQUALIZER BUS
EQUAL LIMIT ENABLE VOLTAGE SENSING
REG PWR
EQUALIZER ENABLE LOAD SENSING
POWER VOLTAGE TRIP
CONTACTOR CONT
FIELD PWR OUTPUT
RED TEST JACK
BLACK TEST JACK
RATE OF CHARGE SCHEDULE
GENERATOR REG/CONTROL
PNL-145 RELAY ASSY (STA 150RH)
TURBINE CONTROL
STOP RUN START
IND LT TEST BUS
IND LT TEST
BLEED AIR
MOTORIZE SWITCH
LH ENG MSTR SW
PNL 46 ENG SW PNL STA 196 G 306-540217 (REF)
CABIN PRESS DUMP PRESS DUCT FAIL & VAC PUMP CONT
PNL 80, RH OVHD C/B PNL 370-540130 (REF)
APU CONTROL PANEL 306-540458
FUEL SUPPLY VALVE AG 2
FRAME 380-541455 B
48 47 46 45 44 43 42 41 40 39 38 37
H G F E D C B A

SEE BLOW-UP FICHE NO. ASE0411-ITEM K

APU Wiring Diagrams (306-540456-91, CN5)
Figure 49-10-3 (Sheet 3 of 6)

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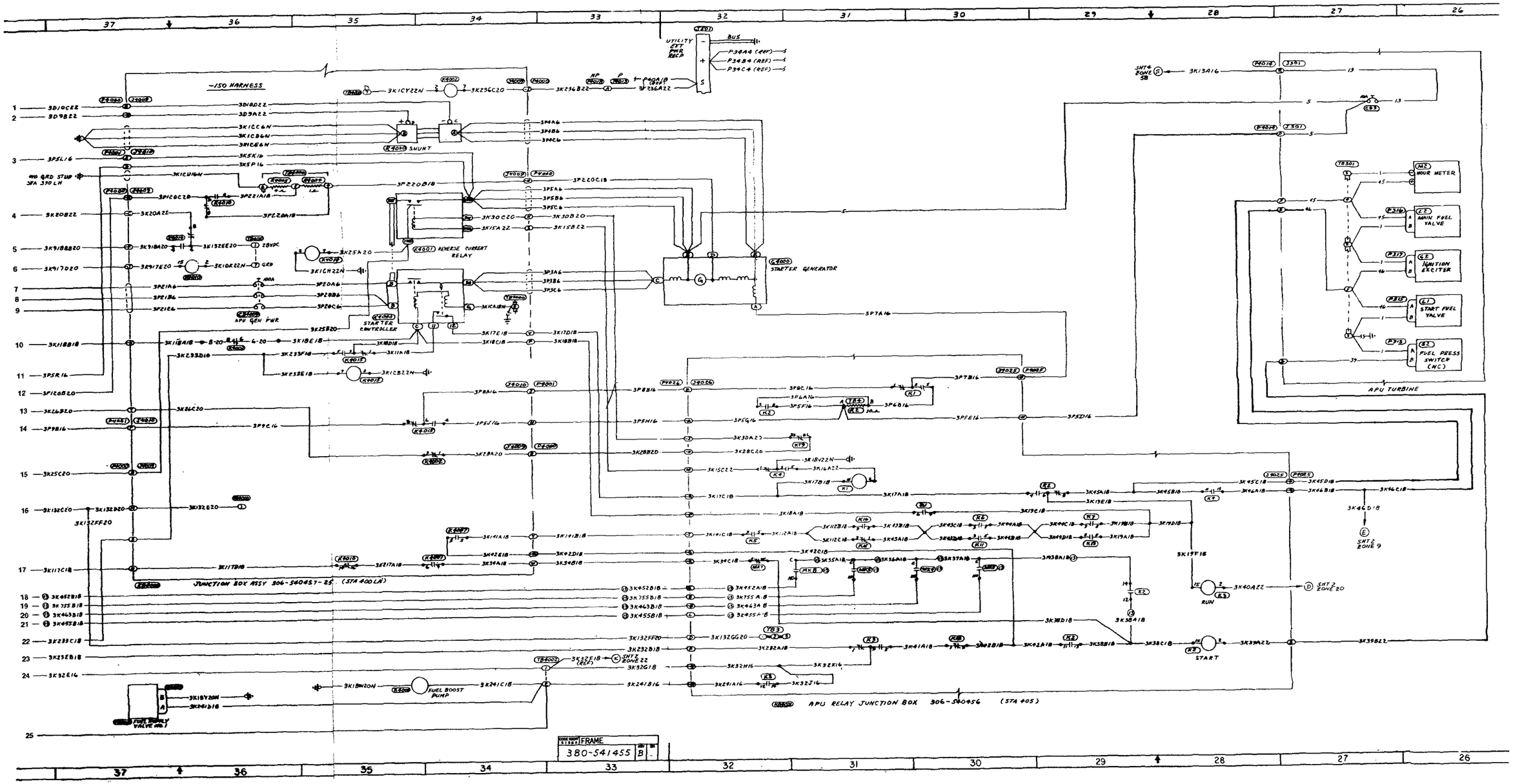

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APU Wiring Diagrams (306-540456-91, CN5)
Figure 49-10-3 (Sheet 4 of 6)

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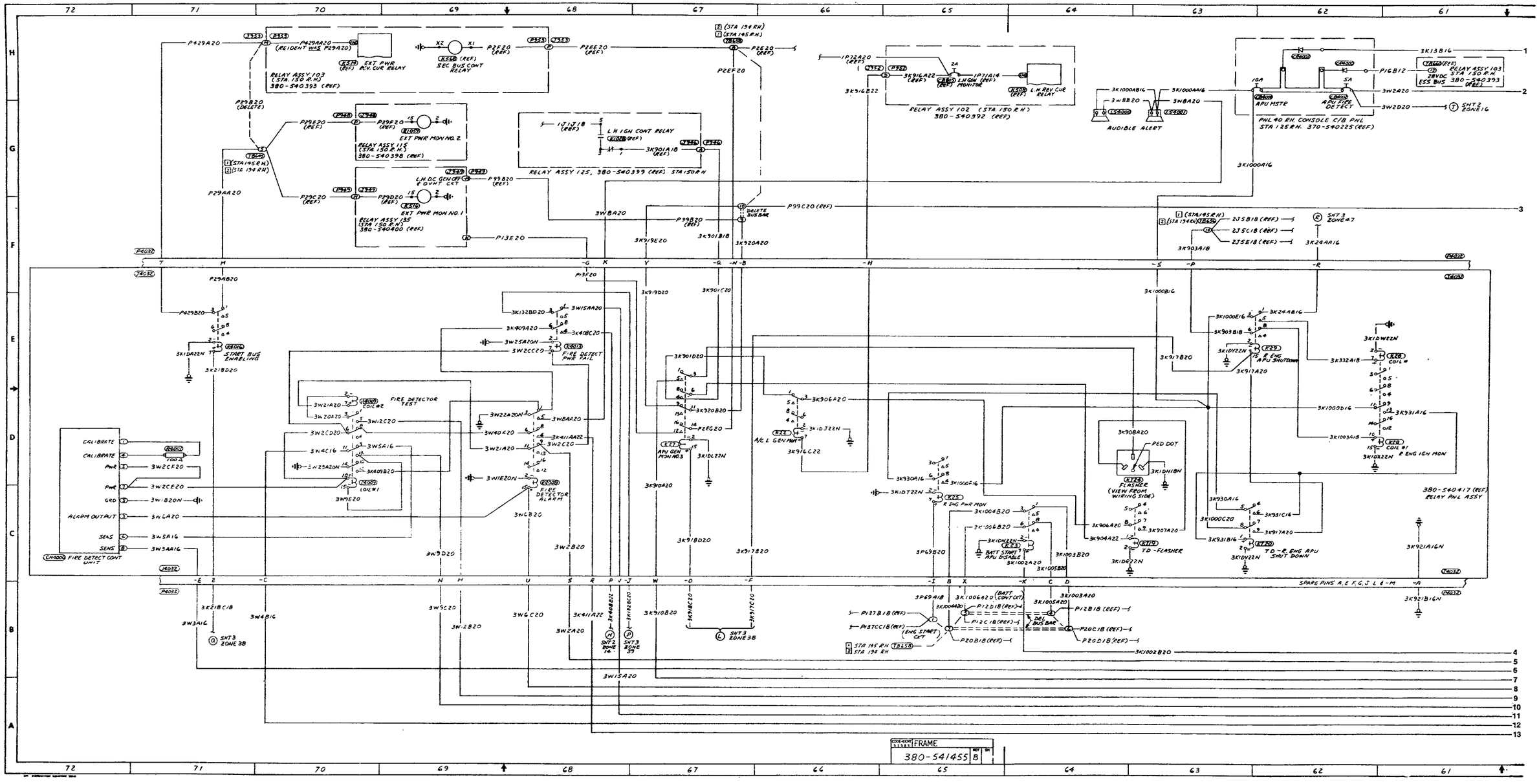
RELAY ASSY 103 (STA 150 R.H.) 380-540393 (REF)
RELAY ASSY 102 (STA 150 R.H.) 380-540392 (REF)
RELAY ASSY 115 (STA 150 R.H.) 380-540398 (REF)
RELAY ASSY 125, 380-540399 (REF) STA 150 R.H
RELAY ASSY 135 (STA 150 R.H.) 380-540400 (REF)
PNL 40 RH. CONSOLE C/B PNL STA 125 R.H. 370-540225 (REF)
380-540417 (REF) RELAY PNL ASSY
AUDIBLE ALERT
FIRE DETECTOR TEST
FIRE DETECTOR ALARM
FIRE DETECT CONT UNIT
APU CONT MON NO 3
APU MSTR
APU FIRE DETECT
FLASHER (VIEW FROM WIRING SIDE)
TD-FLASHER
TD-R ENG APU SHUT DOWN
R ENG IGN MON
BATT START & APU DISABLE
-START BUS ENABLING
SPARE PINS A, E, F, G, J, L & M
FRAME 380-541455 REV B

SEE BLOW-UP FICHE NO. ASE0411-ITEM M

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APU Wiring Diagrams (306-540456-91, CN5)
Figure 49-10-3 (Sheet 5 of 6)

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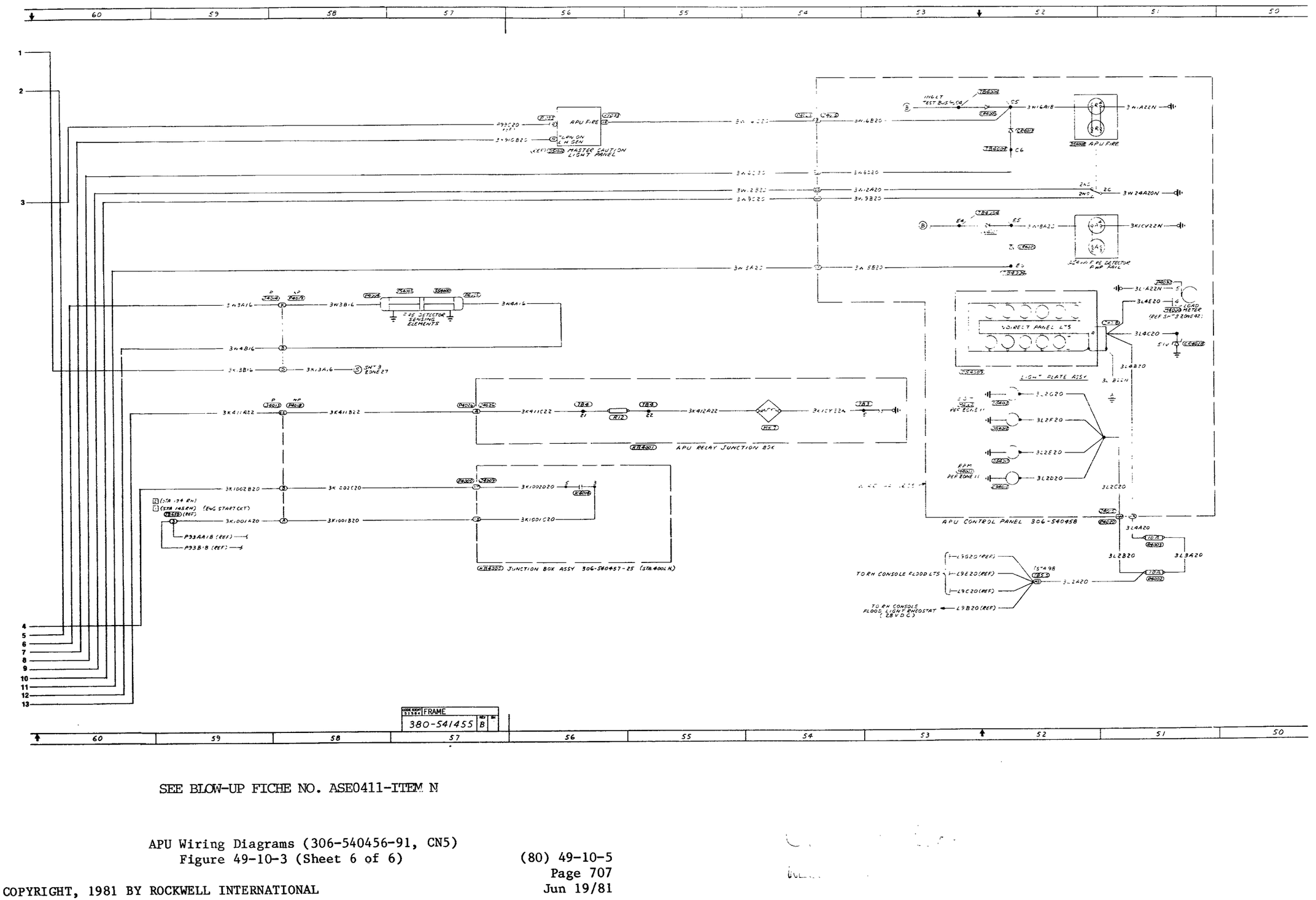

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APU Wiring Diagrams (306-540456-91, CN5)
Figure 49-10-3 (Sheet 6 of 6)

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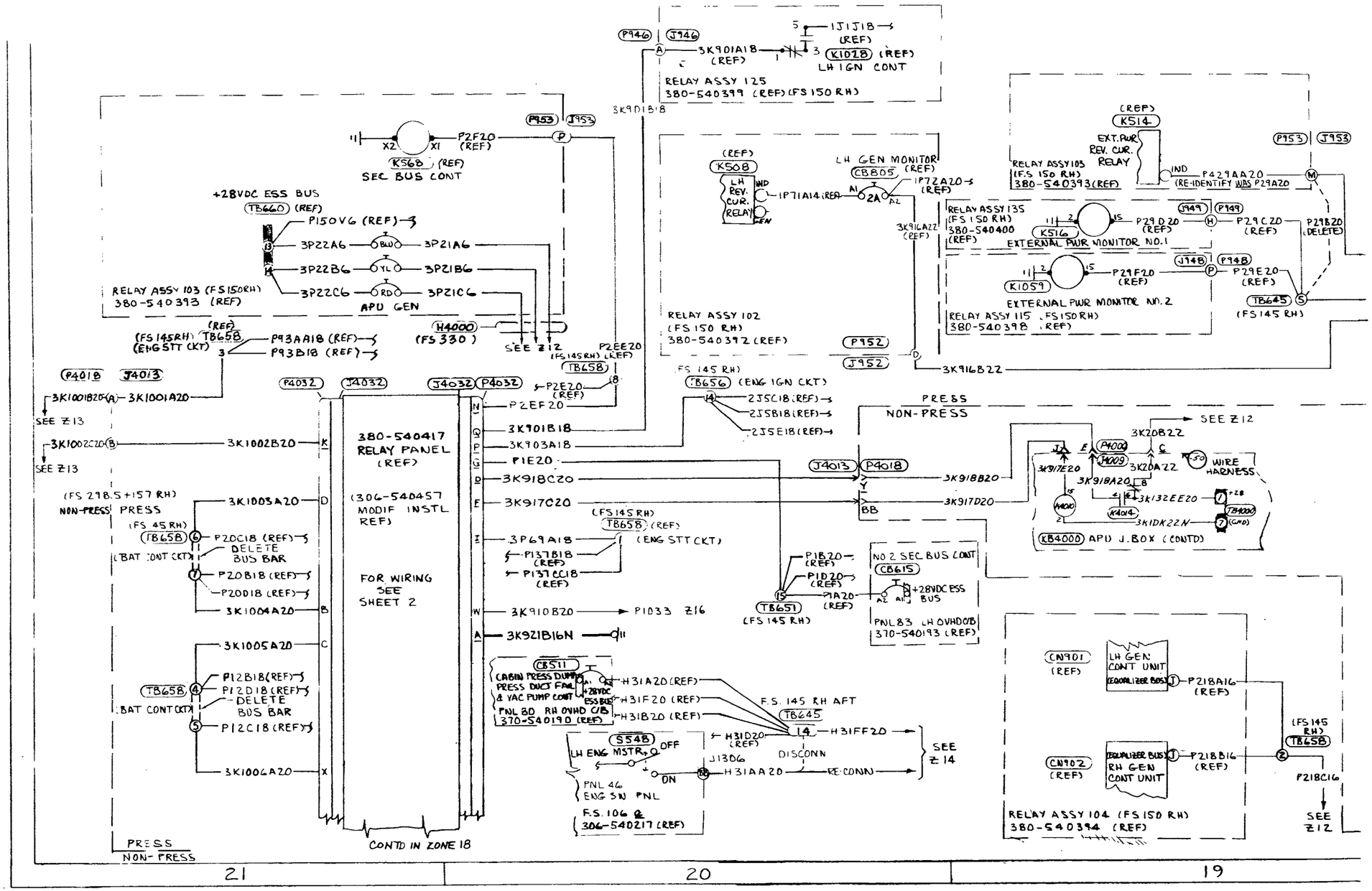

SEE BLOW-UP FICHE NO. ASE0411-ITEM Q

APU Wiring Diagrams (306-540456-11, CN1, CN2, CN4)
Figure 49-10-4 (Sheet 1 of 10)

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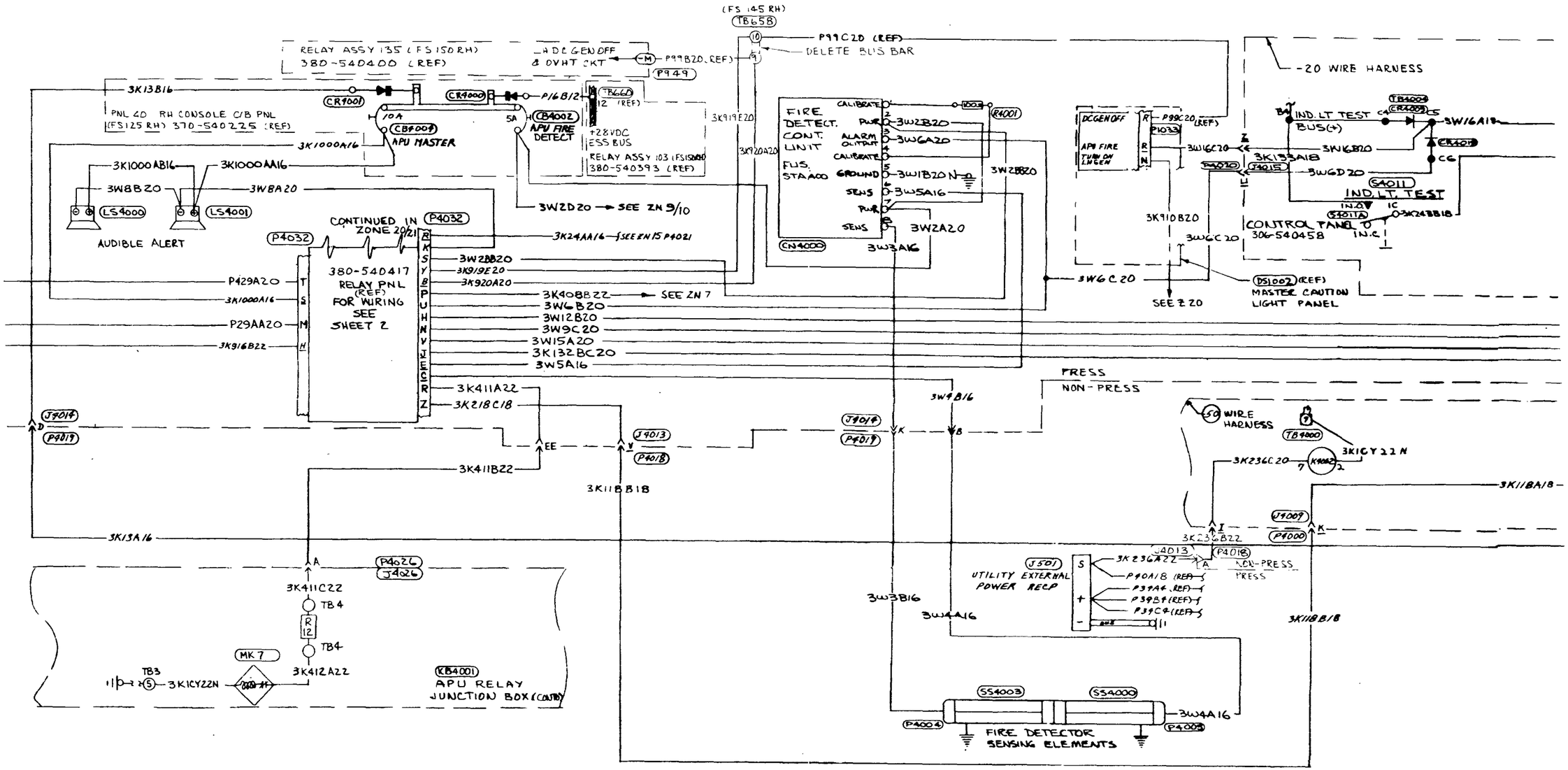

APU Wiring Diagrams (306-540456-11, CN1, CN2, CN4)
Figure 49-10-4 (Sheet 2 of 10)

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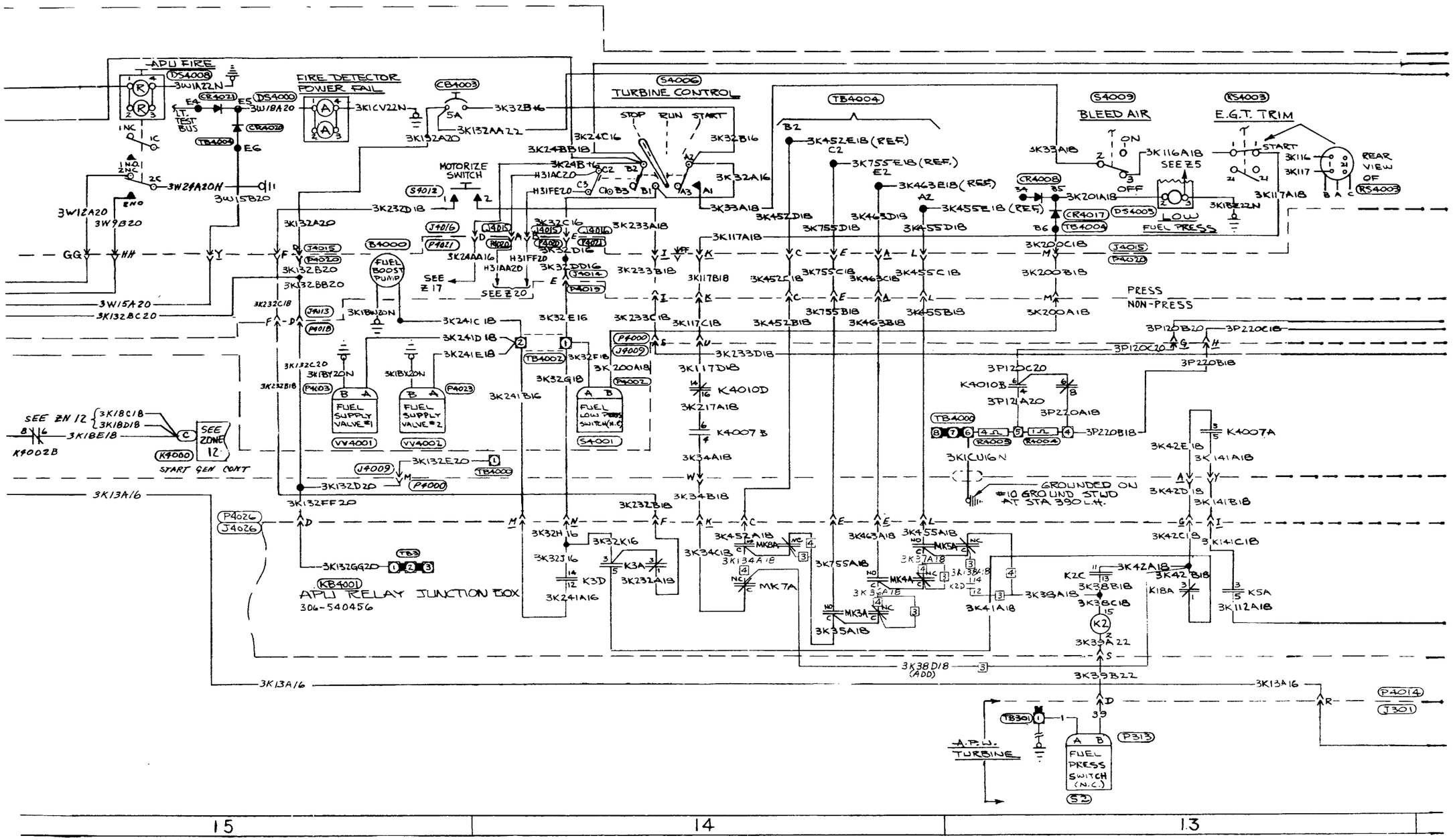

SEE BLOW-UP FICHE NO. ASE0411-ITEM U

APU Wiring Diagrams (306-540456-11, CN1, CN2, CN4)
Figure 49-10-4 (Sheet 3 of 10)

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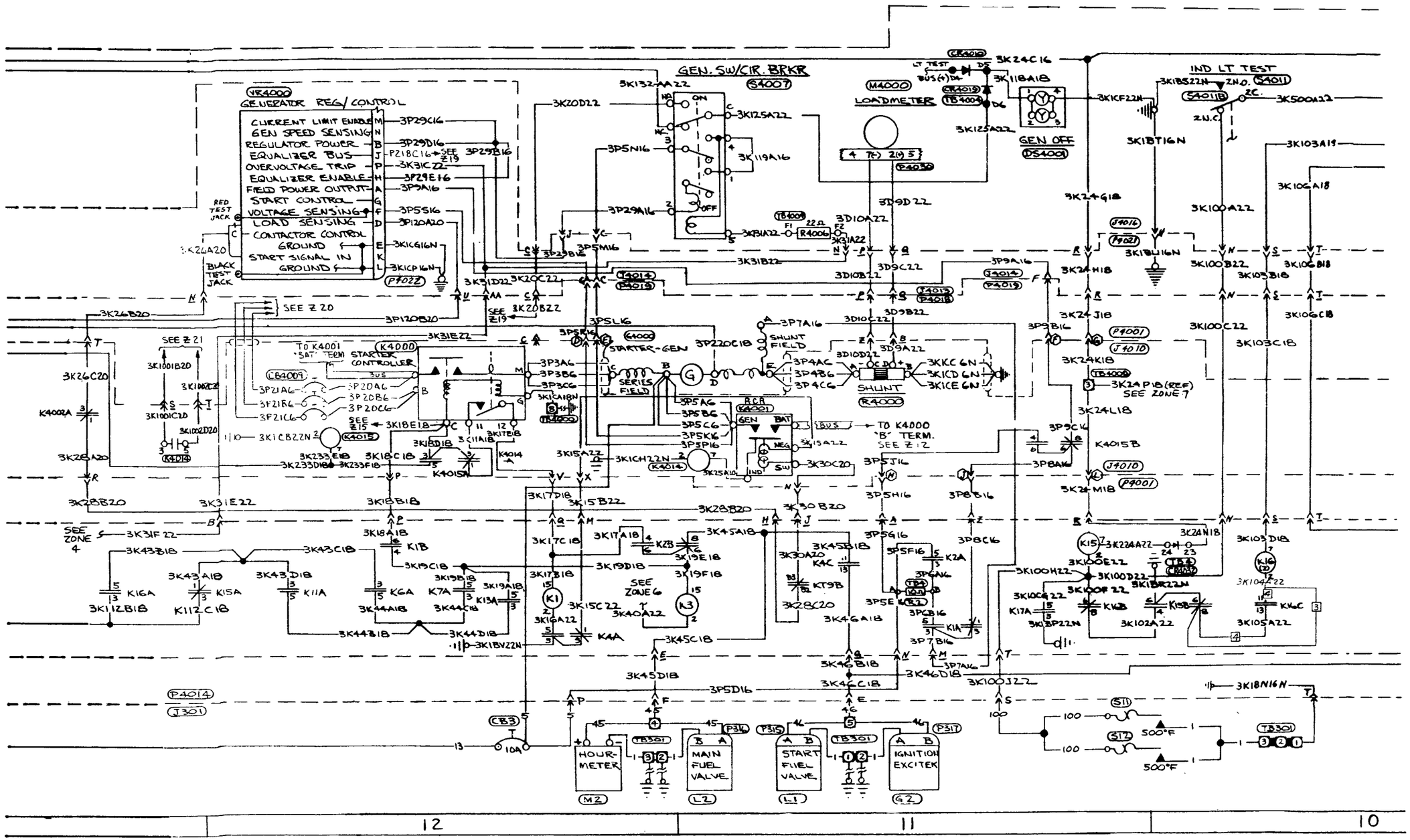

APU Wiring Diagrams (306-540456-11, CN1, CN2, CN4)
Figure 49-10-4 (Sheet 4 of 10)

SEE BLOW-UP FICHE NO. ASE0411-ITEM V

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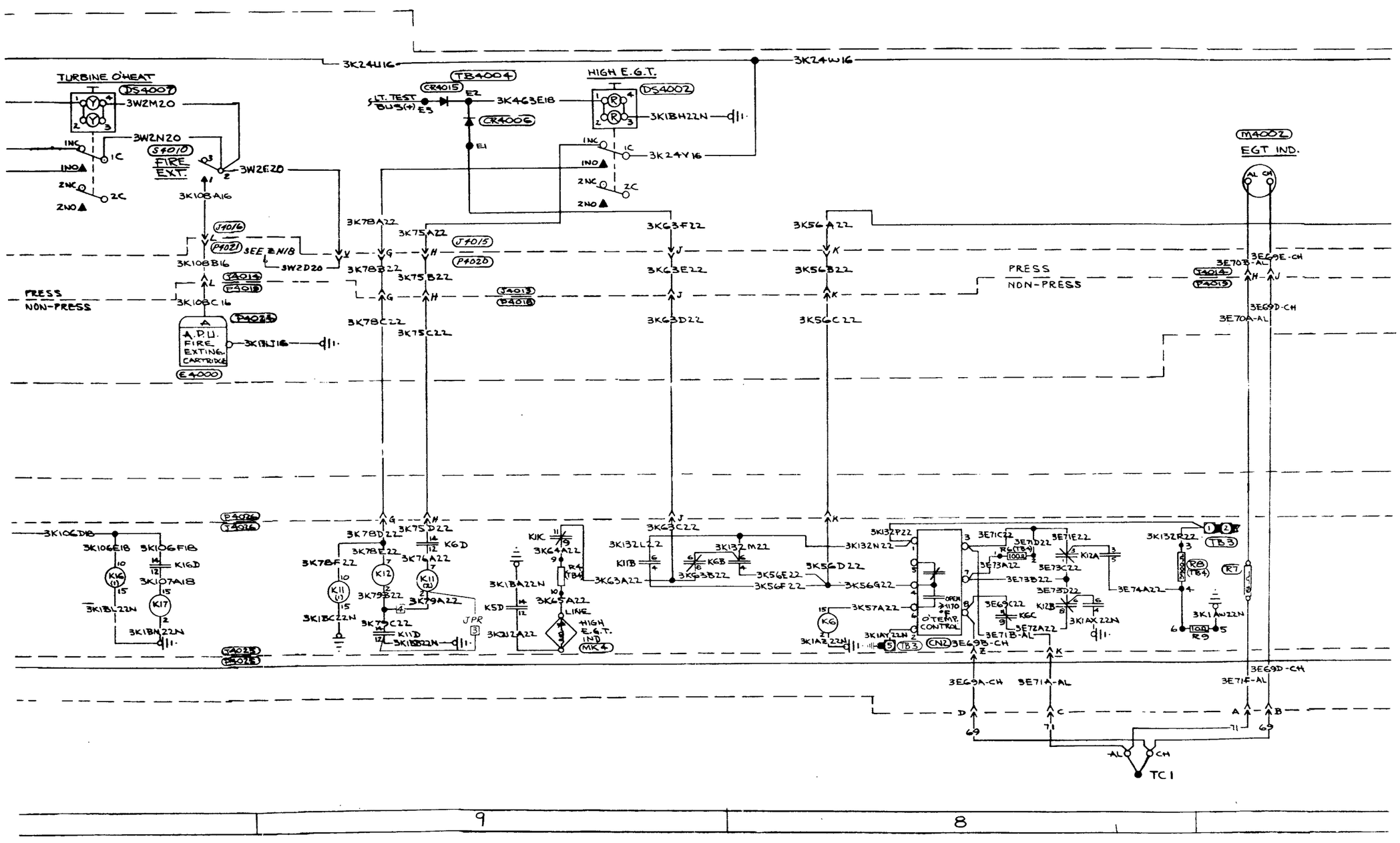

SEE BLOW-UP FICHE NO. ASE0411-ITEM O

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APU Wiring Diagrams (306-540456-11, CN1, CN2, CN4)
Figure 49-10-4 (Sheet 5 of 10)

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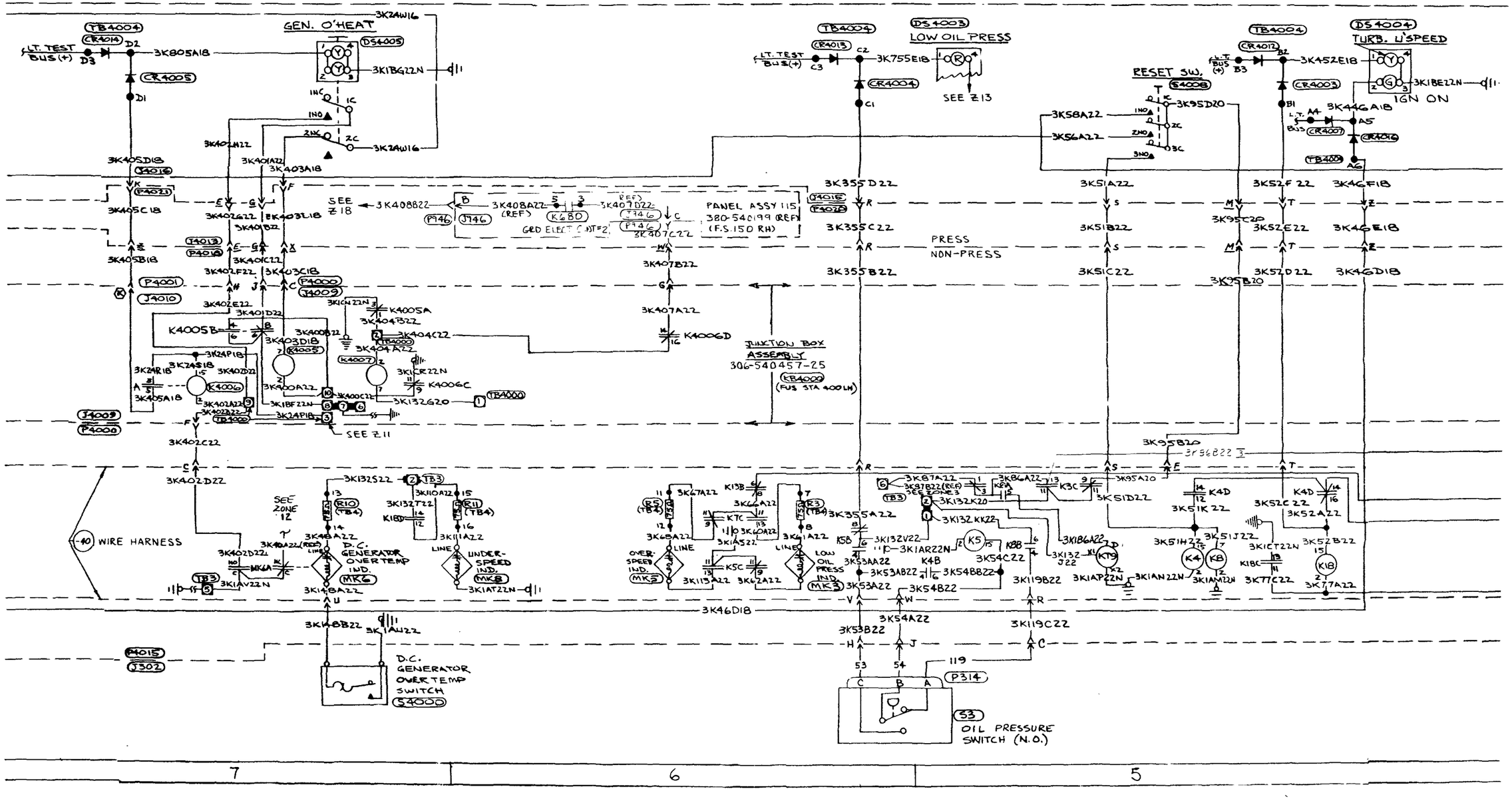

APU Wiring Diagrams (306-540456-11, CN1, CN2, CN4)
Figure 49-10-4 (Sheet 6 of 10)

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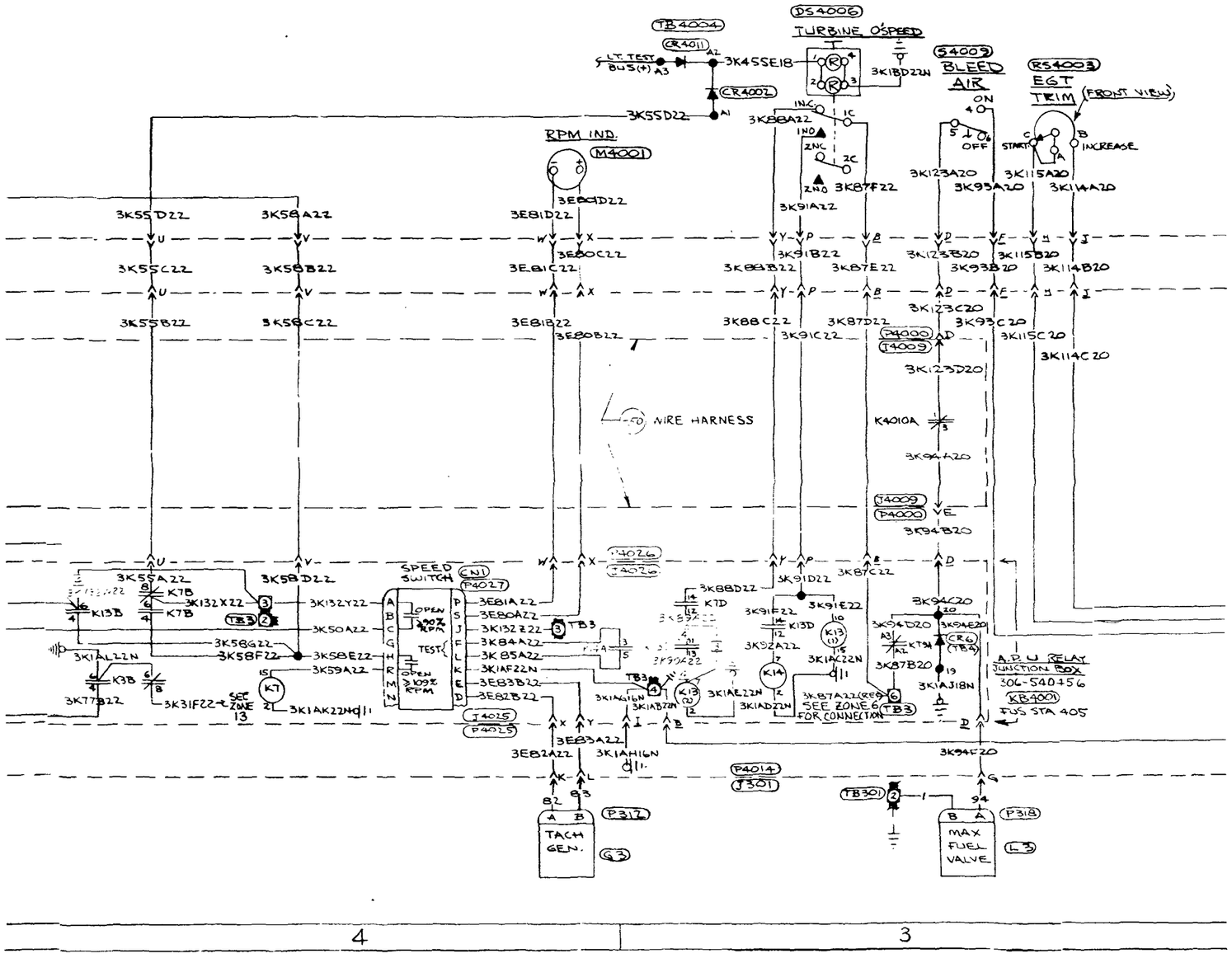

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APU Wiring Diagrams (306-540456-11, CN1, CN2, CN4)
Figure 49-10-4 (Sheet 7 of 10)

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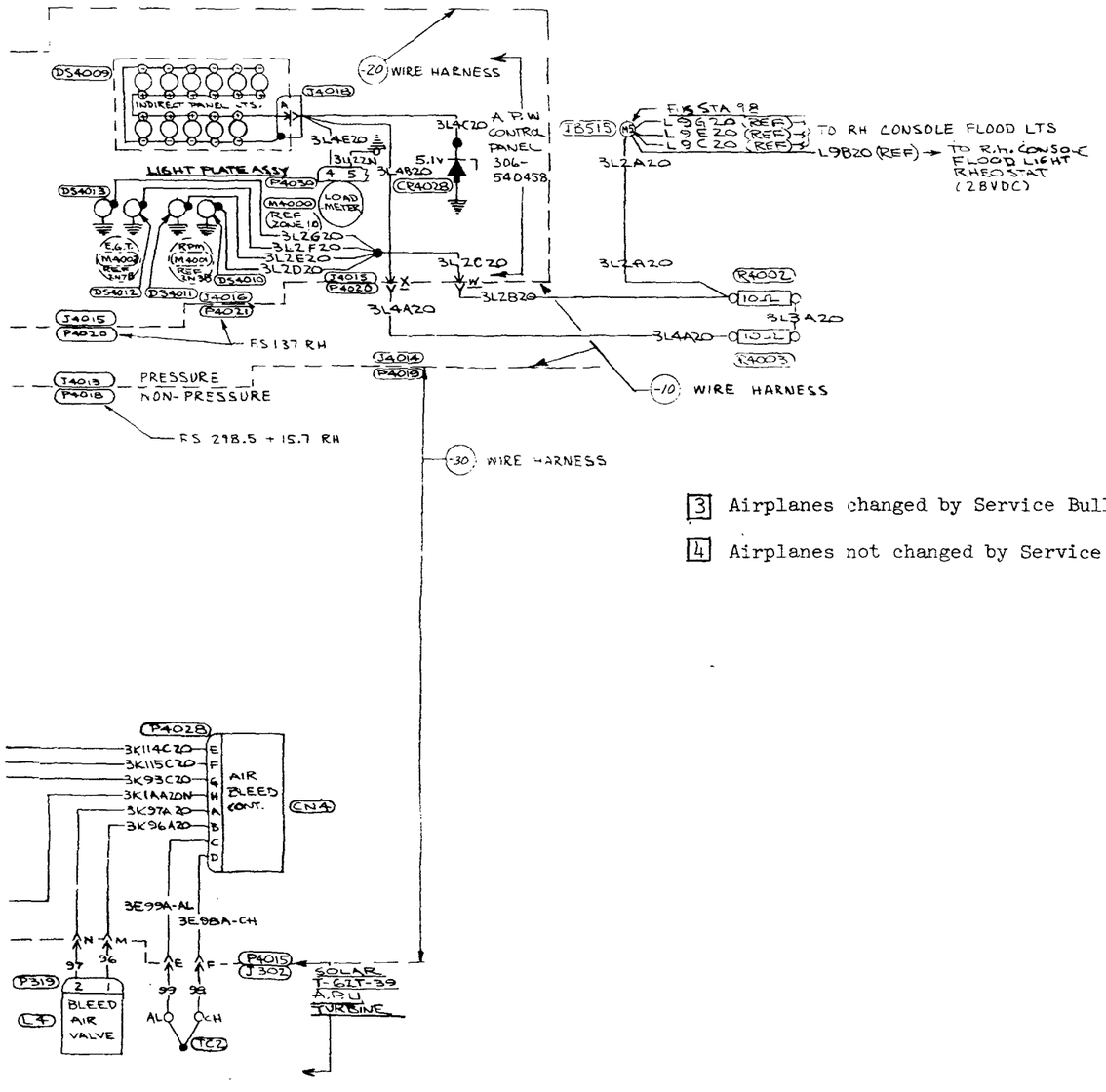

[3] Airplanes changed by Service Bulletin 74-29

[4] Airplanes not changed by Service Bulletin 74-29

APU Wiring Diagrams (306-540456-11, CN1, CN2, CN4)
Figure 49-10-4 (Sheet 8 of 10)

SEE BLOW-UP FICHE NO. ASE0411-ITEM T

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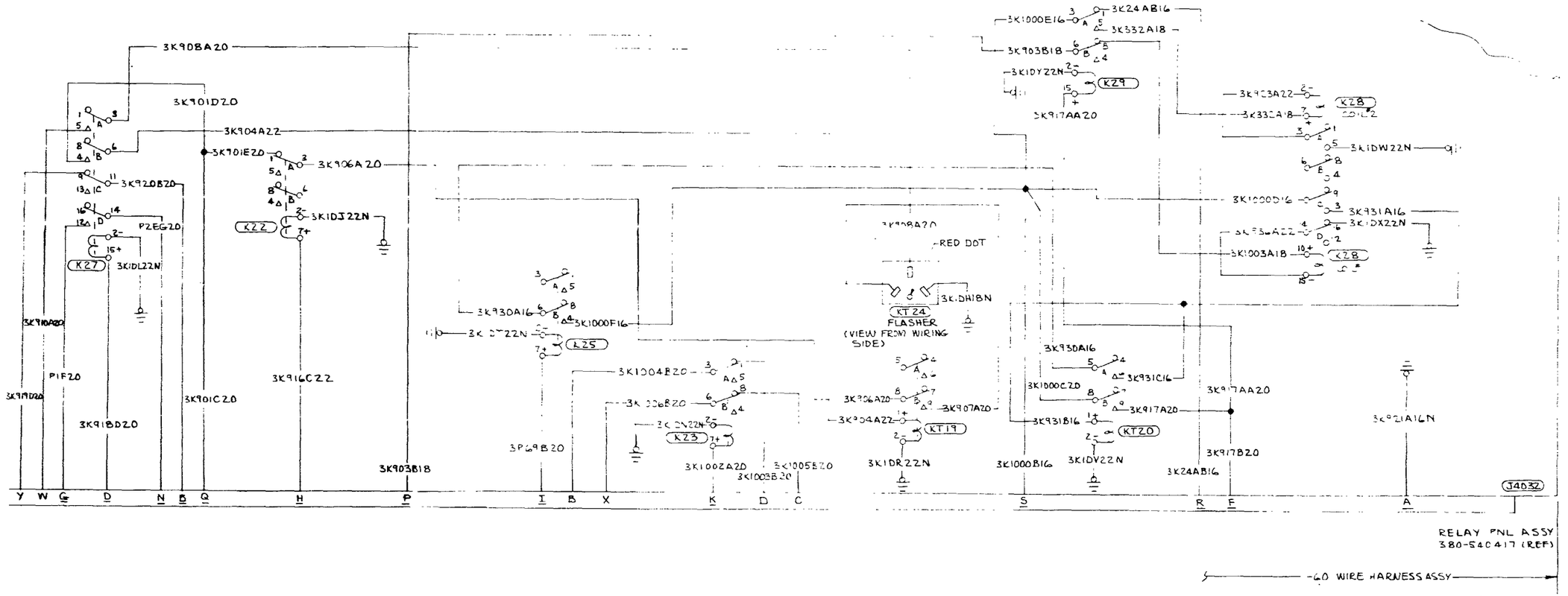

30	29	28	27	26	25	24

SEE BLOW-UP FICHE NO. ASE0411-ITEM W

APU Wiring Diagrams (306-540456-11, CN1, CN2, CN4)
Figure 49-10-4 (Sheet 9 of 10)

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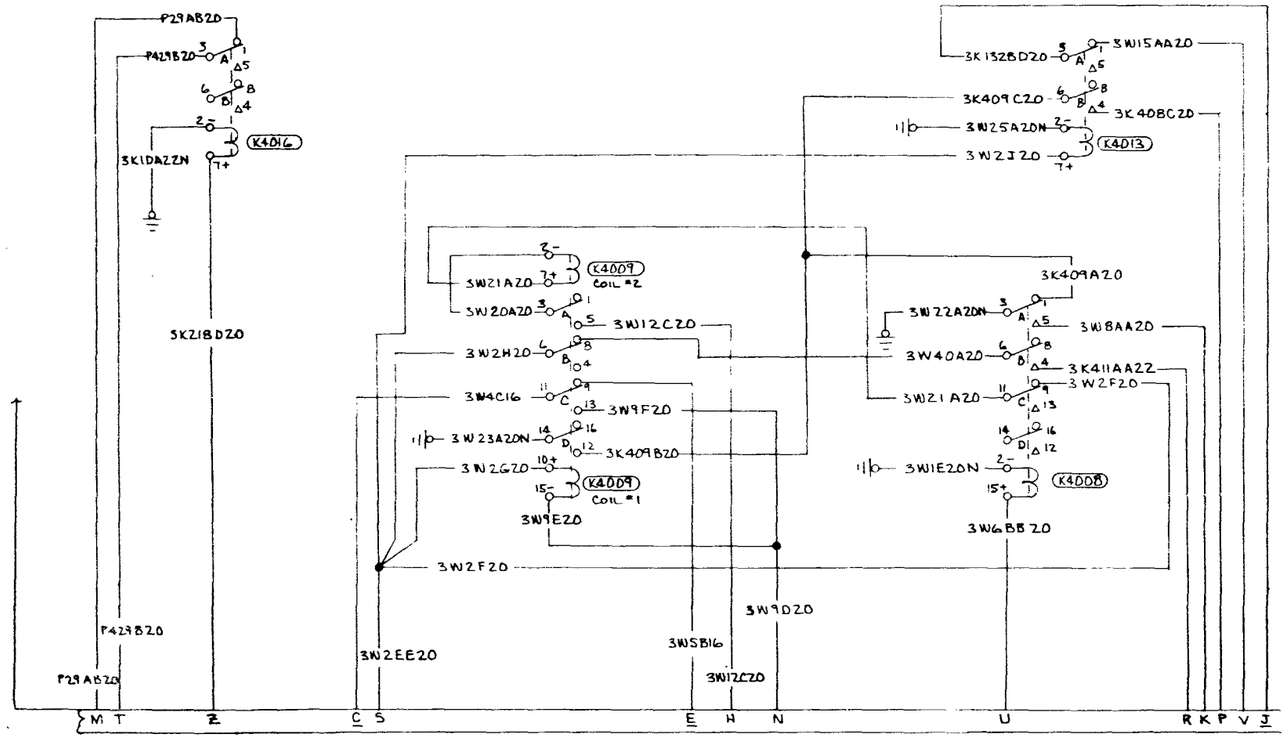

35	34	33	32	31

SEE BLOW-UP FICHE NO. ASE0411-ITEM X

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APU FUEL SUPPLY SYSTEM - DESCRIPTION AND OPERATION

1. General

The APU fuel supply system consists of a fuel boost pump, boost pump pressure switch, two fuel shut-off valves, fuel filter and the necessary electrical wiring and plumbing to complete the system. (See figure 49-00-1.) The boost pump, and the two fuel shut-off valves are located at fuselage station 298.5. The fuel filter and fuel pressure switch are located at fuselage station 324.

The APU fuel supply line is attached to the left-hand boost pump access cover located on the left wing center section fuel tank. This line is routed to fuel shut-off valves and then to the boost pump inlet port. Fuel under pressure supplied by the boost pump, is routed through the filter, and fuel pressure switch to the inlet fitting of the APU fuel containment enclosure.

During starting procedures of the APU, when the turbine start-run-off switch is moved to start, 28 volt dc electrical power is routed through the energized closed contacts (12 and 14) of relay K3 to the boost pump and two fuel shutoff valves. The fuel shut-off valves are then energized open allowing fuel to flow to the inlet port of the boost pump. Fuel (under pressure) is then routed to the APU fuel inlet fitting on the fuel containment enclosure. Refer to Solar Publication SD2-2049 for fuel system operation of the gas turbine.

APU FUEL SUPPLY SYSTEM - TROUBLE SHOOTING

1. Trouble Shooting APU Fuel Supply System

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. UNSTEADY OR ERRATIC FUEL BOOST PUMP PRESSURE		
(1) Loose connection	Perform continuity check	Repair as required
(2) Faulty boost pump bearings (bearings seizing)	Operate pump and listen for rough operation	Replace pump
(3) Voltage variation	Perform voltage check	Repair as required
B. FUEL BOOST PUMP FAILS TO OPERATE		
(1) Loose or broken connection	Check TB4002 for loose or broken connection	Repair as necessary.
(2) Faulty relay (K3) in control circuit	Check for 28 volt dc at coil of relay K3	Replace relay
(3) Pump brushes worn out	Check for high or intermittent current draw	Replace pump
C. FUEL BOOST PUMP NOISY		
(1) Loose mounting	Check pump support strap for looseness	Tighten as necessary
(2) Worn bearing	Operate pump and listen for excessive noise	Replace pump
D. NO FUEL SUPPLY AT INLET OF APU (WARNING LIGHT ON).		
(1) Boost pump not operating	Check for 28 volt dc at terminal 2 of TB4002	If voltage is present, replace boost pump
(2) Faulty fuel shut-off valve(s)	Check for 28 volt dc at terminal 2 of TB4002	If voltage is present, replace defective valve

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
E. LOW PRESSURE WARNING LIGHT REMAINS ON DURING APU OPERATION.		
(1) Faulty low fuel pressure warning switch (S4001)	Check for 28 volt dc at terminal 1 of TB4002	If voltage is present, replace fuel pressure warning switch (S4001)

APU FUEL SUPPLY SYSTEM

MAINTENANCE PRACTICES

FUEL BOOST PUMP ELECTRICAL CHECK

The APU fuel boost pump operational check-out procedures consists of applying 28 VDC electrical power to the APU system and engaging the necessary circuit breakers and switches and then listening for boost pump operation. If boost pump operates and the low fuel pressure light on the APU control panel goes out, the boost pump is considered serviceable. After accomplishing this procedure allow sufficient time for the combustor to drain before attempting an APU start.

WARNING

To prevent possible serious injury to personnel do not touch or hold APU igniter leads or connectors.

1. Gain access to APU and disconnect electrical plug (P317) from ignition exciter. This will prevent APU ignition.
2. Move aircraft ELECTRIC MASTER and BATTERY switches to ON. Check that LOW FUEL PRESS indicator light is on.
3. Momentarily depress APU RESET switch.
4. Position APU TURBINE switch to START.
5. Check that LOW FUEL PRESS indicator light goes out.

CAUTION

Do not exceed 120 seconds of starter operation in any 20 minute period.

6. Position APU TURBINE switch to OFF.
7. Move aircraft ELECTRIC MASTER and BATTERY switches to OFF.
8. Connect electrical plug (P317) to APU ignition exciter.

REPLACEMENT OF FUEL SUPPLY LINE FILTER

Removing Fuel Supply Line Filter

1. Remove access door 3 (see Figure 6-00-4, sheet 4 of 4). Ensure that electrical power is off.
2. Remove filter bowl, back-up ring and O-ring gasket from the filter head assembly.

Installing Fuel Supply Line Filter

1. Install new filter element assembly in filter bowl.
2. Install new back-up ring and O-ring packing on filter bowl.
3. *Install filter bowl in filter head and torque to 150* inch-pounds. Safety bowl with lockwire.
4. Install access door 3.

REPLACEMENT OF BOOST PUMP

Removing Boost Pump

1. Ensure that all electrical power is off.
2. Remove access door 51 (see Figure 6-00-4, sheet 4 of 4).
3. Disconnect fuel shut-off valves electrical plugs and boost pump electrical leads from TB4002.
4. Disconnect inlet and outlet fuel lines. Cap openings.
5. Remove screws (8) securing shut-off valves to support bracket.
6. Remove two bolts from drain fitting support strap.
7. Remove bolts (2) from boost pump support clamp.
8. Remove boost pump and valve assembly from aircraft.
9. Remove boost pump from shut-off valves.
10. Remove and retain fittings from boost pump.

Installing Boost Pump

1. Install new O-ring packings on fittings and install fittings (union fitting in outlet port) in boost pump.
2. Install shut-off valves on boost pump.
3. Position complete assembly in aircraft and install boost pump support clamp and bolts. Do not tighten bolts at this time.
4. Install drain fitting support strap and securing bolts.
5. Install shut-off valves support screws (8) and tighten.
6. Tighten boost pump and drain fitting support strap bolts.
7. Lockwire shut-off valves retaining screws.
8. Connect boost pump electrical leads to TB4002. Connect shut-off valves electrical plugs.
9. Install fuel lines. Then, check complete installation for security.
10. Pressure check system. (Perform procedures of APU Fuel Boost Pump Electrical Check.)

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MAINTENANCE PRACTICES (Continued)

REPLACEMENT OF FUEL PRESSURE SWITCH

Removing Fuel Pressure Switch

1. Removing access door 3. (See Figure 6-00-4, sheet 4 of 4.) Ensure that electrical power is off.
2. Disconnect electrical plug from pressure switch.
3. Remove pressure switch from T fitting.

Installing Fuel Pressure Switch

1. Install jam nut and a new O-ring packing on pressure switch.
2. Install pressure switch in T fitting.
3. Tighten jam nut.
4. Connect electrical plug to pressure switch.
5. Pressure check system. (Perform procedures of APU Fuel Boost Pump Electrical Check.)
6. Close access door 3.

REPLACEMENT OF FUEL SHUT-OFF VALVE(S)

Removing Fuel Shut-Off Valve(s)

1. Ensure that all electrical power is off.
2. Remove access door 51 (see Figure 6-00-4, sheet 4 of 4).
3. Disconnect fuel shut-off valves electrical plugs and boost pump electrical leads from TB4002.
4. Disconnect inlet and outlet fuel lines. Cap openings.
5. Remove screws securing shut-off valves to support bracket.
6. Remove two bolts from drain fitting support strap.
7. Remove bolts from boost pump support clamp.
8. Remove boost pump and valve assembly from aircraft.
9. Remove defective fuel shut-off from assembly, and remove fitting from defective valve.

Installing Fuel Shut-Off Valve(s)

1. Install new O-ring packings on fittings and install fittings in fuel shut-off valve.
2. Install shut-off valves on boost pump. Ensure that flow arrow on valve is pointing aft. Then, install drain valve T fitting at forward end of shut-off valve.
3. Position complete assembly in aircraft and install boost pump support clamp and bolts. Do not tighten at this time.
4. Install drain T fitting support strap and securing bolts.
5. Install shut-off valves support screws and tighten.
6. Tighten boost pump and drain T fitting support strap bolts.
7. Lockwire shut-off valves retaining screws.
8. Connect boost pump electrical leads to TB4002. Connect shut-off valves electrical plugs.
9. Install fuel lines. Check complete installation for security.
10. Pressure check system. (Perform procedures of APU Fuel Boost Pump Electrical Check.)

REMOVING BLEED AIR VALVE FILTER ELEMENT — VAP-AIR TYPE

Removing and Cleaning Vap-Air Bleed Air Valve Filter Element

NOTE

The Vap-Air bleed air valve filter is an integral part of the bleed air valve. To remove and clean the porous glass filter element, proceed as follows:

1. Ensure that all electrical power is off.
2. Open aft fuselage access doors on aft fuselage lower centerline.
3. Remove safety wire from filter retainer (Vap-Air P/N 2561 6257) and unscrew retainer from filter body. Refer to Figure 49-20-1.
4. Remove washer (Vap-Air P/N 2561 6235), spring (Vap-Air P/N 2601 6042), spring retainer (Vap-Air P/N 2601 6041) upper and lower O-rings (Vap-Air P/N 2591 6484), filter element (Vap-Air P/N 2601 6030), air deflector (Vap-Air P/N 2691 6485) and lower gasket (Vap-Air P/N 2591 6487).
5. Using a stiff bristle brush and a suitable solvent, methylethylketone (MEK) or equivalent, clean porous glass filter element and blow dry with clean dry compressed air.

Installing Vap-Air Bleed Air Valve Filter Element

1. Place a new lower gasket and the air deflector with a new lower O-ring in filter cavity of valve body.
2. Install clean filter element, new upper O-ring, spring and spring retainer in filter cavity.
3. Install gasket on filter retainer and screw filter retainer into valve body.
4. Tighten filter retainer and secure with lockwire.
5. Check complete installation for security.
6. Close and secure aft fuselage access doors.

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MAINTENANCE PRACTICES (Continued)

REPLACING BLEED AIR VALVE FILTER — SOLAR TYPE

NOTE

The bleed air valve filter (Solar P/N 918968C1) is a 20-micron, disposable-type, in-line filter connected to a tee fitting in the compressor discharge air pressure port on the turbine housing assembly. To replace the filter, proceed as follows:

1. Ensure that all electrical power is off.
2. Open aft fuselage access doors on aft fuselage lower centerline.
3. Disconnect line between bleed air valve and filter at filter. Refer to Figure 49-20-1.
4. Unscrew filter from tee fitting. Discard filter and O-ring.
5. Install new O-ring (Solar P/N 903266C1) over threaded end of new filter (Solar P/N 918968C1) and install into tee fitting on turbine housing.
6. Connect and tighten line from bleed air valve to filter assembly.
7. Check complete filter installation for security.
8. Close aft fuselage access doors.

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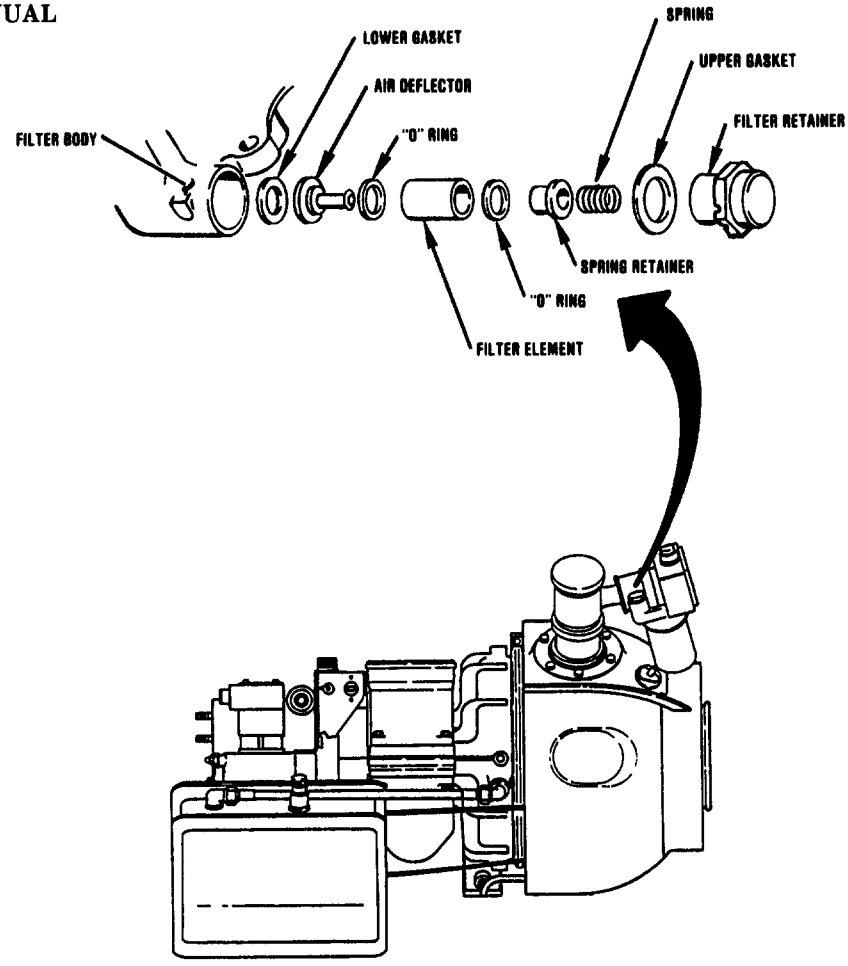
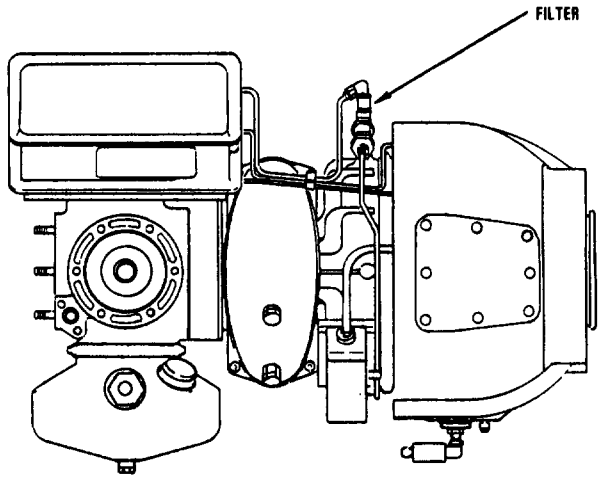


Vap Air Valve Inspection/Cleaning



Solar Bleed Air Filter

Figure 49-20-1 — APU Bleed Air Valve Filter Element

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APU LUBRICATION SYSTEM - DESCRIPTION

1. General

The lubricating system is an integral part of the APU. For information regarding this system, refer to Solar Publication SD2-2049.

APU LUBRICATION SYSTEM - MAINTENANCE PRACTICES

1. Checking Oil Level

   A. To check oil level, proceed as follows:

      (1) Gain access to APU.

      (2) The oil level should be at the center (horizontal) of the circular sight gage located on the APU oil sump.

         NOTE: Ensure that airplane wings are level before checking oil level.

      (3) If oil level is low, proceed as follows:

         (a) Remove oil sump filler cap by pressing down on cap and rotating cap $\frac{1}{4}$ turn counterclockwise.

         (b) Fill sump with approved oil until oil level reaches the full line of sight gage. The full line is indicated by a yellow dot.

            CAUTION: DO NOT OVERFILL SUMP. AN OVER FILLED SUMP WILL CAUSE OIL FOAMING AND LOW OIL PRESSURE.

         (c) Replace oil sump filler cap and close APU compartment access door.

2. Flushing Lubricating System

   The APU lubrication system must be flushed with the same type and brand of oil as used during APU service. To flush system proceed as follows:

      (1) Gain access to APU and remove oil sump pan assembly, then remove oil sump drain plug. After sump has drained, install drain plug. Do not lockwire plug at this time.

      (2) Remove oil sump filler cap and add 2 quarts of approved oil.

      (3) Start APU (refer to APU Starting Procedures).

      (4) Operate APU for three minutes. Then, shutdown APU. (Refer to APU Shutdown.)

      (5) Remove oil sump drain plug and drain all oil. Replace drain plug and torque to 10 inch-pounds. Safety plug with lockwire.

      (6) Remove oil filter element from reduction drive housing and replace with a new filter. (Refer to Solar Publication SD2-2049.)

      (7) Service oil sump. (Refer to step (3) of Checking APU Oil Level.)

APU FIRE PROTECTION AND DETECTION - DESCRIPTION AND OPERATION

1. General

Fire protection provisions for the APU include a detecting and indicating system and an electrically controlled fire extinguishing system. The detecting system consists of sensing cables, a control unit, two APU fire warning indicator lights, two alert speakers, and a fire detector power fail indicator light. The fire extinguishing system consists of an agent container (left hand), discharge valve and the necessary plumbing required to complete the system. The left hand agent container contains two cartridges. One cartridge is used for the engine and the other is used for the APU. When the APU cartridge is fired, the agent is discharged through the delivery line to the APU compartment. Electrical power for both systems is supplied by the aircraft 28 volt dc essential bus. Therefore, during APU operation, the aircraft electrical master and battery switch must be on to provide the necessary electrical power to operate the systems. During APU operation, in the event that a fire occurs or electrical power failure of the fire detection system, the APU will automatically shutdown.

APU FIRE DETECTION SYSTEM - TROUBLE SHOOTING

1. Trouble Shooting APU Fire Detection System

If the APU fire detection system does not work, the cause of the trouble must be determined and eliminated before APU operation. If a detector element is not cause of trouble, check all related systems until cause is definitely determined. If any system unit is replaced, make a complete operational checkout of system.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. APU FIRE WARNING LIGHT FAILS TO LIGHT DURING OPERATIONAL CHECK-OUT.		
(1) Faulty lamps	Remove lamps. Apply power to system. Depress "APU FIRE" switch and check for 28 volt dc at terminal 1 of light	Replace Faulty lamps or switch assembly
(2) Faulty K4009 relay (coil "1")	With electrical power on, check for 28 volt dc at terminal 10 of relay K4009	If voltage is present, replace relay
(3) Faluty fire detector control unit.	Perform control unit trip resistance test	Replace fire detector control unit
(4) Loose connections or broken wire in system	Make detailed continuity check of circuit. Visually check system for loose connections or broken wire	Repair as necessary
B. APU AUDIBLE ALERT SIGNAL DOES NOT SOUND DURING TEST.		
(1) Faulty speakers	Check for 28 volt dc at + terminal of speakers	If voltage is present, replace speakers
(2) Faulty K4008	Energize system and depress "APU FIRE" indicator light. Check for 28 volt dc at terminal 15 of relay K4008	Replace relay
(3) Loose or faulty ground connection at terminal 3 of relay K4008	Perform continuity check of relay terminal 3	Repair as necessary

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
C. APU FIRE WARNING LIGHT COMES ON DURING APU OPERATION (NO FIRE)		
(1) Defective sensing cable	Perform total resistance test	Replace defective sensing cable
(2) Defective or dirty connectors	Check connectors for moisture or metal chips	Clean, dry or replace as necessary
(3) Defective control unit.	Perform control unit trip resistance test	Replace control unit

APU FIRE PROTECTION & DETECTION SYSTEMS - MAINTENANCE PRACTICES

WARNING: CARE MUST BE TAKEN WHEN MAINTENANCE IS DONE ON THE FIRE DETECTION SYSTEM. IMPROPER HANDLING AND INSTALLATION OF THE SENSING CABLES MAY RESULT IN A FALSE WARNING.

NOTE: Do not try to straighten acceptable dents or kinks, because stresses will be set up in the sensing cables creating potential failure.

Any crushed section resulting in a decrease of cable diameter to 0.060 inch is cause for rejection.

Any abrasion of the cable 0.005 inch or more deep is cause for rejection.

Green oxide on surface of sensing cable must be maintained throughout the cable length; however, since the oxide is present for heat transfer purposes only, slight scratches or marring of the oxide can be tolerated.

1. Replacement of Sensing Cable

A. Removal

(1) Place all switches in OFF position and disconnect all external power from airplane.

(2) Gain access to APU compartment.

(3) Observe routing of sensing cable on mounting brackets. The replacement sensing cable must be installed in the same position.

(4) Remove safety wire and disconnect both connections.

(5) Unfasten sensing cable clamps and remove sensing cable.

(6) Install dust covers on open connectors.

B. Installation

(1) Place all switches in OFF position and disconnect external electrical power from airplane.

NOTE: All connectors must be clean and dry and have straight center pins before connection. Dust caps should be removed only just before connection is to be made.

(2) Insert female connector into plate, making sure connector hexnut is properly engaged into hex flange to prevent element from twisting.

(3) Thread hex jam nut into connector which projects through plate and torque 40 to 50 inch-pounds (50 inch-pounds maximum). When torquing hex jam nut to female end connectors that are not supported by connector joint plates, use a wrench on end connector hexnut to prevent element from twisting.

(4) Safety-wire jam nut to connector hexnut.

(5) Route sensing cable from this end to the first cable clamp; taking care not to bend cable excessively ($\frac{1}{2}$ inch minimum radium) or damage cable in any other way.

(6) Pinch clamp open and insert cable in clamp. Release clamp. Make sure cable is centered in spring clamp.

(7) Proceed as in steps (5) and (6) until cable has been routed through all clamps.

NOTE: Minor adjustments may be necessary to permit correct mating of sensing cable male end fitting.

(8) Mate the two fittings and torque swivel nut from 50 to 70 inch-pounds.

(9) Safety-wire swivel nut to support bracket.

(10) Make an operational check of APU fire detection system.

2. Replacement of Control Unit

A. Removal

(1) Place all airplane switches at OFF and disconnect all external electrical power from airplane.

(2) Remove safety wire that locks knurled nut to retainer bracket on control unit.

(3) Loosen and swing knurled nut outward and downward, until nut clears retainer bracket.

(4) Slide control unit forward on mounting bracket, until connector on back of control unit becomes unplugged from receptacle on mounting bracket.

(5) Remove control unit from mounting bracket.

B. Installation

(1) Place all airplane switches at OFF and disconnect all external electrical power from airplane.

(2) Place control unit on mounting bracket and align pins of connector with receptacle on bracket.

(3) Slide control unit aft on mounting bracket until control unit connector pins are fully engaged with receptacle. Apply pressure to both control unit and back of mounting bracket to ensure proper mating of connector with receptacle.

(4) Swing knurled nut upward into position over retainer bracket, and finger-tighten retainer nut.

(5) Safety-wire knurled nut to retainer bracket on control unit.

(6) Make an operational check-out of system.

3. Servicing, Inspections and Replacement of Agent Container, Discharge Valve, and Indicator Disks

NOTE: Maintenance practices, servicing and inspections of the agent container is similar to the engine agent containers. The major difference being the addition of a second cartridge with the necessary plumbing to the APU compartment and the total weight of the APU container. The APU agent container weight is 17.62 pounds when weighed with the discharge valves and cartridges. When performing maintenance on system, refer to Chapter 26.

MAINTENANCE PRACTICES (Continued)

TESTING FIRE DETECTION SYSTEM

1. Place all switches at OFF.
2. Place BATTERY and ELECTRIC MASTER switches at ON.
3. Press APU FIRE PUSH TO TEST indicator light cap. Ensure that APU FIRE indicator lights illuminate and an audible sound is heard.
4. Release APU FIRE PUSH TO TEST indicator light cap. Both lights should go out and audible sound should cease.
5. Place BATTERY and ELECTRIC MASTER switches at OFF.

APU Detector Element Total Resistance Test

CAUTION

Do not attempt to perform this test if APU compartment ambient temperature or temperature of any detector element exceeds 100°F. Failure to observe this caution will result in permanent damage to detector element.

CAUTION

Do not attempt to perform this test during APU operation or immediately after shutdown, because APU compartment ambient temperature will have exceeded 100°F.

1. Remove fire detector control unit (CN4000) from mounting rack on electrical equipment rack at FS 150. Install a jumper wire between pins 6 and 8 of receptacle on control unit mounting rack.
2. Connect one lead of a multimeter to the jumper wire.
3. Connect other lead of multimeter to airplane structure. Select a clean metal surface which is not anodized or painted.
4. Check resistance with multimeter. The resistance should not be less than 500,000 ohms.

NOTE

If total resistance is less than that stated in step 5, make an individual resistance test of each component and wiring. (Refer to the following table for individual resistance values.) Replace faulty component or wiring, and repeat test procedure.

5. Discharge fire detector wiring by operating discharge switch on megohmmeter or by shorting megohmmeter test leads with screwdriver having a well-insulated handle.
6. Disconnect multimeter leads from airplane wiring harness.
7. Remove jumper wire between pins 6 and 8.
8. Install fire detector control unit in mounting rack.
9. Make an operational test of system.

EDISON SYSTEM COMPONENTS	PART NO.	MEGGER TEST (500 VDC) MIN MEGOHMS	CENTER WIRE RES. MAX OHMS
Sensing cable	244-24082		12.0
Male connector	42991-00	must pass [1]	0.05
Female connector	42990-00	must pass [1]	0.05

[1] Detector elements must be disconnected from system wiring before test is made.

APU Control Unit Trip Resistance Test

NOTE

The following test requires a resistance decade box and a 28 VDC external power source.

1. Disconnect all external power from airplane.

NOTE

The APU Fire Detector Control Unit could be located in various areas. Many have been located on the lower RH side of the aft fuselage just forward of F.S. 333. Check the IPC for your aircraft model for other locations.

2. Disconnect airplane wiring from mounting bracket terminal 6.
3. Connect one lead of decade box to terminal 6 of control unit mounting rack electrical receptacle, and the other lead to a clean ground.

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MAINTENANCE PRACTICES (Continued)

APU Control Unit Trip Resistance Test (Cont'd)

4. Adjust decade box to 200 ohms.
5. Connect a 28 VDC external power source to airplane.
6. Slowly decrease decade box resistance until APU FIRE indicator light comes on. The light should come on when the decade box resistance is between 90 and 110 ohms.
7. Slowly increase decade box resistance until APU FIRE indicator light goes out. The light should go out when the decade box resistance is at or below 150 ohms.

NOTE

If indicator light does not go out within limits specified in step 7, check resistance of external resistor connected to terminals 1 and 4 of fire detector control unit mounting rack receptacle. The resistor should measure 100 ohms (±1 percent). If resistor is not correct, remove and replace resistor; then repeat test. If resistor value is correct, remove and replace control unit. Repeat test.

8. Disconnect external power from airplane.
9. Disconnect decade box and reconnect airplane wiring previously removed from mounting bracket terminal 6.
10. Check and reset MK7 APU FIRE magnetic circuit breaker (FS 401 APU relay box in aft equipment bay).
11. Make an operational checkout of system.

CHECKOUT OF FIRE-EXTINGUISHING SYSTEM

1. Disconnect electric lead to APU discharge valve at agent container.
2. Install a General Electric 1819 lamp between disconnected electrical lead and the discharge valve. This lamp limits the testing current to 0.035 amp, the max safe testing current.
3. Place BATTERY and ELECTRIC MASTER switches to ON.
4. Lift guard of the FIRE EXT switch and position switch to ON.
5. Check that test light comes on.
6. Position FIRE EXT switch to OFF, lower guard and safety with 0.018 copper wire.
7. Place BATTERY and ELECTRIC MASTER switches to OFF.
8. Remove test light and connect electrical lead to discharge valve.

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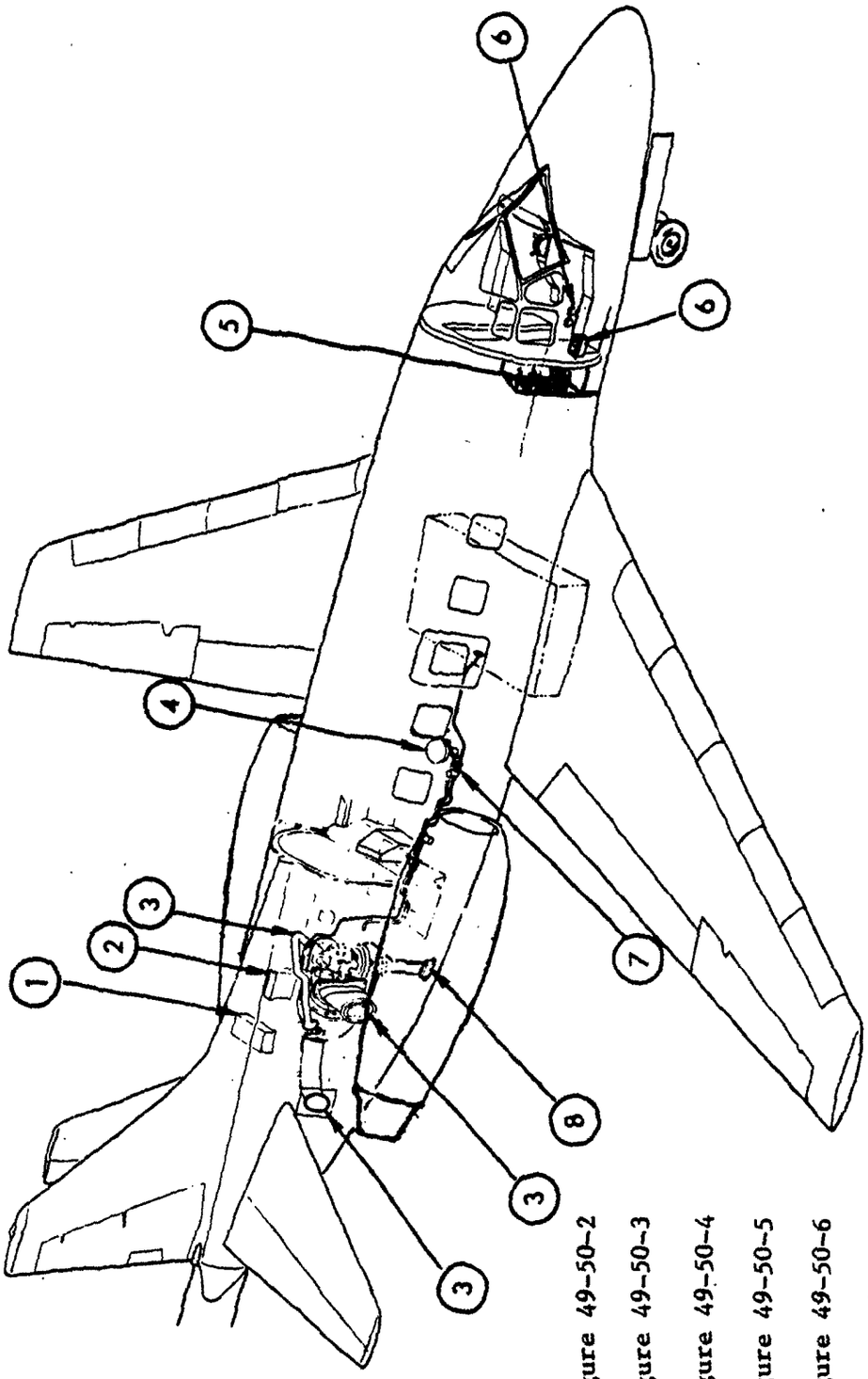
APU PARTS LIST - DESCRIPTION

1. General

This portion of Chapter 49 lists the major assemblies required for an auxiliary power unit (APU) installation in the NA-265-80 models of the Sabre 75. This parts list includes only those items peculiar to the APU installation. A parts listing of all Solar part numbers may be found in the Solar Publication SD2-2049.

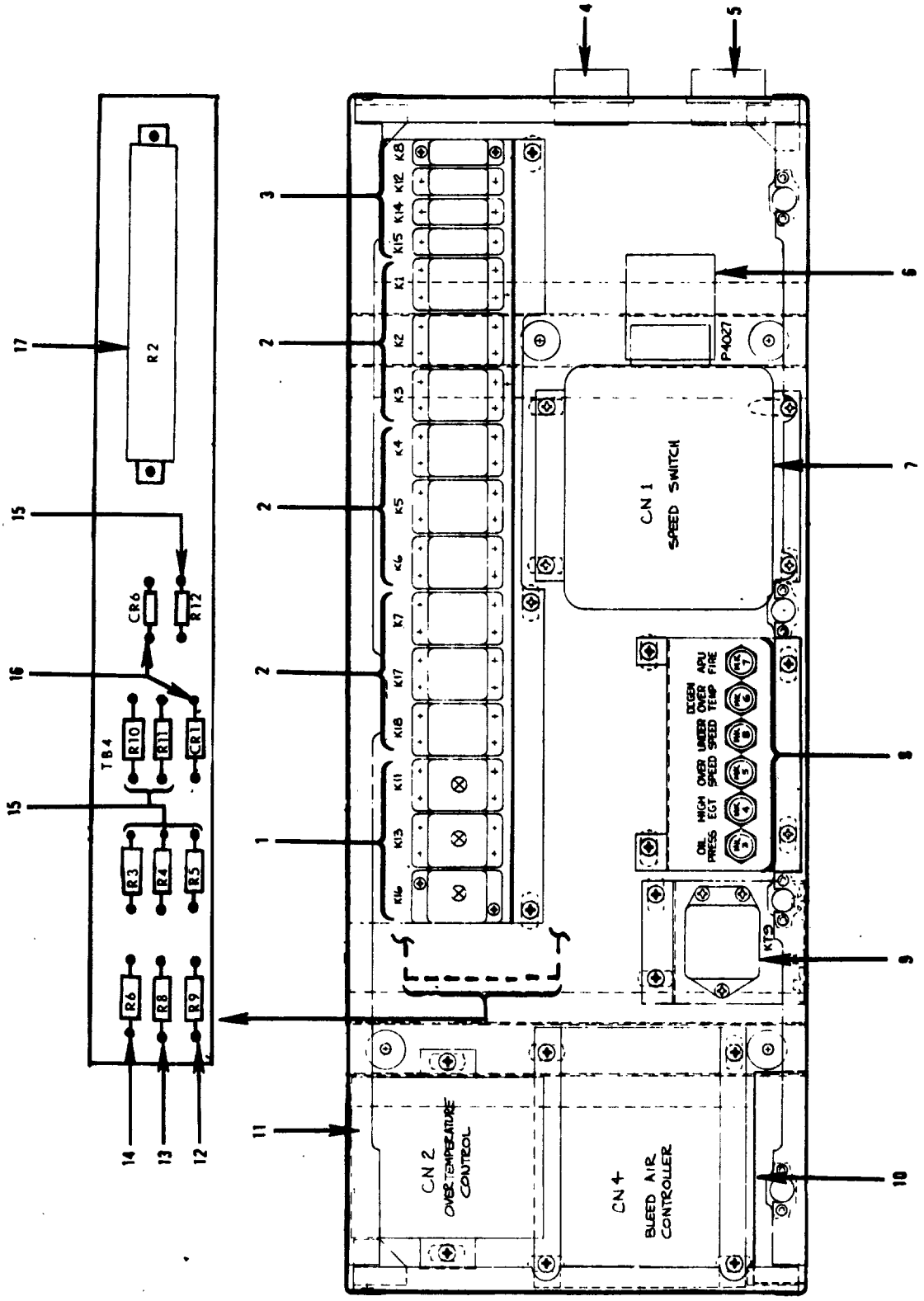
This parts list consists of an index number, part number, vendor, description and quantity columns. The index number column contains the numbers that index the part or subassembly to those shown on the corresponding illustration. The part number of each part or subassembly is listed in the second column. The vendor column consists of a number which corresponds with the applicable number found at the bottom of the parts list page. The vendor's name and address is provided for future procurement. All part numbers with the prefix 265, 306, 370 or 380 are Rockwell International numbers and they will not be coded as such. The description column gives the nomenclature of the item. Attaching or detailed parts required for the major installations along with the part numbers are also listed under the description. The major portion of these parts are not assigned an index number. The quantity column indicates the number of parts (as shown on the corresponding illustration), that are required per assembly.

Figure 49-50-1 is used as a major assembly locator. The figure reference shown in this figure may be used in locating the major assembly, which in turn lists the applicable part.



(1) Figure 49-50-2
(2) Figure 49-50-3
(3) Figure 49-50-4
(4) Figure 49-50-5
(5) Figure 49-50-6
(6) Figure 49-50-7
(7) Figure 49-50-8
(8) Figure 49-50-9

Part Assembly Locator
Figure 49-50-1



NA-265-80-1, -2, -3, 4, -6, -7, -8, -9 AND -11 AIRPLANES

Parts List
Figure 49-50-2 (Sheet 1)

Key to figure 49-50-2 (Sheet 1)

INDEX NO.	PART NO.	VENDOR	DESCRIPTION	QTY
1.	BR23AXC2-V3	1	Relay	3
	AN515-6R10		Screw	6
	AN936-A6B		Washer	6
2.	BR14X-150A2-26V	1	Relay	9
	AN515-6R6		Screw	18
	AN936A6B		Washer	18
3.	BR7X-300E8-26V	1	Relay	4
	AN515-6R4		Screw	8
	AN936A6B		Washer	8
4.	KPTM00E-22A41S	2	Receptacle	1
	AN515-4R5		Screw	4
	MS21042L04		Nut	4
	031-9154-000		Socket(alumel)	1
	031-9154-001		Socket(chromel)	1
5.	KPTM00E-22A1P	2	Receptacle	1
	AN515-4R5		Screw	4
	MS21042L04		Nut	4
6.	CA3106RX24-5S	2	Plug	1
7.	CL-629-1YA	3	Speed Switch	1
	AN515-8R6		Screw	4
8.	3MC3-068	4	Breaker, Circuit	6
9.	R410AKC-1501-BC	5	Relay	1
	S0-1009-7126		Socket	1
10.	26930986	6	Controller	1
	AN515-8R6		Screw	4
11.	CL-694-1VA	3	Control, Overtemp	1
	AN515-8R6		Screw	2
12.	4361	7	Resistor(10 ohm $3\frac{1}{4}$W)	1
13.	4438	7	Resistor(3900 ohm $3\frac{1}{4}$W)	1
14.	4392	7	Resistor(100 ohm 50W)	1
15.	4087	7	Resistor(75 ohm $1\frac{1}{2}$W)	6
16.	MR2068	8	Diode	2

Key to figure 49-50-2 (Sheet 1)

INDEX NO.	PART NO.	VENDOR	DESCRIPTION	QTY
17.	0400B	7	Resistor	1

1. Babcock Relays, Costa Mesa, Calif.
2. Cannon Electric Co., Los Angeles
3. Crydom Controls Div., International Rectifier Corp. Santa Ana, Calif.
4. Metals & Controls Inc. Attleboro, Mass.
5. Leach Relay Co., Los Angeles
6. Vap-Air Division, Vapor Corp, Chicago, Illinois
7. Ohmite Co., Skokie, Ill.
8. Motorola Corp., Phoenix, Ariz.

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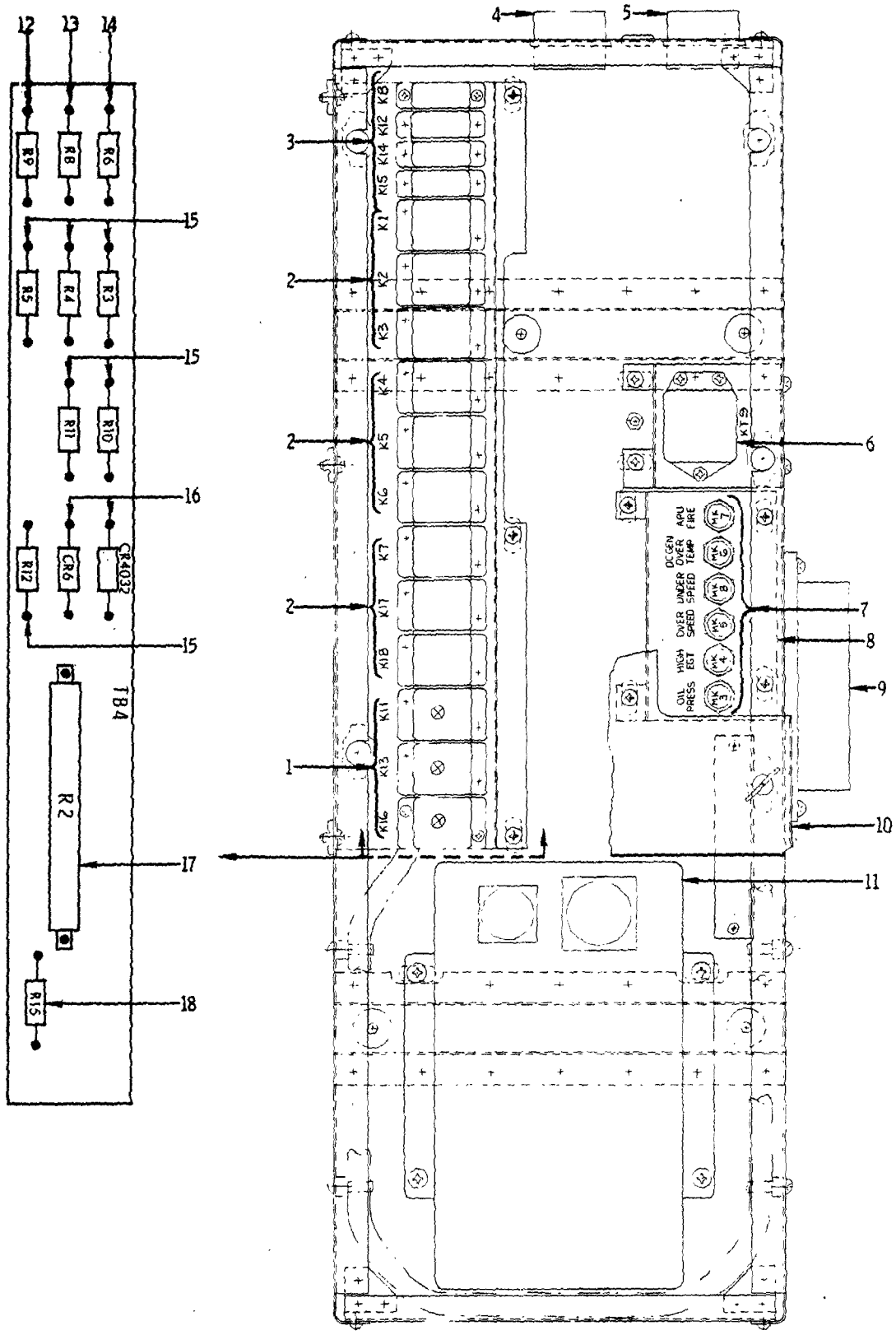
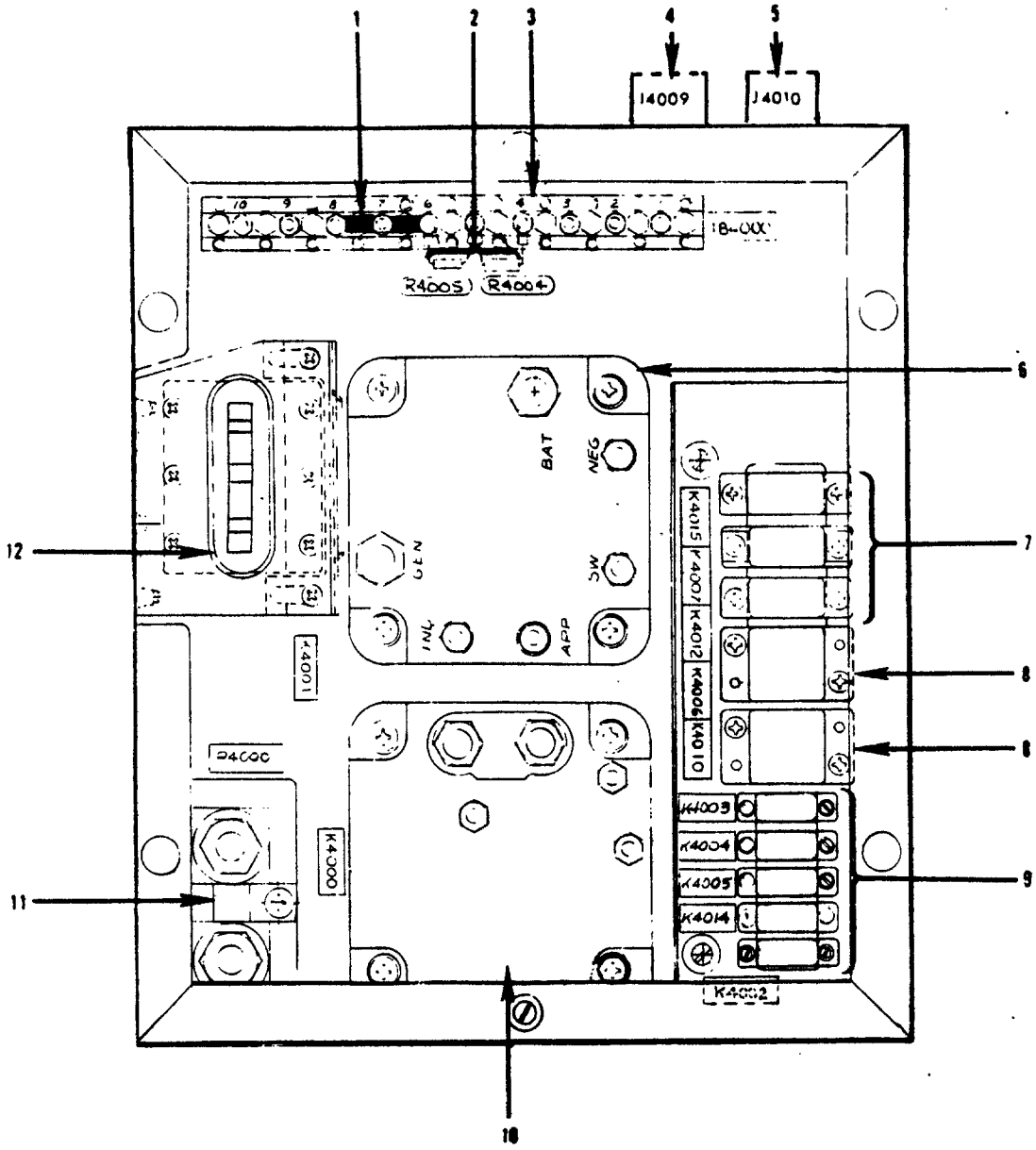
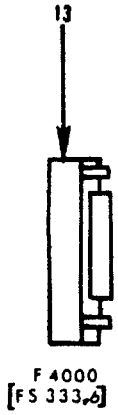


NA-265-00-5, -10, -12 AND SUBS.

Key to figure 49-50-2 (Sheet 2)

INDEX NO.	PART NO.	VENDOR	DESCRIPTION	QTY
1.	BR23AXC2-V3	1	Relay	3
	AN515-6R10		Screw	6
	AN936-A6B		Washer	6
2.	BR14X-150A2-26V	1	Relay	9
	AN515-6R6		Screw	18
	AN936A6B		Washer	18
3.	BR7X-300E8-26V	1	Relay	4
	AN515-6R4		Screw	8
	AN936A6B		Washer	8
4.	KPTM00E-22A41S	2	Receptacle	1
	AN515-4R5		Screw	4
	MS21042L04		Nut	4
	031-9154-000		Socket (alumel)	1
	031-9154-001		Socket (chromel)	1
5.	KPTM003-22A1P	2	Receptacle	1
	AN515-4R5		Screw	4
	MS21042L04		Nut	4
6.	R410AKC-1501-BC	5	Relay	1
	SO-1009-7126		Socket	1
7.	3MC3-068	4	Breaker, Circuit	6
8.	306-540456-31		Circuit Breaker Panel	1
9.	AN5534-2		Thermocouple	1
10.	306-540456-81		Relay Box Cover	1
11.	116076-0		Control Assembly	1
12.	4361	7	Resistor (10 ohm $3\frac{1}{4}$W)	1
13.	4438	7	Resistor (3900 ohm $3\frac{1}{4}$W)	1
14.	4392	7	Resistor (100 ohm 50W)	1
15.	4087	7	Resistor (75 ohm $1\frac{1}{2}$W)	6
16.	MR2068	8	Diode	2
17.	0400B	7	Reisitor	
18.	84475		Resistor (30.1K, 5W)	



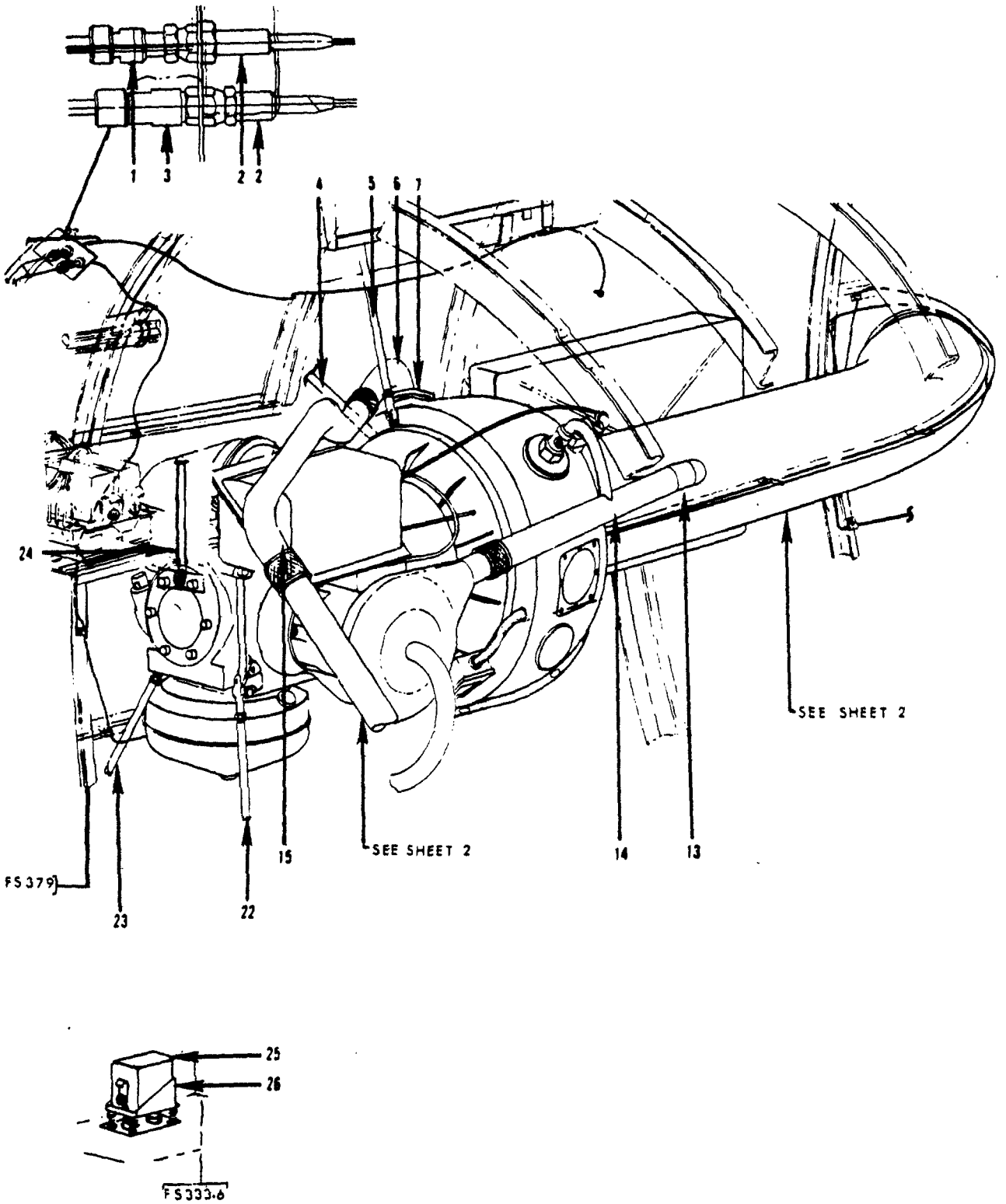


F 4000
[FS 333,6]

Key to figure 49-50-3

INDEX NO.	PART NO.	VENDOR	DESCRIPTION	QTY
1.	MS25226-2-3		Busbar	1
2.	Type 884 31885 Terminal	1	Resistor (4.02 ohm)	2 4
3.	MS27212-1-10		Board Terminal	1
4.	KPTM00E-20-41P	2	Plug	1
5.	KPTM00E-18-11-P	2	Plug	1
6.	A-700AW	3	Cutout	1
7.	BR7X-300E8-26V 137VB8/1LA11-42	4	Relay Socket	2 2
8.	BR14X-150A2-26V 137VB16/1LE20-41	4	Relay Socket	2 2
9.	BR12-675A8-S73	4	Relay	3
10.	AM-711CJ	3	Relay	1
11.	MS91587-4		Shunt	1
12.	940-006-100	5	Breaker Circuit	1
13.	FLLJ	7	Fuse	1

1. Ohmite Manufacturing Co., Skokie, Illinois
2. Bendix Corporation, Scintilla Division, Sidney, New York
3. Hartman Electrical Mfg. Co., Mansfield, Ohio
4. Babcok Control Products, Costa Mesa, Calif.
5. Mechanical Products Inc., Jackson, Michigan
6. General Electric, Erie, Pa.
7. Burndy Corp., Norwalk, Conn.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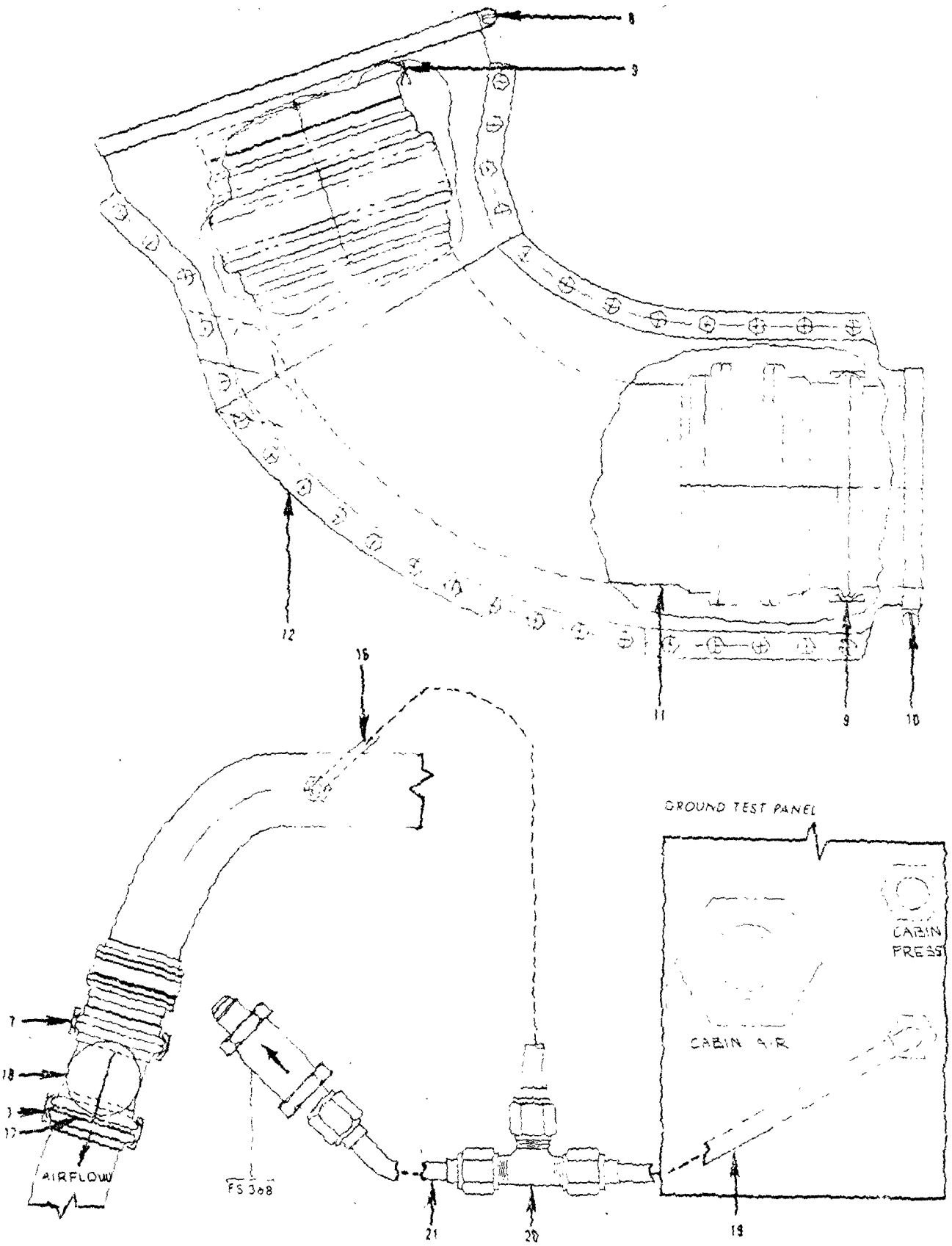

Parts List
Figure 49-50-4 (Sheet 2)

Key to figure 49-50-4 (Sheet 2)

INDEX NO.	PART NO.	VENDOR	DESCRIPTION	QTY
1.	42991-00	1	Connector	1
2.	244-24082	1	Cable, Fire Detector	2
3.	42990-00	1	Connector	1
4.	370-313026-15		Tube, Support	1
5.	370-313026-13		Tube, Support	1
6.	012332	2	Duct	1
7.	MV85623-K3-300V		Clamp	1
8.	MS35842-16 MS35842-15		Clamp Clamp	1 1
9.	4276-623M	3	Clamp	2
10.	MS35842-16		Clamp	1
11.	380-870007-11		Duct, Exhaust	1
12.	380-870005-11 380-870005-21 (Attaching Parts) AN3-3A LD153-0002-2403		Shroud, Bottom Shroud, Top Bolt Washer	1 1 33 33
13.	24540-250	3	Clamp	1
14.	306-870011-213		Duct, Gen.	1
15.	012332	2	Duct, Bleed Air	1
16.	370-870003-215		Tube	1
17.	370-8700083-3 NAS1593-037		Valve, Check "O" Ring	1 2
18.	26940980	4	Valve, Bleed Air Control	1
	or 115351-1	5	Valve, Bleed Air Control	1
19.	370-870003-5		Tube	1
20.	AN824-4		"T" Fitting	1
21.	370-870003-7		Tube	1
22.	370-313026-41		Tube, Support	1

Key to figure 49-50-4 (Sheet 2)

INDEX NO.	PART NO.	VENDOR	DESCRIPTION	QTY
23.	370-31026-51		Tube, Support	1
24.	370-313026-31		Tube, Support	1
25.	227-28-7	1	Control, Fire Det.	1
26.	41531	1	Mount	1
	41250-100	1	Resistor	1

1. Thomas A. Edison Ind. Instrument Div. West Orange, New York
2. Flexonics Division Universal Oil Products Co., Bartlett, Illinois
3. Marman Division of Aeroquip Corp. Los Angeles, Calif.
4. Vap-Air Division, Vapor Corp., Chicago, Illinois
5. Solar Mfg. Co., San Diego, Calif.

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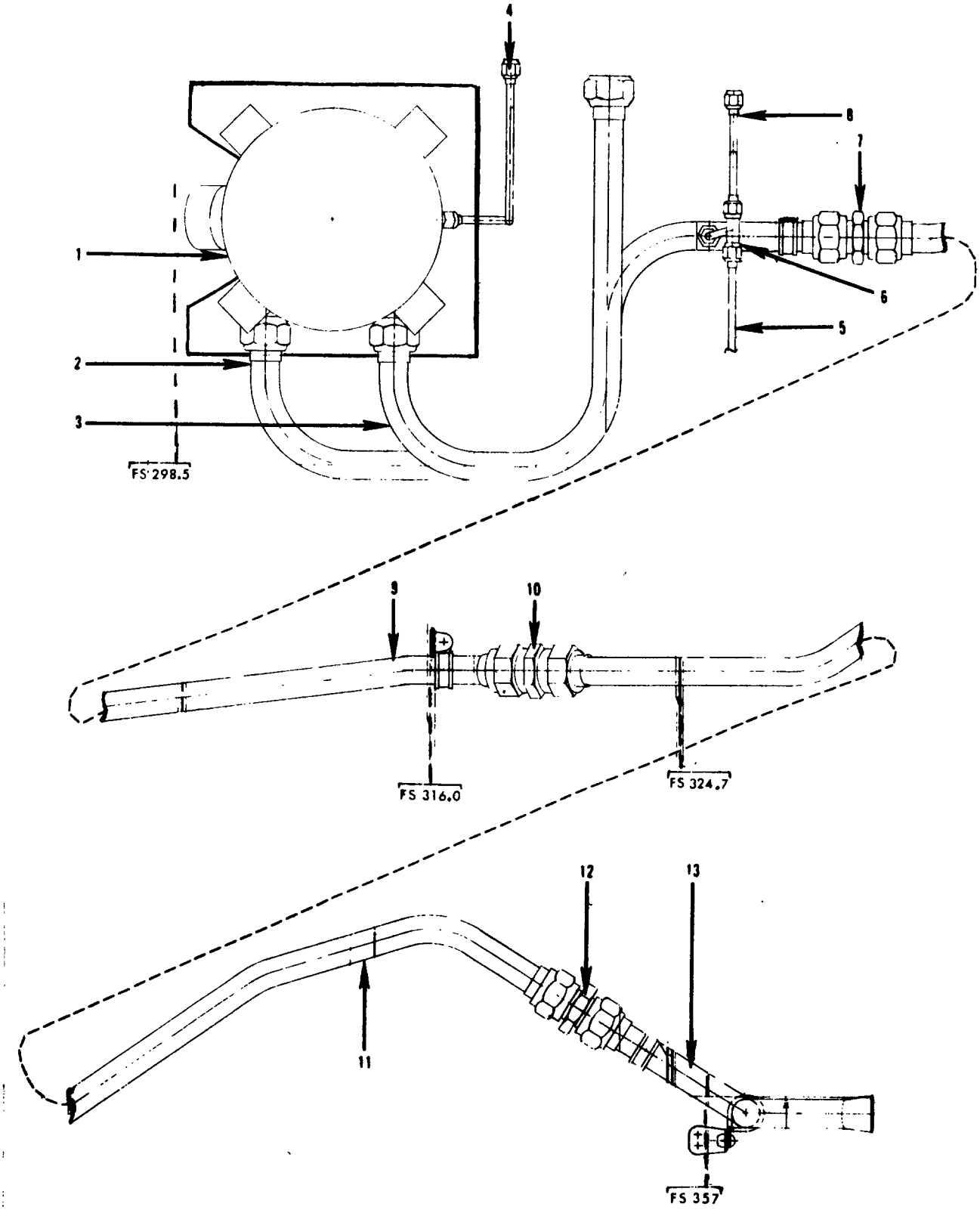
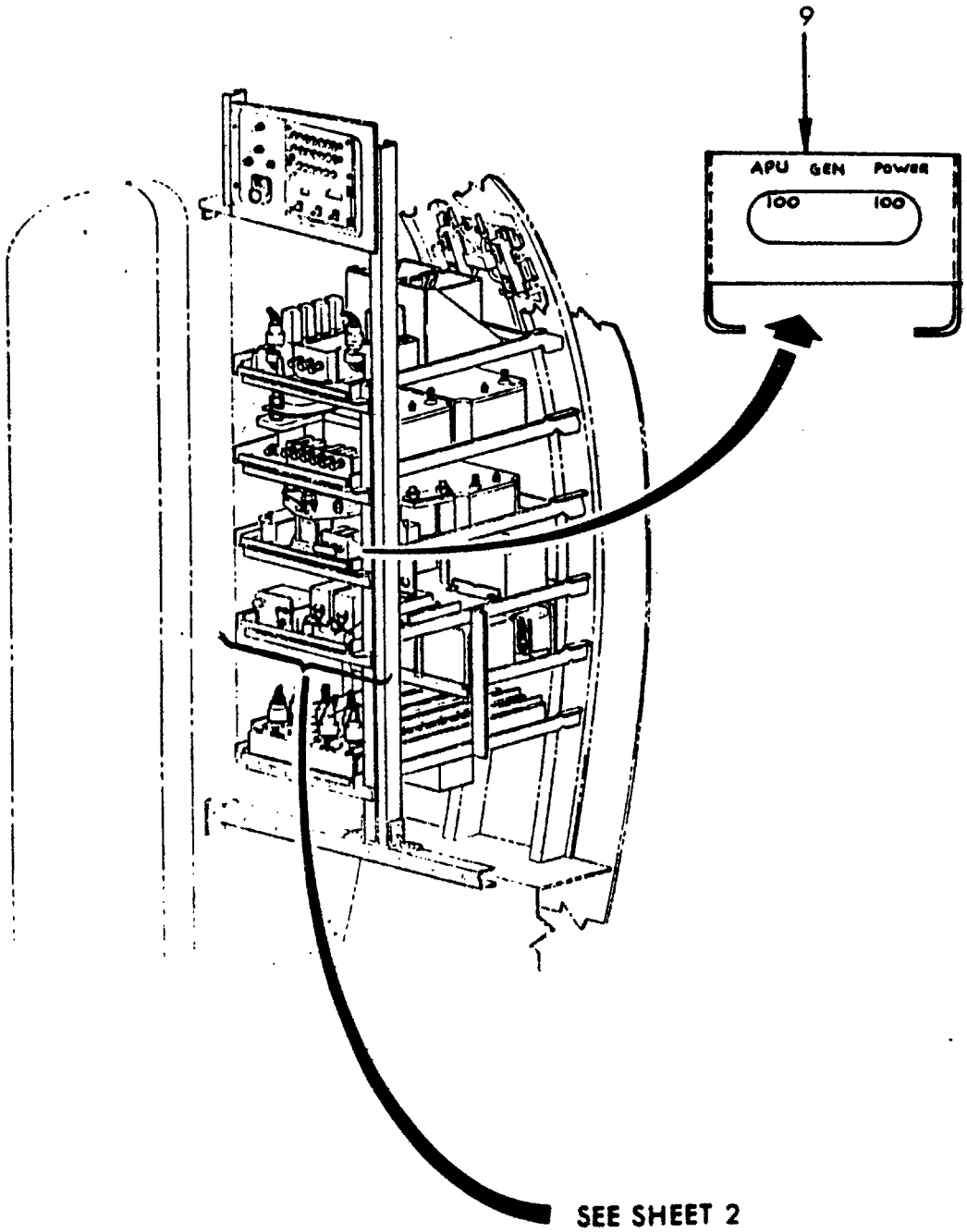


Parts List
Figure 49-50-5

Key to figure 49-50-5

INDEX NO.	PART NO.	VENDOR	DESCRIPTION	QTY
1.	895724	1	Container	1
2.	370-735053-3		Tube	1
3.	370-735053-5		Tube	1
4.	370-735053-13		Tube	1
5.	370-735053-19		Tube	1
6.	AN824-4D		"T" Fitting	1
7.	AN815-16D		Union	1
8.	370-735053-17		Tube	1
9.	370-735053-7		Tube	1
	MS21919DG16		Clamp	1
	NAS1100-3-7		Screw	1
	MS21042L3		Nut	1
	LD153-0002-1203		Washer	1
10.	AN815-16D		Union	1
11.	370-735053-9		Tube	1
	MS21919DG16		Clamp	1
	NAS1100-3-7		Screw	1
	MS21042L3		Nut	1
	LD153-0002-1203		Washer	1
12.	AN815-16D		Union	1
13.	370-735053-11		Tube	1
	MS21919DG16		Clamp	2
	NAS43DD3-48		Spacer	1
	NAS1100-3-18		Screw	1
	MS21042L3		Nut	1
	LD153-0002-1203		Washer	1

1. Walter Kidde & Co., Inc. Belleville, N.J.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APU GEN POWER
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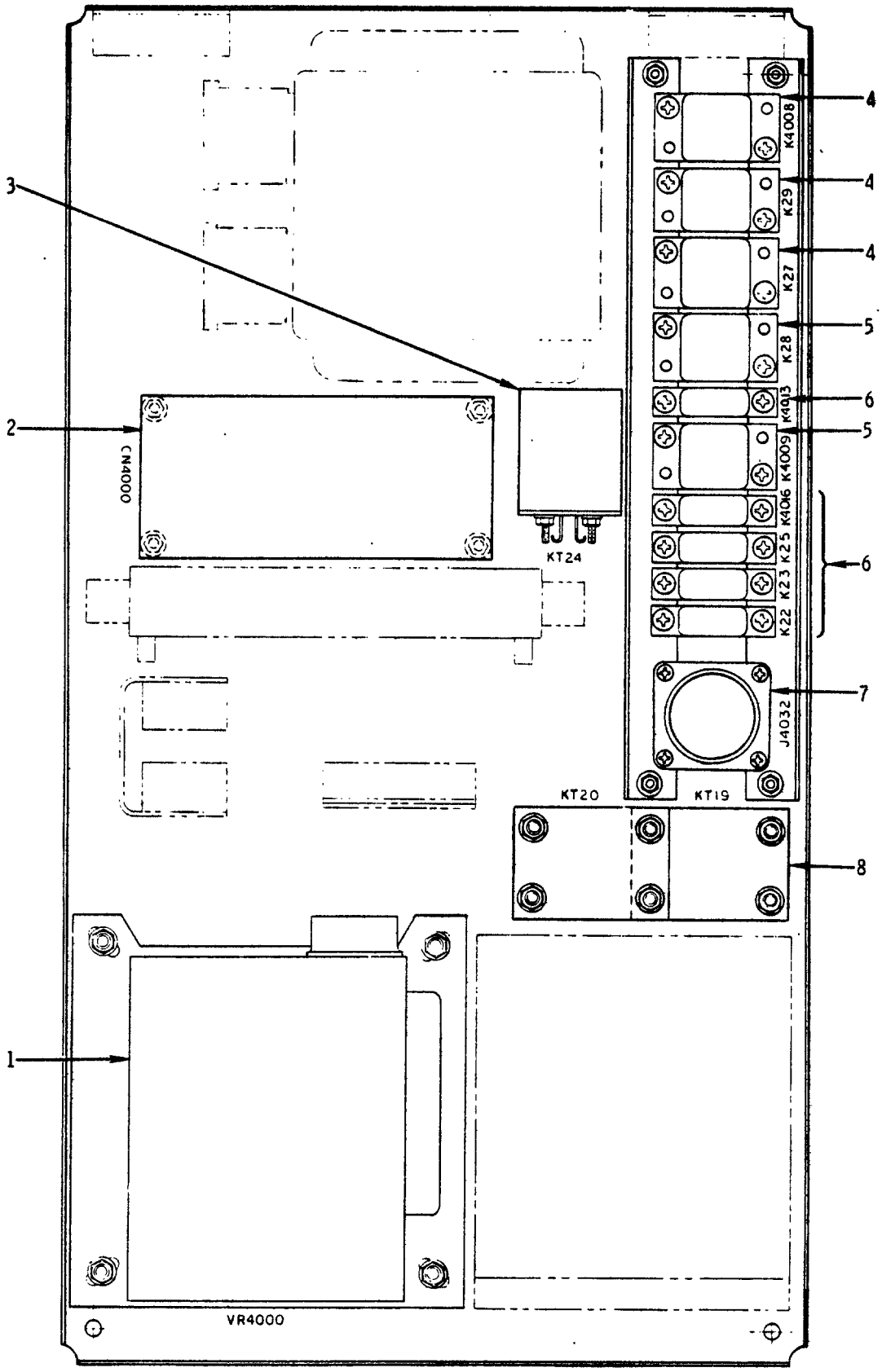
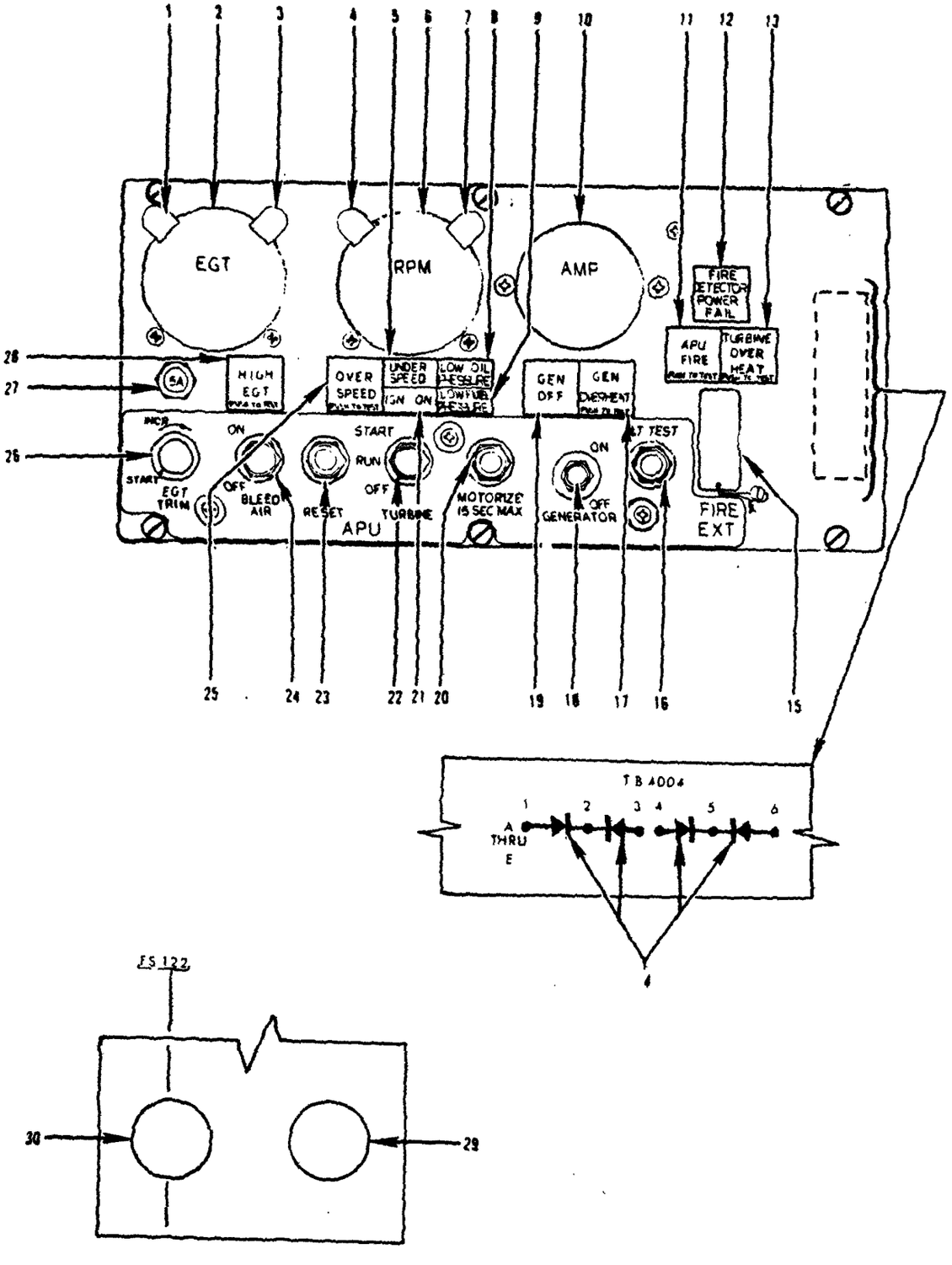


Parts List
Figure 49-50-6 (Sheet 2)

Key to figure 49-50-6 (Sheet 2)

INDEX NO.	PART NO.	VENDOR	DESCRIPTION	QTY
1.	3S2060DC168C1B	5	Generator Control Unit	1
2.	22702807	5	Control	1
	41531	5	Mount	1
	41250-100	5	Resistor	1
3.	38-12-02292	3	Flasher Unit	1
4.	BR14X-150A2-26V	1	Relay	3
	137VB16/1LE20-41	4	Socket	3
5.	BR23AXC2-V3	1	Relay	2
	137VB16/1LE20-41	4	Socket	2
6.	BR7X-300E8-26V	1	Relay	5
	137VB81LA11-42	4	Socket	5
7.	KPTM00E-22A41P	4	Receptacle	1
8.	415ACS-2	2	Time Delay Relay	2
9.	940-006-100		Breaker, Circuit	1

1. Babcock Relays, Costa Mesa, Calif.
2. Thomas A. Edison Ind., Instrument Div., West Orange, N.Y.
3. Electronic Resources Inc., Los Angeles, Calif.
4. Viking Industries Inc., Chatsworth, Calif.
5. General Electric, Erie, Penn.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EGT
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TURBINE OVER HEAT
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OVER SPEED
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LOW OIL PRESSURE
IGN
ON
LOW FUEL PRESSURE
GEN OFF
GEN OVERHEAT
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Key to figure 49-50-7

INDEX NO.	PART NO.	VENDOR	DESCRIPTION	QTY
1.	A-8970-B-4-Y	1	Light, Post	1
2.	47560-2	2	EGT Meter	1
	AN515-6R12		Screw	2
	AN961-6		Washer	2
	MS21042L06		Nut	2
3.	A-8970-B-4-Y	1	Light	1
4.	A-8970-B-4-Y	1	Light	1
5.	453-850-5952-007	6	Light	1
6.	47558-2	2	RPM Meter	1
	AN515-6R12		Screw	2
	AN961-6		Washer	2
	MS21042L06		Nut	2
7.	A-8970-B-4-Y	1	Light	1
8.	453-850-5952-005	6	Light	1
9.	453-850-5952-005	6	Light	1
10.	8DW84LAA220	3	Ammeter	1
	MS28042-1		Clamp	1
11.	451-850-5951-003	4	Swtichlight	1
12.	453-850-5952-009	4	Light	1
13.	451-850-5951-010	4	Switchlight	1
14.	MR2068	11	Diode	20
15.	MS35059-26		Switch	1
	MS25224-3		Guard	1
16.	2PB11-T2	6	Switch, Pushbutton	1
17.	451-850-5951-009	4	Switchlight	1
18.	AP13RC-549-5	7	Switch	1
	AN507-632R3		Screw	2
19.	453-850-5952-008	4	Light	1
20.	10-9	8	Switch, Pushbutton	1

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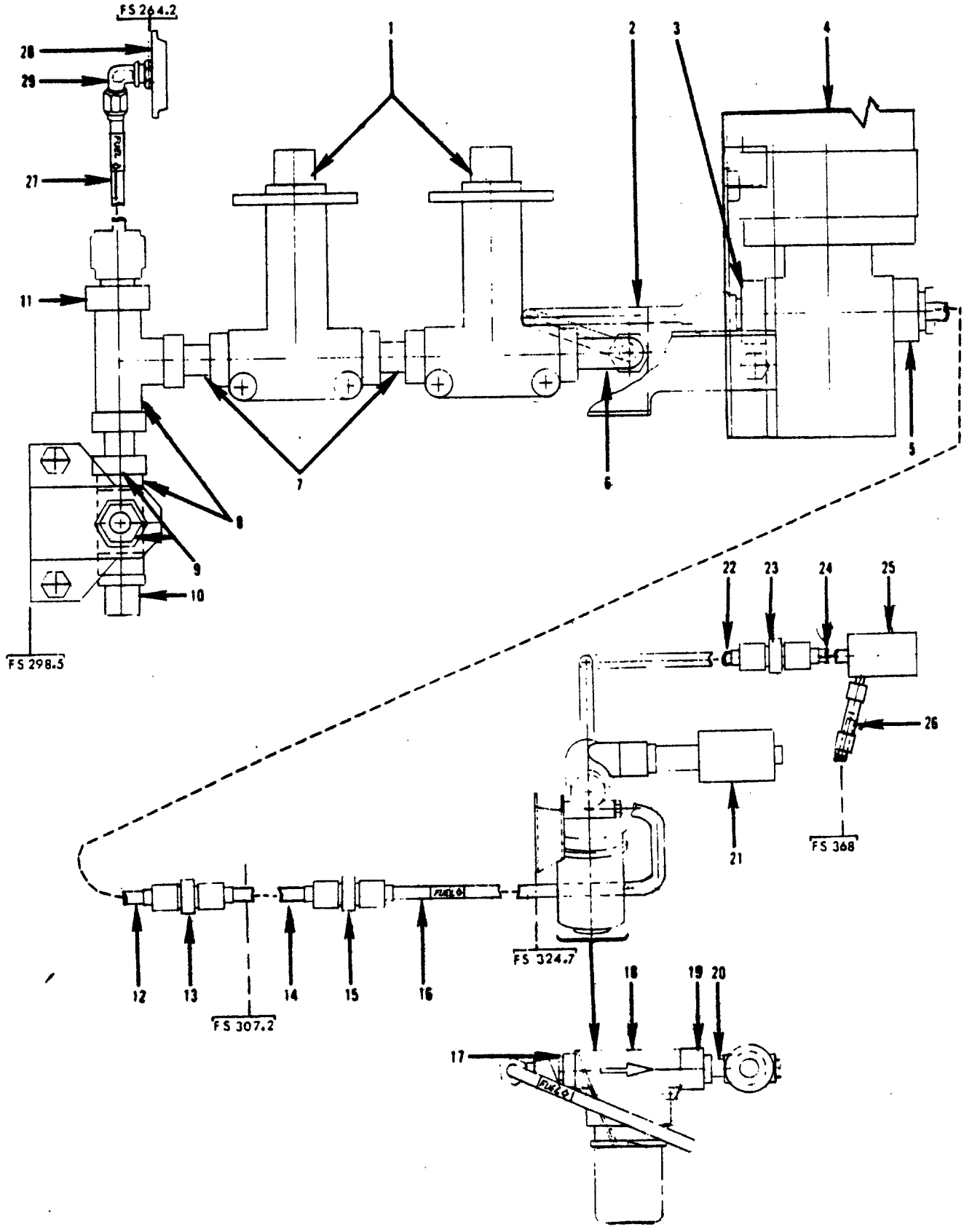
Key to figure 49-50-7

INDEX NO.	PART NO.	VENDOR	DESCRIPTION	QTY
21.	453-850-5952-007	4	Light	1
22.	4TL11-50M	6	Switch, Lever Lock	1
23.	3PB11-T2	6	Switch, Pushbutton	1
24.	MS35059-23		Switch	1
25.	451-850-5951-005	4	Switchlight	1
26.	B14/KR-13 1914C (NA265-80-1-2-3-4 -6-7-8-9 and -11 airplanes)	9 12	Potentiometer Knob	1 1
27.	7274-2-5	10	Breaker, Circuit	1
28.	451-850-5951-004	4	Switchlight	1
29.	SC628H	13	Speaker	1
30.	SC628W	13	Speaker	1

1. Grimes Mfg. Co., Urbana, Ohio
2. Solar Mfg. Co., San Diego, Calif.
3. General Electric Co., West Lynn, Mass.
4. Korry Mfg. Co., Seattle, Washington
5. Allen Bradely Co., Newark Electronics, Inglewood, Calif.
6. Micro Switch Co., Freeport, Illinois
7. General Electric Co., Waynesboro, Virginia
8. Grayhill Inc., La Grange, Illinois
9. Centralab Co., Milwaukee, Wisc.
10. Metals & Controls Inc., Texas Inst. Inc., Attleboro, Mass.
11. Motorola Corp., Phoenix, Arizona
12. Harry Davies Co., (Newark Electronics, Inglewood, Calif.)
13. P.R. Mallory & Co. Inc., Indianapolis, Ind.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FS 324.7
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FS 307.2
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Key to figure 49-50-8

INDEX NO.	PART NO.	VENDOR	DESCRIPTION	QTY
1.	45905-1	1	Valve	2
2.	370-480002-71		Tube	1
3.	AN919-6C		Reducer	1
	MS 29512-06		"O" Ring	1
4.	4140-00-232	2	Pump, Fuel	1
5.	AN 815-6D		Union	1
	MS29512-06 "O" Ring		"O" Ring	
6.	AN833-4C		Elbow	1
	MS29512-04		"O" Ring	1
7.	AN832-4D		Union	2
	MS29512-04		"O" Ring	4
8.	AN938 D4		"T" Fitting	
	MS29512-04		"O" ring	4
9.	AN832-4D		Union	2
	MS29512-04		"O" Ring	3
10.	CCA-5800-3	3	Valve, Drain	1
	MS29512-04		"O" Ring	1
11.	AN919-6C		Reducer	1
	MS29512-04		"O" Ring	1
12.	370-480002-153		Tube	1
	(Attaching Parts)			
	MS21919DF6		Clamp	1
	MS21919DF8		Clamp	1
	MS27039-1-04		Screw	1
	MS21042L3		Nut	1
13.	AN815-6D		Union	1
14.	370-480002-157		Tube	1
	(Attaching Parts)			
	MS21919DF6		Clamp	1
	MS27039-1-04		Screw	1
	MS21D42L3		Nut	1
	2C29-8		Clip	1
15.	AN815-6D		Union	1

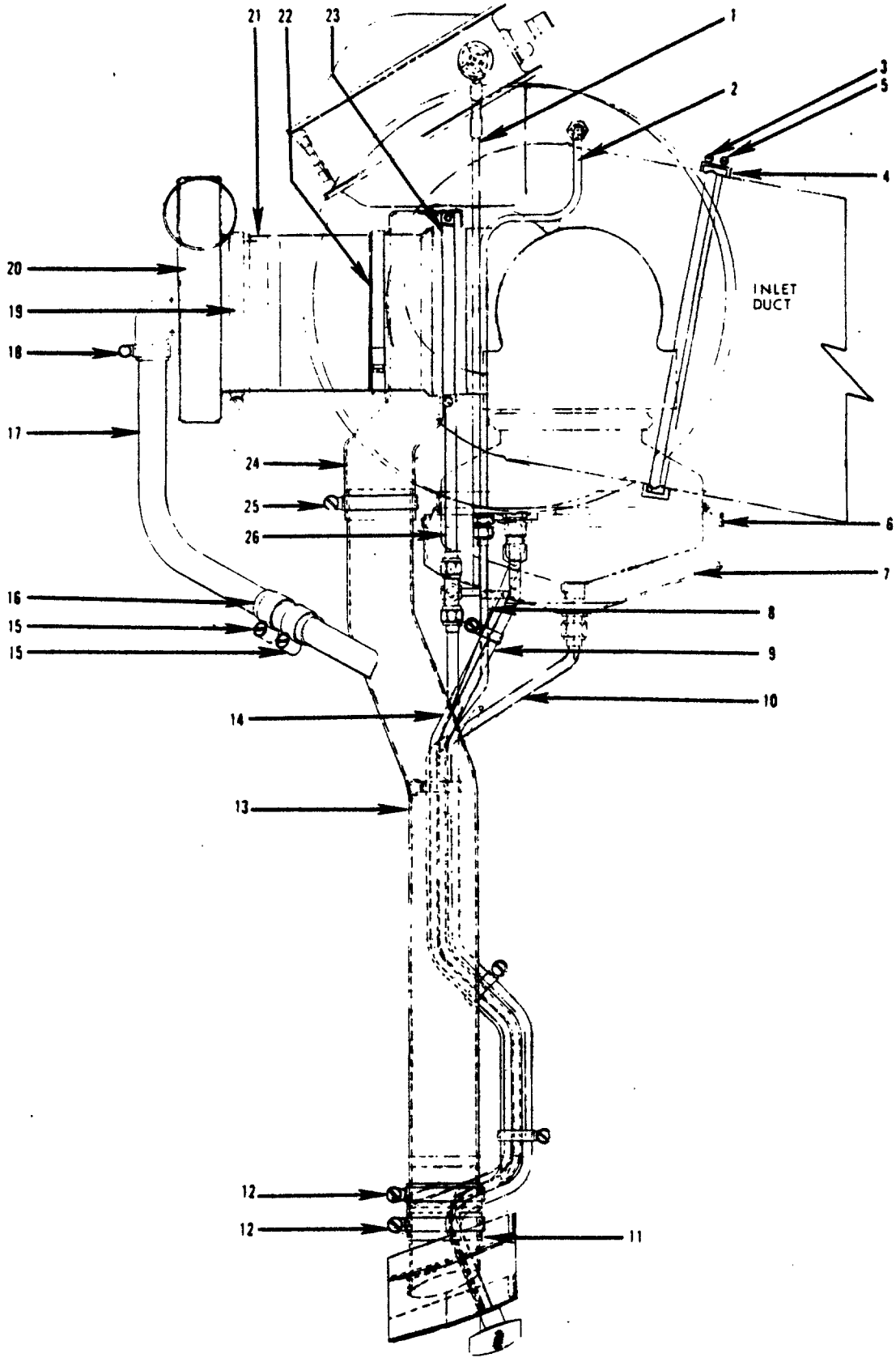
Key to figure 49-50-8

INDEX NO.	PART NO.	VENDOR	DESCRIPTION	QTY
16.	370-480002-75		Tube	1
	(Attaching Parts)			
	MS20819-6D		Nut	1
	LD172-0003-1001		Washer	1
17.	AN893-12D		Bushing	1
	AN833-6C		Elbow	1
	AN924-6D		Nut	1
	MS29512-08		"O" Ring	2
18.	28090-D	4	Filter	1
	AN6235-3A		Element Assembly	1
	AN6244-4		Ring - Backup	1
	AN6230-4		Gasket - "O" Ring	1
19.	AN893-12D		Bushing	1
	MS29512-08		"O" Ring	1
	MS29512-06		"O" Ring	1
	AN924-6D		Nut	1
20.	AN783-6D		Tee	1
	MS29512-06		"O" Ring	1
	AN893-2D		Bushing	1
	AN924-4D		Nut	1
	MS29512-04		"O" Ring	1
21.	1B2522-25	5	Switch	1
22.	370-480002-47		Tube	1
23.	AN815-6D		Union	1
24.	370-480002-161		Tube	1
25.	370-480002-11		Housing	1
	(Attaching Parts)			
	370-480002-3		Cup	1
	AN737TW46		Clamp	1
	AN931-6-10		Gromet	1
26.	3H2L-4-10		Hose	1
	AN737TW24		Clamp	2
27.	370-480002-25		Tube	1

Key to figure 49-50-8

INDEX NO.	PART NO.	VENDOR	DESCRIPTION	QTY
28.	370-480002-31		Flange	1
	(Attaching Parts)			
	370-480002-69		Tube	1
	AN837-6D		Elbow	1
	AN9246D		Nut	1
	MS29512-06		"O" Ring	1
	MS29513-227		Flange "O" Ring	1
29.	AN833-6C		Elbow	1
	AN29512-06		"O" Ring	1
	AN924-6D		Nut	1

1. Solar Aircraft, San Diego, California
2. Dukes Astronautics Co., Canoga Park, California
3. Curtis Automotive Devices Inc., Westfield, Ind.
4. Bendix Co., Madison Heights, Mich.
5. Century Electronics Inc., Tulsa, Okla.



Parts List
Figure 49-50-9

Key to figure 49-50-9

INDEX NO.	PART NO.	VENDOR	DESCRIPTION	QTY
1.	370-480002-67		Tube, Drain	1
2.	306-870009-281		Tube, Vent	1
3.	MS35842-15		Clamp	2
4.	380-533390-1	1	Seal, Inlet	1
5.	MS35842-16		Clamp	2
6.	380-953391		Strap	1
7.	380-953390		Strap	1
8.	306-870010-321		Pan, Drain	1
9.	306-870009-281		Tube, Drain	1
10.	306-870009-273		Tube, Drain	1
11.	306-870009-309		Tube, Drain	1
12.	21A04-250-12	1	Hose	1
13.	MS35842-13		Clamp	2
14.	306-870009-301		Tube	1
15.	306-870009-275		Tube, Drain	1
16.	MS35842-11		Clamp	2
17.	21A04-100-12	1	Hose	1
18.	306-870009-205		Tube	1
19.	MS35342-11		Clamp	1
20.	MS35342-16		Clamp	1
21.	10A124	1	Shroud, Gen.	1
22.	47595-1	2	Generator, DC	1
	(Attaching Parts)			
	M221F300050711	3	Thermostat	1
	21563 Bracket			1

Key to figure 49-50-9

INDEX NO.	PART NO.	VENDOR	DESCRIPTION	QTY
23.	MS35342-16		Clamp, Shroud	1
24.	10A361	2	Clamp, Shroud	1
25.	306-870010-249		Shroud, Gen.	1
26.	MS35842-13		Clamp	1
27.	306-870010-277		Tube, Drain	1

1. General Connectors Corp., Burbank, Calif.
2. General Electric, Erie, Pa.
3. Texas Instruments, Attleboro, Mass.

CHAPTER 51

STANDARD PRACTICES-STRUCTURES

CHAPTER 51

STRUCTURES GENERAL

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	2	Mar 1/70

* The asterisk indicates pages revised, added, or deleted by the current revision.

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STRUCTURES GENERAL

1. Description

The North American Rockwell Corp Sabreliner is an all-metal airplane with full-contilever wings and horizontal stabilizer. The fuselage is of semi-monocoque construction and supports an engine nacelle on each side in the area above and aft of the wing.

A. Doors (Refer to Chapter 52-00.)

The airplane is equipped with a passenger/crew entrance door that, when opened, provides a stairway into the airplane, and two emergency ground escape hatches. Both entrance and escape doors may be opened from inside or outside the airplane. The escape hatches open inboard and completely detach from the airplane. The doors are sealed to maintain cabin pressurization. The entrance door seal is inflatable, using engine compressor air. A warning system is provided for the entrance door to indicate when the door is not closed or locked. Numerous access doors are installed in the exterior surfaces of the fuselage, wings, nacelles, and empennage, for ease of service and maintenance.

B. Fuselage (Refer to Chapter 53-00.)

The fuselage is semimonocoque in construction and consists of bulkheads, frames, and stringers, with a Chemi-Milled skin covering.

C. Nacelles (Refer to Chapter 54-00.)

The nacelles are pod-type and are mounted on pylons on the fuselage, just aft of the wing. Removable access panels provide for each of access to all components of the power plant.

D. Stabilizers (Refer to Chapter 55-00.)

The stabilizers consist of the vertical fin and left and right horizontal stabilizers. Provisions are made for installation of inflatable deicer boots on stabilizer leading edges.

E. Windows (Refer to Chapter 56-00.)

The cockpit windows comprise nine fixed windshield and window panels, and one sliding window on the pilot's side. Two of the windows are directly overhead of the pilot and copilot, providing optimum forward and overhead vision. The windshield, overhead windows, and direct-vision windows are electrically heated to provide anti-icing. Passenger compartment windows are of double-paneled construction. All windows, with the exception of the windshield, direct-vision windows, and forward overhead windows, are provided with a dehydrator system.

F. Wing (Refer to Chapter 57.)

The full-cantilever wing has a span of 44 feet 9 inches and is of box-type construction, using two spars and tapered machine-milled upper and lower skins. The front and rear spars, along with the wing skins and ribs, form four fuel bays in each wing. Each wing contains a 449-gallon fuel cell. The leading edge of the wing, permanently attached to the wing box section, contains the slats. Panels are provided in the leading edge for access to the slat tracks. A skid is attached to the bottom of the lower splice joint to protect the wing in case of a wheels-up landing. Provisions are made for installation of wing deicing boots on the wing slats and lower wing leading edges.

CHAPTER 52

DOORS

CHAPTER 52

DOORS

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CHAPTER 52

DOORS

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DOORS

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52-80-00	1	Jun 28/74

* The asterisk indicates pages revised, added, or deleted by the current revision.

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DOORS - DESCRIPTION

1. General (See figure 52-00-1.)

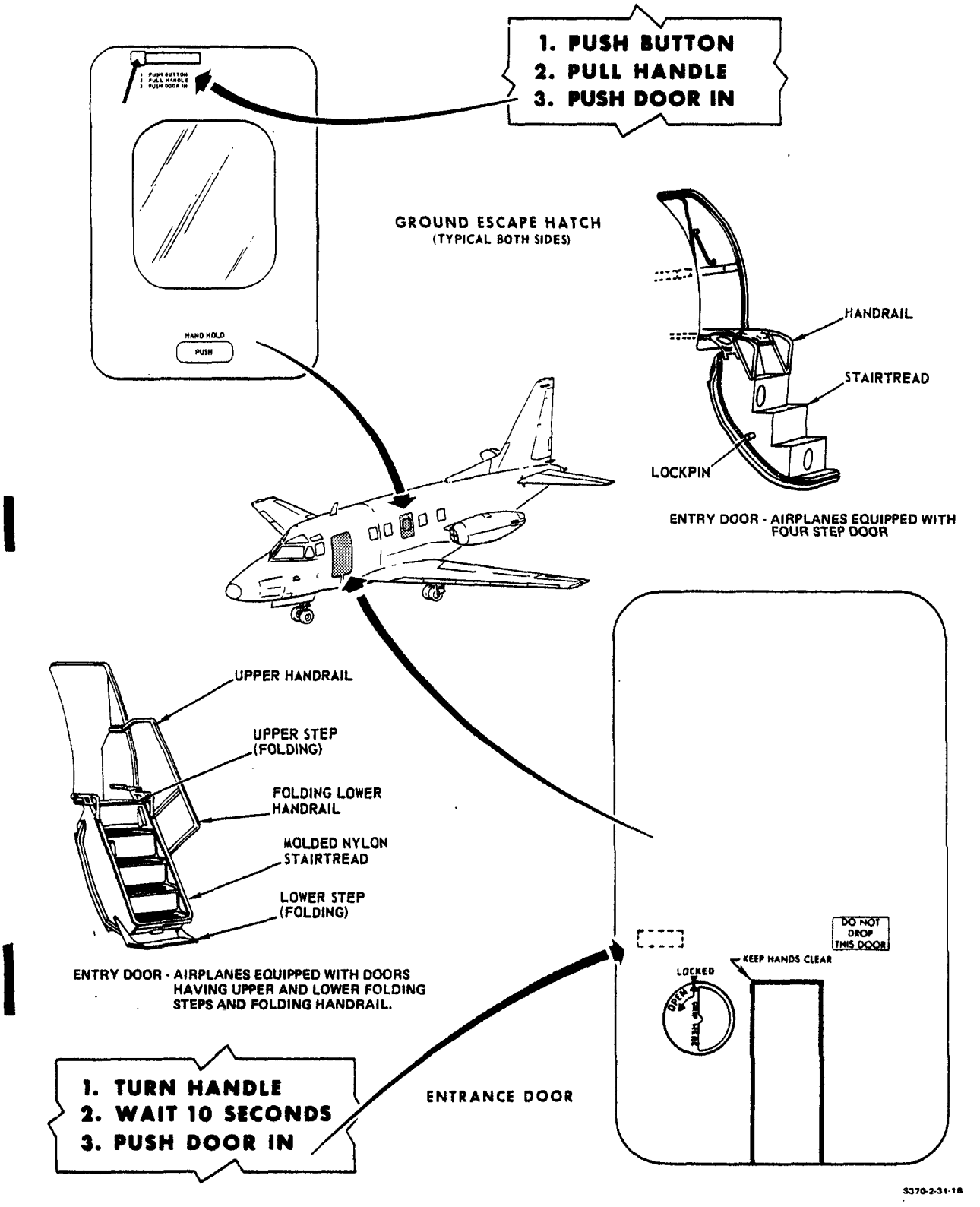
The passenger/crew entrance door is on the left side of the fuselage, just aft of the crew compartment. When opened, the door provides an entrance stairway into the airplane.

There are two ground emergency exit doors located approximately midway in the passenger compartment, one on the right side and one on the left side of the airplane.

A door warning system is provided for the entrance door. If the door is not properly closed and locked, the door warning light on the instrument panel will illuminate.

The main landing gear doors consist of an upper and lower strut fairing door for each landing gear strut. The upper strut fairing door is hinged from the outboard end of the landing gear strut well in the wing lower surface and attached to the landing gear strut by an adjustable link. The lower strut fairing door is hinged from the lower edge of the upper strut fairing door and is actuated by mechanical linkage connected between the upper and lower strut doors.

Two nose gear fairing doors attach to the fuselage, one on each side of the nose gear wheel well, and completely cover the nose gear wheel well upon retraction of the nose gear.



Entrance Door and Emergency Ground Escape Hatch
Figure 52-00-1

PASSENGER/CREW - DESCRIPTION AND OPERATION

1. General

The passenger/crew entrance door, on the left side of the fuselage between the cabin and the cockpit, consists essentially of a combination door and stairway structure. On NA-370-7 and NA-380-1 and later airplanes, either one of two types of entry doors are installed; a four step door of all aluminum construction, or a four step door with upper and lower folding steps, a folding handrail and molded nylon stair treads. The operation and opening and closing sequence for both doors is essentially the same and described in the following paragraphs.

The door is locked in the closed position by two sliding lockpins, which are mechanically linked to the manually operated internal and external lock handles. When the door is closed and locked, the aft locking pin actuates a switch which controls pressurization and depressurization of the inflatable door seal and extinguishes the pilot's door-open warning light. A key-operated lock is just forward of the external door lock handle.

The door is opened and closed manually with spring assist from two bungees. The inflatable door seal is supplied with bleed air from the engine compressors. A door seal pressure reservoir is provided to maintain seal inflation in case loss of source air pressure occurs. The seal is inflated when the seal pressure regulator valve is actuated by the rear locking pin of the door lock mechanism. When the seal is inflated, the serrated edge of the seal is forced against the door land, forming a tight seal.

A manually operated dump valve installed on the bulkhead just aft of the entrance door relieves door seal pressure in case the seal regulator fails to move to the vent position when de-energized on the ground.

A. To open door, proceed as follows:

(1) Unlock door, if locked.

(2) Turn outer door handle one-quarter turn counterclockwise to unlatch, and wait 10 seconds.

(3) Apply an even pressure on lower section of door. The door will move inward several inches, and then the top of the door will rotate outward and start a down swing.

(4) Support door as it rotates outward, and lower door to extend position.

(5) On airplanes having the entry door with folding steps and handrail, raise the handrails on the aft side of the door to actuate and unfold the upper and lower steps and rotate the upper handrail 90° and attach to the receptacle on fuselage bulkhead at station 174.

CAUTION: DO NOT ALLOW DOOR TO DROP, AS STRUCTURAL DAMAGE CAN RESULT.

WARNING: KEEP HANDS CLEAR OF SCISSOR LINKAGE.

ENTRANCE DOOR

MAINTENANCE PRACTICES

REMOVING ENTRANCE DOOR

1. Unlock and fully open door.
2. Remove attaching bolts and spacers from lower door cover.
3. With an assistant to support door, carefully remove bolts and washers from lower rod ends of door center link.

CAUTION

Secure door support arms when removing bolts, to prevent sudden release of spring tension from door assist bungees.

5. Place door on padded area to prevent damage.
6. To remove forward door assist bungee, remove trim panels on each side of bulkhead at FS 143. Remove bolt from bungee rod end, and remove clevis attachment and spacers from bungee. Lift out bungee.
7. To remove aft door assist bungee, remove trim panels from bulkhead at FS 174, and repeat removal procedure in step 6.

WARNING

To prevent serious injury to personnel, do not disassemble door bungees without the required special tools.

ADJUSTING DOOR SEAL MICROSWITCH

With entrance door closed and locked and with door locking pin contacting the switch, adjust limit switch to allow 1/16-inch overtravel before switch is bottomed.

INSPECTION

1. Inspect entrance door for cleanness, skin and structure damage, and loose or missing rivets, and fairing door for damage.
2. Check door seals for security of attachment, and for cuts, tears, and deterioration. Inspect door for alignment, free movement, and positive locking.

INSTALLING ENTRANCE DOOR

NOTE

On NA 265-70 and Subs and NA 265-80 airplanes, two different types of entry doors are installed. The following installation procedures are applicable to either type of entry door.

CAUTION

Do not allow door to scrape fuselage structure as damage to the noninflatable door seal could result in possible leakage of cabin pressure during flight.

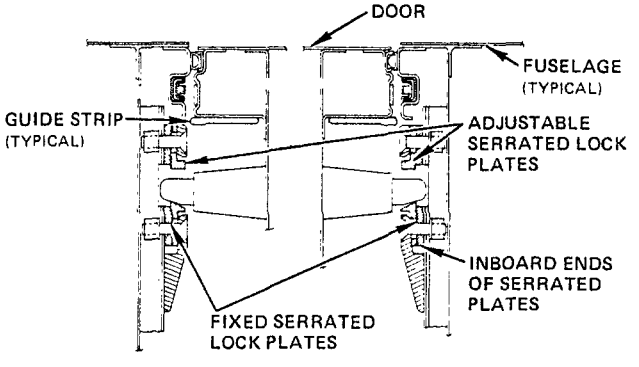
NOTE

If a new door is being installed, the door skin should be trimmed only the amount necessary to allow the door to fit snug into the opening. The door skin can be trimmed later, when the door is being adjusted, until it is of the proper skin gap.

1. Before installing door, check that lockpins extend 31/32 (±1/16) inch from lockpin support fitting in door-locked position and that pins are flush with, or below, support fitting in door-unlocked position. Adjust vertical and/or horizontal lockpin rod ends as necessary to obtain proper extension of both lockpins.

NOTE

Gain access to door lock mechanism by removing screws from stair plate. Lift stair plate from stair frame structure.



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MAINTENANCE PRACTICES (Continued)

INSTALLING ENTRANCE DOOR (Cont'd)

2. Loosen attaching screws and adjust serrated lock plates on door so that adjustable plate inboard edge is flush with fixed plate inboard edge. Tighten screws.

NOTE

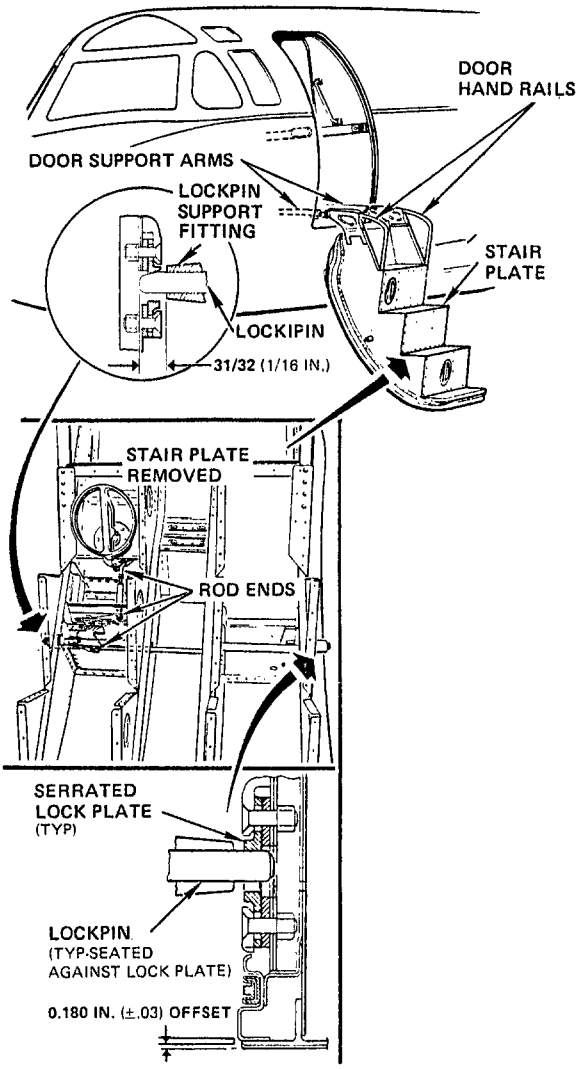
The secondary pressure seal rubs against the fuselage structure as the door opens and closes and can be subjected to excessive compression which may tear the seal on an improperly rigged door. If a replacement door is being installed on the airplane, the secondary seal should be removed until the door is rigged. Guides are installed on both sides of the door and are used to prevent the seals from being excessively compressed (when adjusted properly) when the door is opened or closed.

3. Position door in open position in door opening and attach door support arms to door handrails. Install attach bolts with boltheads toward door centerline and install nuts fingertight. This is to allow partial removal of bolts with the door closed. Do not install washers at this time.

WARNING

The door support arms are connected to spring bungees. Inside of the fuselage structure, which, if installed must be compressed to connect support arms to door handrails. The bungees are strong enough to injure personnel if arms are released while under tension.

4. Carefully close and lock door. If necessary, apply pressure to mold line surface of door at lockpin locations to ensure lockpins seat properly against inboard surfaces of cutouts in adjustable serrated lock plates.



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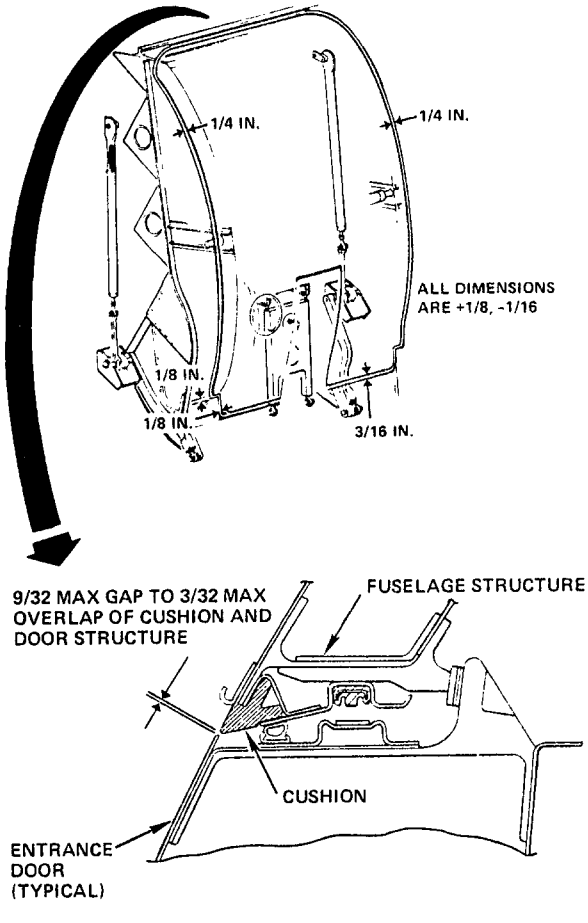
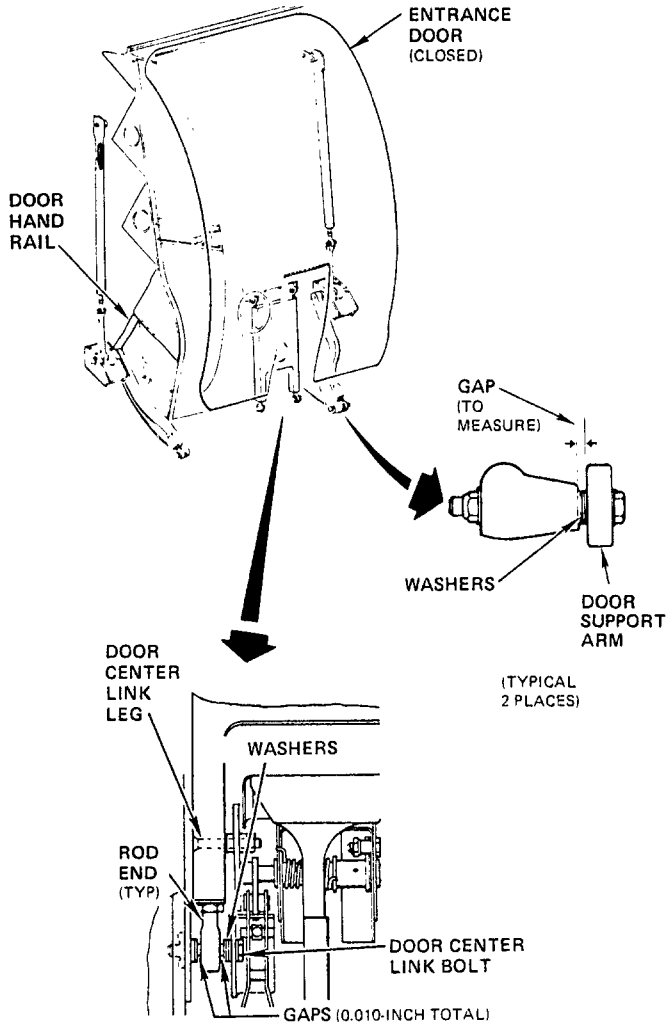
MAINTENANCE PRACTICES (Continued)

INSTALLING ENTRANCE DOOR (Cont'd)

5. Check that door mold line is 0.180 (±.03) inch inside of fuselage mold line at lockpin locations. Carefully open door and adjust serrated lock plates as necessary to obtain 0.180 (±.03) inch offset. Mark location of inboard ends of adjustable lock plates when adjustment is complete. This mark is used for reference in final adjustments.
6. Carefully close and lock door, making sure that lockpins are seated as in step 4.
7. Partially remove bolts and disconnect door support arms for door handrails and, using tapered pieces of wood or plastic, equalize gap between forward and aft edges of door and fuselage structure.
8. Loosen door center link lower rod-end jam nuts and adjust rod ends so that attach bolts have a slip fit; then tighten jam nuts. Install attach bolts temporarily.
9. Measure gap on each side of rod ends and install washers as required to obtain 0.010-inch total gap. It may be necessary to machine the surface of one washer to obtain the required tolerance. Tighten attach bolts.
10. Loosen door support arm rod-end jam nuts and adjust rod ends so that door-support-arm-to-handrail attach bolts have a slip fit; then tighten jam nuts. Temporarily install attach bolts and nuts.
11. Measure and record gap between door support arm rod ends and door handrails. Remove wood or plastic wedges and carefully open door.
12. Remove door-support-arm-to-handrail attach bolts and reinstall bolts with boltheads away from door centerline. Use washers as necessary to remove any gap noted in step 11.





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MAINTENANCE PRACTICES (Continued)

INSTALLING ENTRANCE DOOR (Cont'd)

NOTE

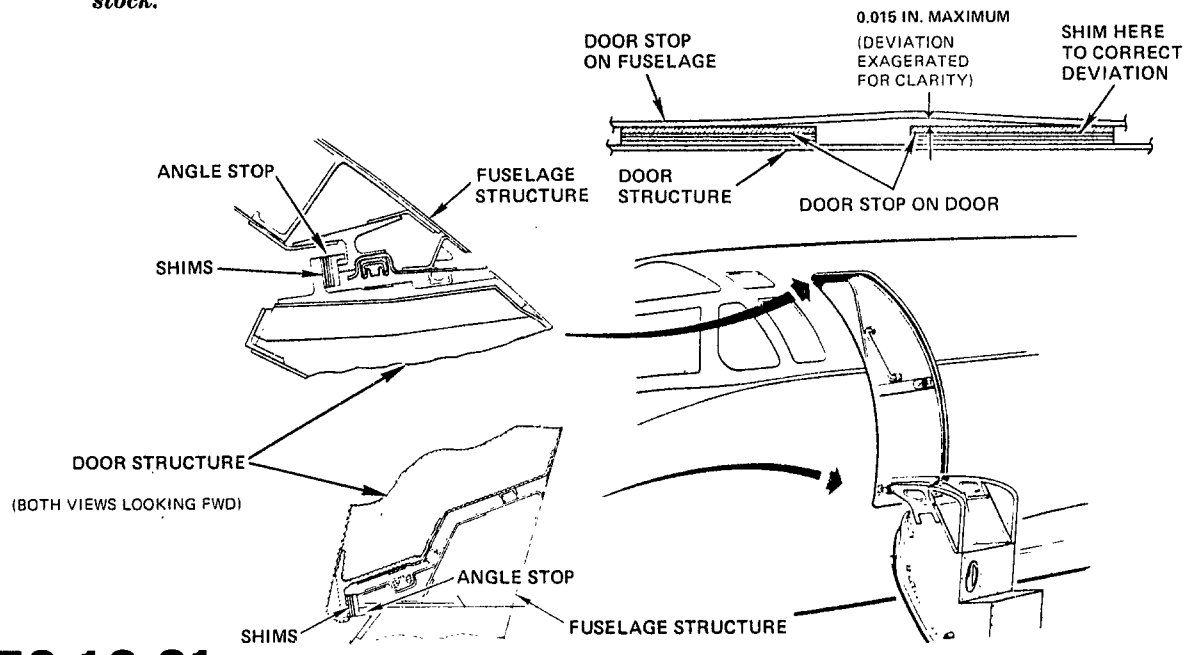
Four washers are installed at each support-arm-to-handrail attach point. Washers not used to fill gap must be installed under nut or bolthead.

13. Install nuts on attach bolts and tighten. Actuate door from full open to full closed. If door rubs excessively at forward or aft edges of opening, it may be necessary to adjust the gaps in step 11 to obtain a satisfactory fit and actuation of the door.
14. With door open, check bearing surfaces of lower doorstop on fuselage with a straightedge. If surface is convex or concave, note amount and location of any deviation exceeding 0.015 inch.
15. With door open check bearing surfaces of the two angle stops at the bottom of the door with straightedge (the two surfaces are to be considered one continuous surface). If surface is convex or concave, note amount and location of any deviation exceeding 0.015 inch.
16. Compare measurements in steps 14 and 15. The stops must coincide with the respective doorstop. If surfaces do not coincide, the angle stops must be shimmed until surfaces coincide within 0.015 inch.

NOTES

- ***Angle stop on door must be shimmed so that surface is flat or convex, never concave.***
- ***Shims are to be made from 0.040-inch thick 2024-T3 or -T2 aluminum sheet stock.***

17. Carefully close and lock door, making sure that lockpins are properly seated against inboard surfaces of cutouts of adjustable serrated lock plates as in step 4. Check and record offset of door mold line with fuselage mold line at the two lower corners of the door and at the upper centerline of the door.
18. With door closed check gap between forward and aft ends of upper doorstop on fuselage and angle stop on door. This gap must be the same as the offset recorded in step 17, within 1/16 inch, and must be the same at both ends within 1/32 inch. Shim door angle stop as necessary with tapered shims to obtain proper gaps.
19. Repeat step 18 for the lower fuselage doorstop and the two stop plates on the door. The gap on the extreme forward and aft ends must be the same as the offset recorded in step 17, within 1/16 inch and must be the same at both ends within 1/32 inch. Shim door angle stops as required to obtain proper gaps.
20. After stop adjustments are complete, close and lock door. If door lockpins are difficult to engage, move adjustable serrated lock plates one serration inboard and recheck. Repeat adjustment, one serration at a time untill lockpins engage easily, but do not move adjustable serrated plates more than 1/16 inch from position marked in step 5.
21. Check that door does not rub inflatable seal at bottom of door. If door is rubbing seal, close and lock door; then remove door center-link lower attach bolts, noting number and position of spacer washers, loosen center-link lower rod-end jam nuts, and lengthen rod ends one-half turn. Tighten jam nuts and reinstall attach bolts and spacer washers. Adjust outboard ends of door arms in a like manner as required. Reinstall rigidized stair plate on door.



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MAINTENANCE PRACTICES (Continued)

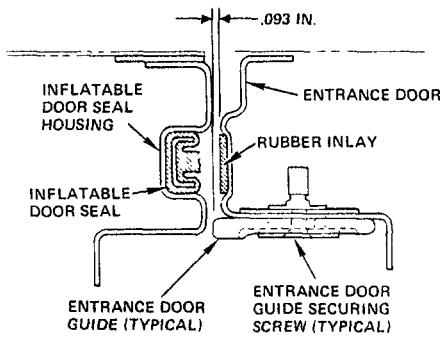
INSTALLING ENTRANCE DOOR (Cont'd)

22. Install secondary seal (if removed). Refer to Replacing Inflatable Door Seal.

23. Open entrance door and loosen guide(s) securing screws. Adjust guides to extend past the rubber inlay for the inflatable seal by .093 inch. Secure guide screws, then open and close door to ensure that secondary seal does not excessively compress. Readjust guides as required.

NOTE

The .093-inch clearance may be adjusted as required to prevent the secondary seal from being excessively compressed. This will prevent possible damage to the secondary seal.

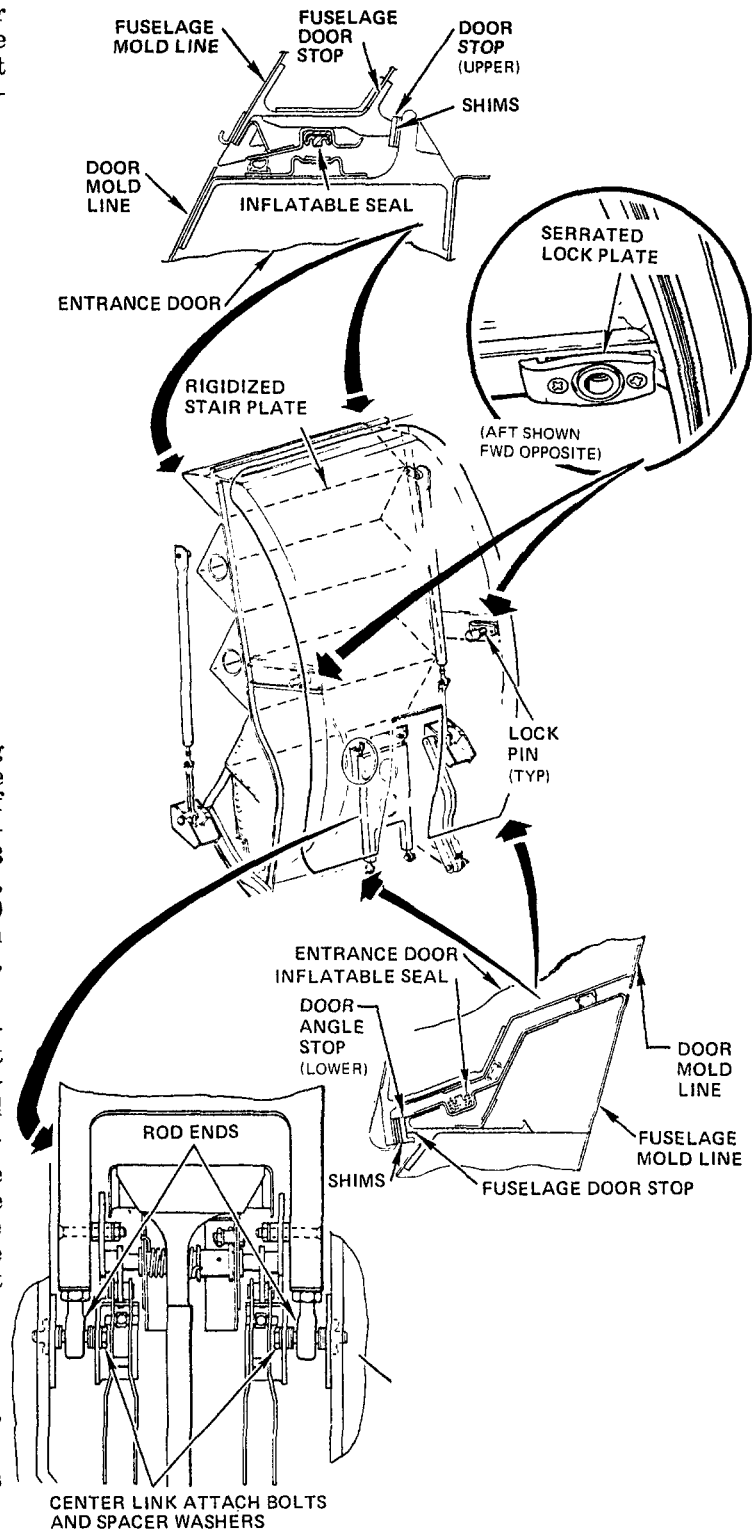


24. With door open, install a strip of masking tape along entire bearing surface of upper and lower angle stops on door. Close door and check for full contact of doorstop bearing surfaces. If unable to close door sufficiently to make stops contact, pressurize cabin to 3 psi (5 psi maximum) on the ground (refer to 20-30-3, Pressurization Control System Check-out Procedure) or in flight. If full contact is not obtained with cabin pressurized, shim upper and lower angle stops, using tapered shims until full contact is obtained.

25. With cabin pressurized to 5 psi, check lockpin clearance by operating door locking mechanism to retract the pins. If more than normal resistance is experienced, depressurize the cabin and move serrated plates inboard until normal pin operation is obtained.

26. With cabin pressurized to 5 psi the mold line of the outside door skin should be within 1/16 inch of the fuselage skin mold line in the center one-third of the top of the door and along the door periphery from the latch pins down. The upper forward and upper aft corners of the door should not exceed (±) 1/8 inch.

27. Install bungees, if removed, using same parts as were removed. Close entrance door; then adjust bungee rod end so that bungee is extended 1/32 inch minimum [1] ([2] 1/32 inch min. 1/16 inch max.) against spring, and attach bungee to attach link.



[1] S/N 370-1 and Subs and 380-1 thru -26, -28 thru -33, -35, and -37 not changed by S/B 74-27.

[2] S/N 380-27, -34, -36, -38 and Subs and prior airplanes changed by S/B 74-27.

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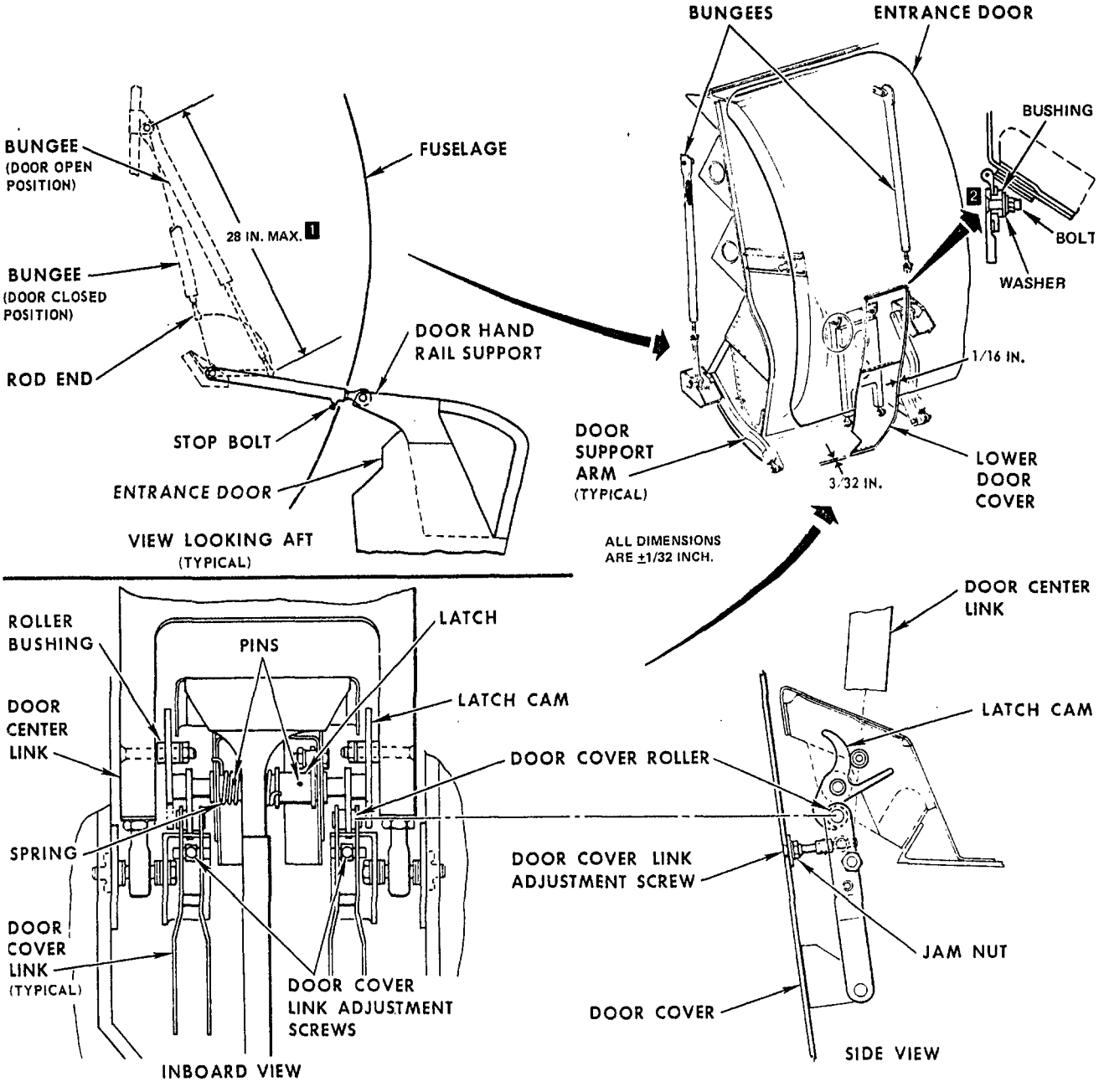
MAINTENANCE PRACTICES (Continued)



1 S/N 370-1 and Subs and 380-1 thru -26, -28 thru -33, -35, and -37 not changed by S/B 74-27.

2 S/N 380-27, -34, -36, -38 and Subs and prior airplanes changed by S/B 74-27.

Figure 51-10-201 — Installing Entrance Door

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MAINTENANCE PRACTICES (Continued)

INSTALLING ENTRANCE DOOR (Cont'd)

28. Install roller bushings on each leg of door center link.
29. When installing a new lower door cover, trim and fit it as necessary, so that it matches the mold line and so that it provides the proper skin gap.
30. Loosen jam nuts on door cover link adjustment screws.
31. Close entry door and make sure door cover links and rollers will clear when latching cam is approaching the overcenter closed position.
32. With cabin pressurized to 5 psi, check that door covers properly fit and fair (within ±.04 inch) around periphery of cover. The edges of the cover may be crimped, if required. Adjust both screws in door cover links until both the center link rollers and cover link rollers contact latch hooks. Latches can be inspected through windows provided for that purpose.
33. Open door and tighten jam nuts on door cover link adjusting screws. With door open, adjust stop bolts on door support arm so that both stop bolts contact stops at the same time. A minimum of six threads should be engaged on each stop bolt.
34. Adjust inflatable seal microswitch.

INFLATABLE DOOR SEALS

NOTES

- ***If door is to be removed, refer to Removing Entrance Door and observe caution in procedure.***
- ***The trim panel aft of the door opening must be removed so that a new inflatable seal can be installed.***
- ***RTV-732 (Dow Corning) is an approved substitute for RTV-108 (GE) for installing the inflatable door seal.***

PRIMARY DOOR SEAL

1. Remove damaged inflatable seal from entrance door.
2. Remove all paint, adhesives, grease, dirt and foreign material from seal channel in door.
3. Using a clean cloth moistened with toluene, thoroughly clean seal channel in door. Before toluene evaporates, thoroughly dry surfaces with a clean dry cloth. If toluene is allowed to dry on the surfaces, the cleaning process is not satisfactory and the surface must be recleaned and dried as directed.

NOTES

- ***Refer to Figure 52-10-202 for proper procedure for application of sealant. Excessive sealant will not permit door seal to seat properly, resulting in a tight fitting door.***
- ***Squeeze a test strip or bead of sealant to achieve the appropriate pressure to apply a 1/8" bead.***

4. Apply a 3/16 bead of RTV-108 or 732 Silicone Rubber adhesive/sealant to bottom center of seal channel completely around periphery of door.
5. Position new inflatable seal in position in channel with the tube on the side of the seal extending through hole in the channel. Make certain that there is sufficient adhesive/sealant around tube base to ensure a proper seal.
6. Connect seal tube to air conditioning system line. Care should be exercised not to stretch the seal during installation.

NOTE

The manufacturers recommended cure time is 24 hours; however, if airplane operation is absolutely necessary, the seal may be cemented into place in the seal retainer and the main entrance door closed and the seal inflated to 25±4 psi. The seal must maintain the 25±4 psi pressure for a minimum of 1-1/2 hours and a minimum relative humidity of 20% must be present. After the minimum 1-1/2 hours, the seal may be deflated, the main entrance door opened, and the aircraft flown with no flight restrictions.

7. Using masking tape, tape seal into channel to secure in place until adhesive/sealant has set up. The adhesive/sealant will cure "tack free" in about 30 minutes, and should be fully cured and ready for flight in 24 hours.

SECONDARY DOOR SEAL

1. Remove damaged secondary seal from seal retaining channel around periphery of entrance door.
2. Check seal channel on door to make certain channel is free of dirt, oil, and grease and is undamaged.
3. Insert new secondary seal tube into hole and guide tube through panel inside of door. (The stair tread must be removed so that tube may be guided through inner structure.) Apply a bead of RTV-108 sealant around seal tube grommet.

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MAINTENANCE PRACTICES (Continued)

SECONDARY DOOR SEAL (Cont'd)

4. Firmly press seal into channel. Make certain seal is correctly installed in channel around complete periphery of door.
5. With seal installed, close door, forcibly if necessary. Repeat several times to seat seal in retainer and to flex seal walls, until the door can be closed by one man without forcing.

NOTE

Greasy lubricants should be avoided as they tend to accumulate dirt.

6. Check that seal, and seal bearing surfaces are clean and coated with a dry lubricant such as Fabriglide or S122 Fluorocarbon.

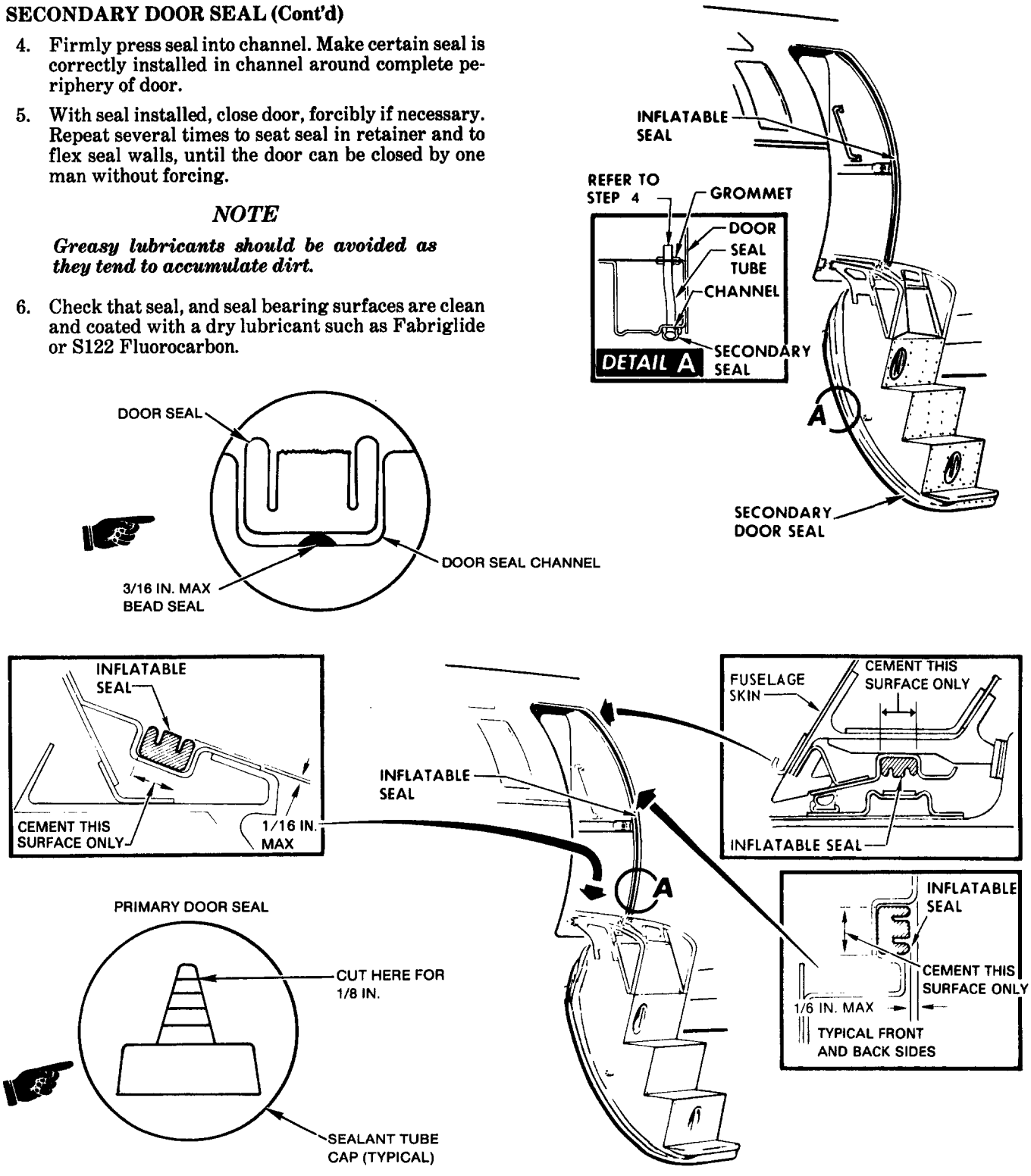

Figure 52-10-202 — Door Seal Replacement

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MAINTENANCE PRACTICES (Continued)

CHECK-OUT OF MAIN DOOR SEAL

1. Connect an air-pressure source and a gauge to door seal ground pressure fitting on cabin ground pressure panel inside aft section equipment compartment.
2. Close and latch main cabin door. Check that DOOR OPEN caution light is off.
3. Pressurize seal pressure lines to 28 ±1 psig by applying pressure at test fitting. Allow 2 minutes for air pressure to stabilize.

WARNING

Do not allow air pressure in lines to exceed 29 psi during this check.

4. Close inlet air pressure valve. Ensure that air pressure does not decrease below 21 psig after a 20-minute period.
5. Disconnect test unit and cap test fitting.

REPLACEMENT OF ENTRANCE DOOR BUNGEE HYDRAULIC SEALS - S/N 380-1 THRU -4, -6 THRU -9, -11 THRU -13, -15, -17, -19, -21, -23, -25, -30 AND -32 NOT CHANGED BY S/B 74-27

Two bungees are installed in the airplane. One is installed on the forward side of FS 143 and the other is installed on the aft side of FS 174. To disassemble the bungee(s), proceed as follows:

WARNING

These bungees (when assembled) are under spring tension which if disassembled without the proper equipment may cause serious injury to personnel.

1. Remove trim panels as required to gain access to bungee being removed.
2. Close main entry door; then remove the lower bungee rod end to bellcrank bolt on applicable bungee.

CAUTION

With one or both bungees removed, ensure that door is supported from the outside prior to opening to prevent damage to door or structure.

3. Open and slowly lower door until it is fully open.
4. Remove the top pin from bungee and remove bungee from airplane. Retain pin, bushings and washer for reinstallation.
5. Remove servicing plug from bungee and drain hydraulic fluid.
6. Loosen rod end lock nut; then remove piston rod end.
7. Remove the three cotter pins from bottom end of cylinder.

WARNING

Do not unscrew lower end fitting more than 3/16" prior to installing tool. This is required so that end fitting will engage wrenching boss.

8. Unscrew lower end fitting not more than 3/16".
9. Using a locally manufactured tool (Figure 52-10-205), screw shaft extension in bungee piston rod.
10. Place wrenching boss over shaft extension so that boss engages the ears on bungee lower end fitting.
11. Screw shaft with wing nut into shaft extension; then screw wing nut down against wrenching boss.

WARNING

To prevent possible injury to personnel, ensure that locally manufactured tool is properly installed prior to end fitting removal.

WARNING

Do not allow shaft or shaft extension to rotate when turning wrenching boss and wing nut.

12. Loosen wrenching boss and wing nut simultaneously until the lower end fitting is disengaged from bungee cylinder. Do not allow shaft or shaft extension to rotate. Continue to loosen wing nut until all spring tension is removed.
13. Remove tool from bungee piston rod; then remove springs, piston rod, and spring retainer from cylinder.
14. Remove the three cotter pins from upper end of cylinder and then remove upper cylinder end. Remove piston rod nut and washer; then remove piston and floating piston.
15. Remove and discard O rings. Clean all components using Stoddard solvent (or equivalent), then inspect components for scoring, pits and corrosion. Replace components as required.

To Assemble Bungee, Proceed as Follows:

NOTE

During assembly, ensure that the back-up packings are installed on the proper side of the O rings (Figure 52-10-204). Lubricate all O rings with hydraulic fluid MIL-H-5606.

1. Install O rings and back-up rings, if removed, on the floating piston and piston. Install floating piston and piston on piston rod; then install washer and nut. Torque nut to 160-190 in.-lbs.

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MAINTENANCE PRACTICES (Continued)

To Assemble Bungee, Proceed as Follows: (Cont'd)

2. Install servicing plug and O ring in upper bungee fitting. Tighten fitting. Install upper fitting in cylinder; then install the three cotter pins.
3. Install spring retainer on piston rod; then install piston rod assembly and springs in cylinder.
4. Install O rings on the bungee lower end fitting.
5. Fill cylinder with hydraulic fluid MIL-H-5606 to within 1-7/16 inches from the lower end of the cylinder.
6. Install tool shaft extension in piston rod. Slide bungee lower end fitting over tool extension and down against spring.

WARNING

Ensure that locally manufactured tool is properly installed prior to compressing springs to prevent possible injury to personnel.

WARNING

Do not allow tool shaft extension to rotate when compressing springs.

7. Assemble tool over lower end of fitting; then compress springs. When lower end fitting enters the cylinder threads, screw end fitting into cylinder as far as possible.
8. Remove tool from bungee piston rod. Screw end fitting into cylinder until ears are flush with end of cylinder. Install cotter pins; then, install rod end. Adjust rod end to obtain 20 inches between hole centers of bungee mounting points. Tighten rod end jam nut.
9. Remove servicing plug from bungee and completely fill bungee with hydraulic fluid, MIL-H-5606. Install and tighten servicing plug.
10. Place bungee in mounting position and install pin, bushings and washer. Install cotter pin.
11. Close entrance door and connect bungee rod end to bellcrank. Install cotter pin.
12. Open entrance door and check for smooth continuous lowering of the door. Check bungee for leaks.
13. Install trim panels as required.

REPLACEMENT OF ENTRANCE DOOR BUNGEE HYDRAULIC SEALS - S/N 370-1, -7, S/N 380-1 THRU -4, -6 THRU -9, -11 THRU -13, -15, -17, 19, -21, -23, -25, -30, -32, AIRPLANES CHANGED BY S/B 74-27, AND S/N 380-27, -34, -36, -38 and SUBS

Two bungees are installed in the airplane. One is installed on the forward side of FS 143 and the other is installed on the aft side of FS 174. To disassemble the bungee(s), proceed as follows:

1. Remove trim panels as required to gain access to bungee being removed.
2. Close main entry door; then remove the lower bungee rod end to ballcrank bolt on applicable bungee.

CAUTION

With one or both bungees removed, ensure that door is supported from the outside prior to opening to prevent damage to door or structure.

3. Open and slowly lower door until it is fully open.
4. Remove the top bolt from bungee and remove bungee from airplane. Retain bolt and bushings for reinstallation.
5. Remove rod end, lock and jam nut from lower end of piston rod. (Figure 52-10-202A.)
6. Remove valve from upper cylinder end. The valve is accessible through hole in cap. Drain hydraulic fluid from bungee.
7. Remove lockwire from upper cylinder end jam nut; then loosen jam nut.
8. Remove upper cylinder end from cylinder; then remove piston rod, springs, slides and lower gland from piston rod.
9. Remove pin from upper gland; then remove, unscrew, upper gland from inner piston rod.
10. Remove orifice from piston rod using spanner wrench (Figure 52-10-203).
11. Remove orifice and inner piston rod from piston rod.
12. Remove and discard O rings and seals. Do not remove retainers from slides. Clean all components using Stoddard solvent, (or equivalent, then inspect components for scoring, pits, and corrosion. Replace components as required.

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MAINTENANCE PRACTICES (Continued)

To Assemble Bungee, Proceed as Follows:

NOTE

During assembly of the bungee, special tools are required to install the seals. These special tools are depicted in Figure 52-10-203. Lubricate all seals with MIL-H-5606 hydraulic fluid. To assemble bungee, proceed as follows:

1. Remove spring from inner piston seal; then place inner piston seal and inner piston rod, large end, in boiling water. Allow sufficient time to thoroughly heat both items.
2. Place installation tool A over end of inner piston rod. Place inner piston seal, open end toward top of piston, see detail A, Figure 52-10-202A, in groove of piston. Install spring in seal; then slide tool B over seal and allow to remain long enough to size the seal to its original dimension.
3. Place piston seal and piston in boiling water. Allow sufficient time to thoroughly heat both items.
4. Place installation tool C over end of outer piston. Install outer piston seal, open end toward bottom of piston, see detail B, Figure 52-10-202A, in groove of piston.
5. Slide tool D over outer piston seal and allow to remain long enough to size the seal to its original dimension.
6. Using dry air or nitrogen, thoroughly dry both inner piston rod and piston rod.
7. Install scraper, lip down, in lower gland. Size scraper by inserting tool E in scraper.
8. Install O ring seal in upper gland groove.
9. Using tool F, install inner piston rod in piston rod.
10. Install orifice over inner piston rod and screw into piston. Tighten orifice with spanner wrench until orifice is bottomed.
11. Install upper gland on inner piston rod and install pin.
12. Install lower spring and slide, center spring, upper slide and spring on piston rod; then install lower gland nut, seal end towards bottom of cylinder, over piston rod.
13. Place tool G on end of piston rod and install assembled piston rod assembly in cylinder; then install upper end of fitting on cylinder. Screw upper end fitting on cylinder until it is bottomed. Torque jam nut to 40-60 in.-lbs and lockwire.
14. Install nut, lock and rod end on piston rod. Adjust rod end to 22.52 inches. (Center of mounting holes.) Tighten and lockwire nut.
15. Position cylinder upright, rod end down, and fill (approx 80-90 CC) cylinder with MIL-H-5606 hydraulic fluid.

NOTE

To assist in servicing the cylinder, a suitable fixture may be used to actuate the bungee piston rod three or four inches.

16. Install O ring on valve and install valve in upper gland. Lockwire valve.
17. Place bungee in mounting position and install bolt, spacers and nut through upper end fitting.
18. Close entrance door and connect bungee rod end to bellcrank with pin. Install cotter pin.
19. Open entrance door and check for smooth continuous lowering of the door. Check bungee for leaks.
20. Install trim panels as required.

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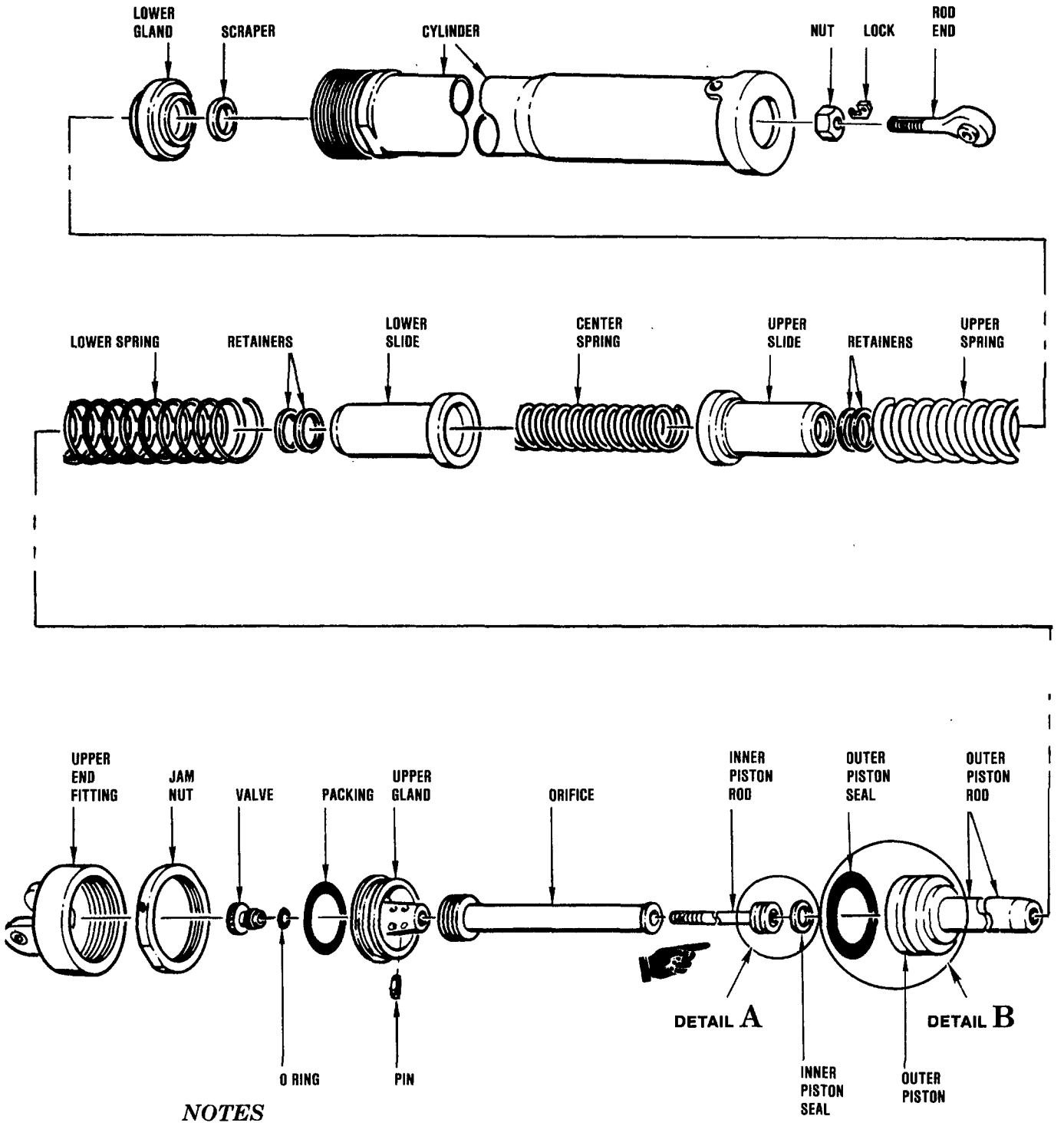

NOTES

- ***Refer to Illustrated Parts Catalog for component part numbers.***
- ***This bungee applicable to airplanes after S/B 74-27 or SOK 82-5.***

Figure 52-10-202A — Main Entrance Door Bungee (Sheet 1 of 3)

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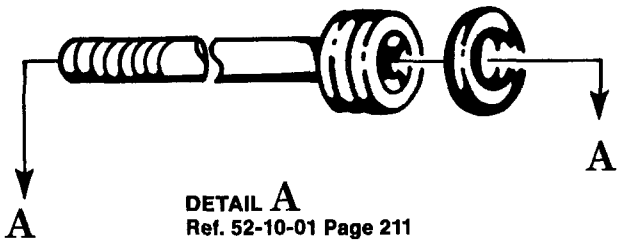
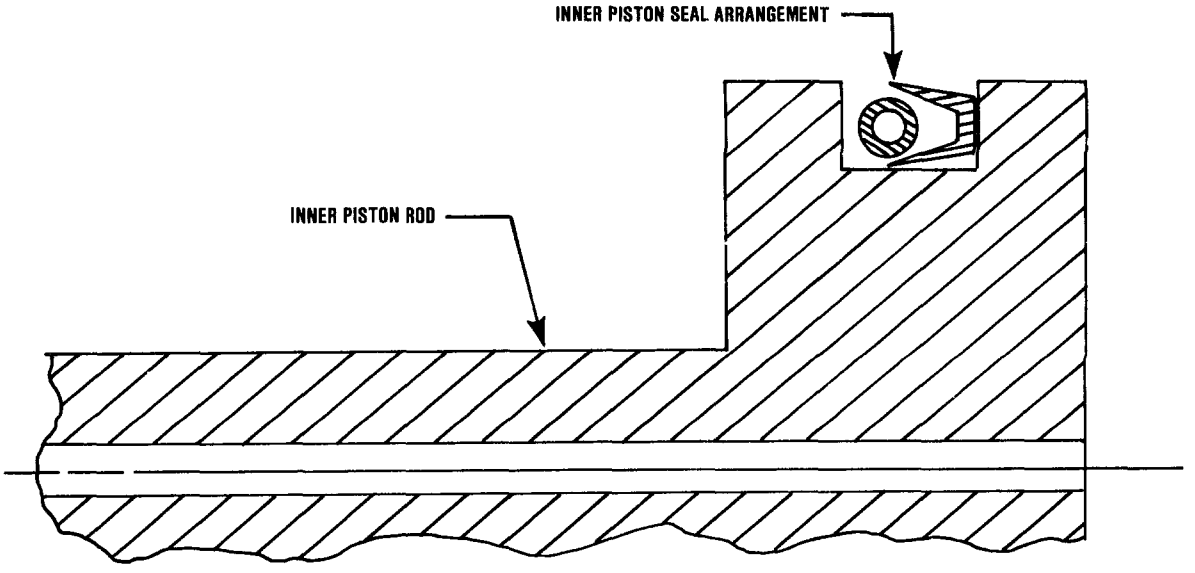


DETAIL A
Ref. 52-10-01 Page 211



SECTION A-A

Figure 52-10-202A — Main Entrance Door Bungee Inner Piston Rod Seal Arrangement (Sheet 2 of 3)

NEW

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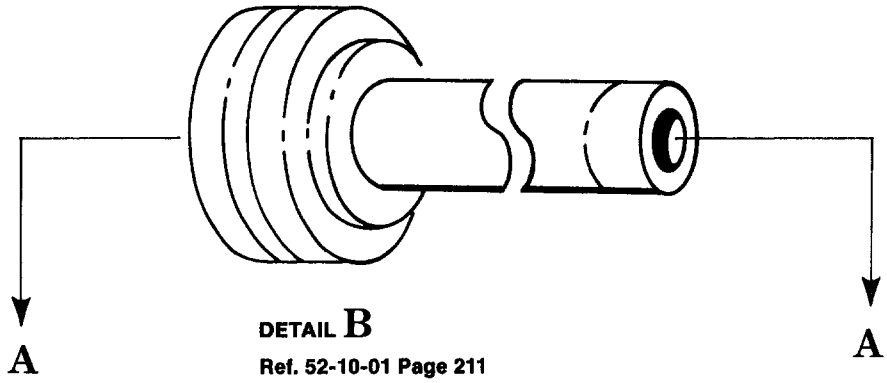

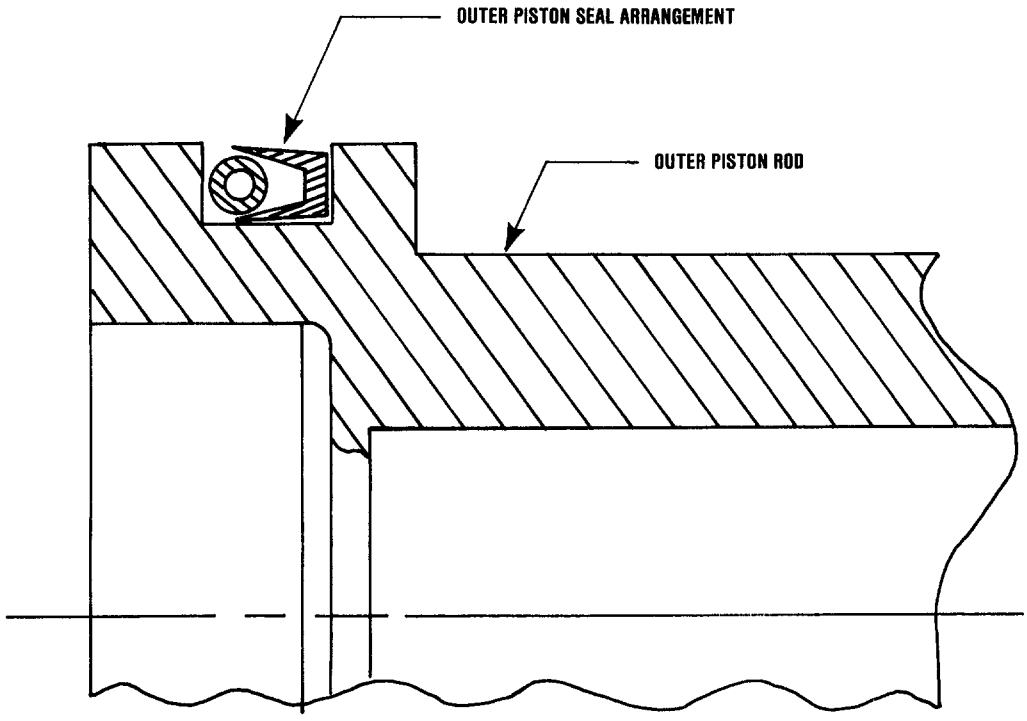

SECTION A-A

Figure 52-10-202A — Main Entrance Door Bungee Outer Piston Rod Seal Arrangement (Sheet 3 of 3)

NEW

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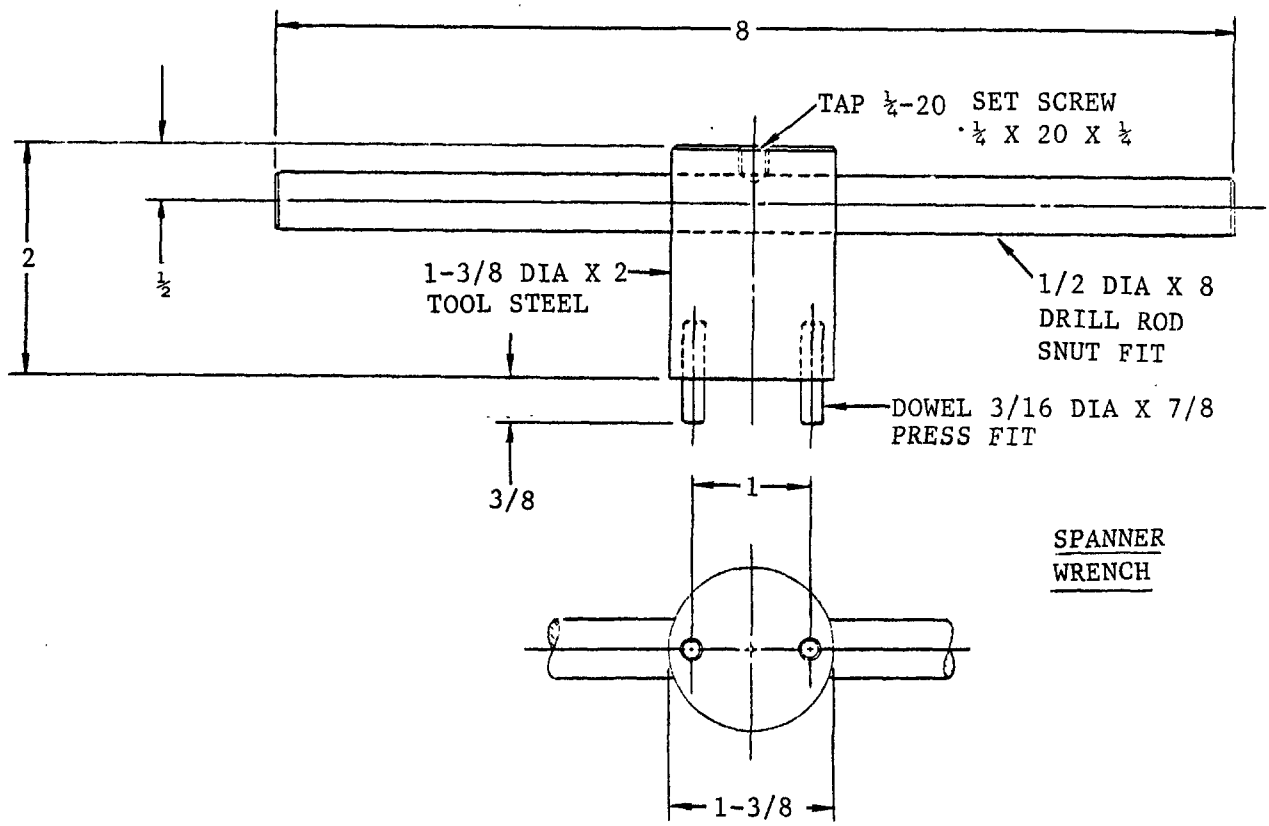
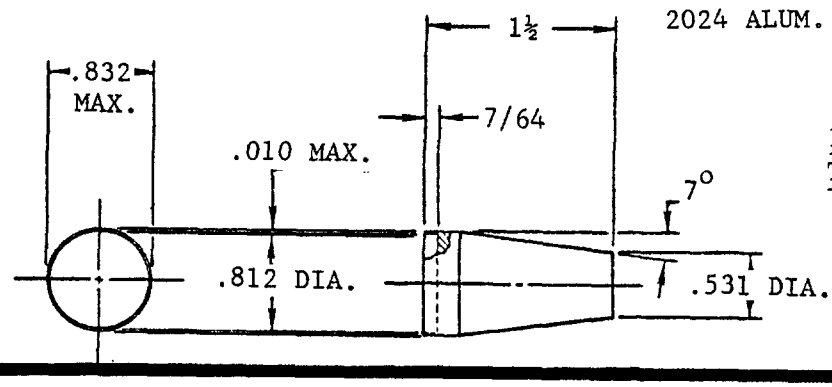


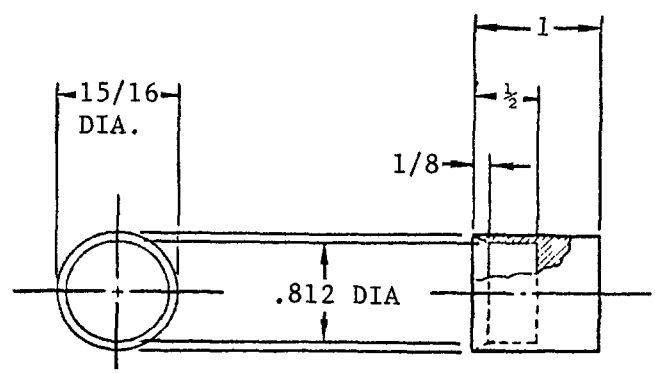
SPANNER WRENCH

2024 ALUM. 7/8 DIA. X 1½



INSTALLATION TOOL "A"

2024 ALUM. 1" DIA. X 1"

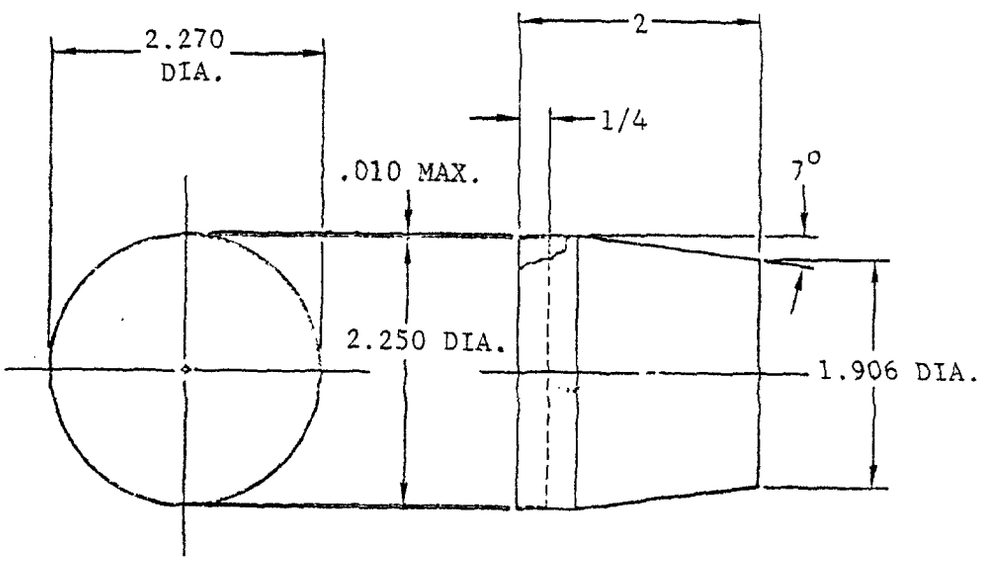


SIZING TOOL "B"

NOTE: All dimensions are in inches.

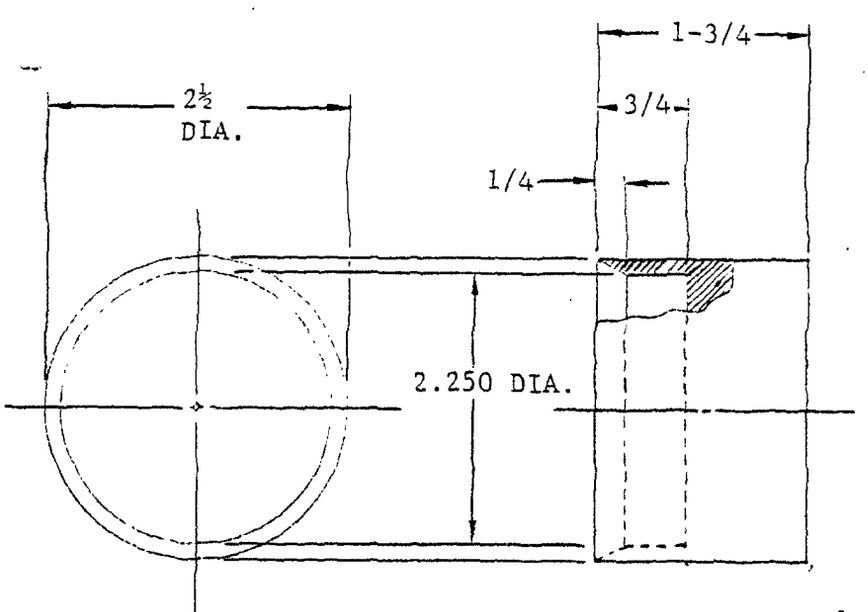
Entrance Door Bungee Tools (Local Manufacturer)
Figure 52-10-203 (Sheet 1 of 3)

INSTALLATION
TOOL "C"



2024 ALUM. 2-3/8 DIA. X 2

SIZING
TOOL "D"

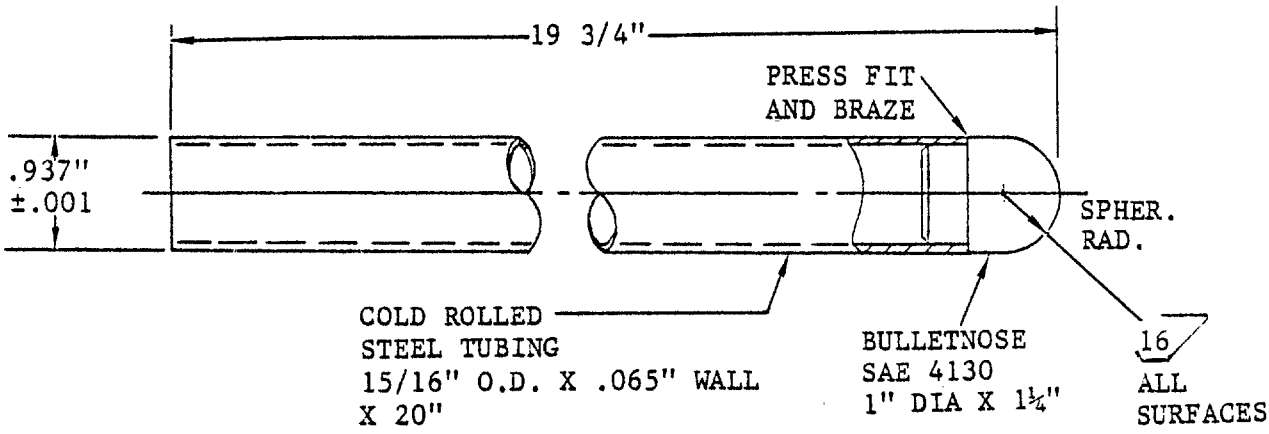


ORIGINAL
As Received By
ATP

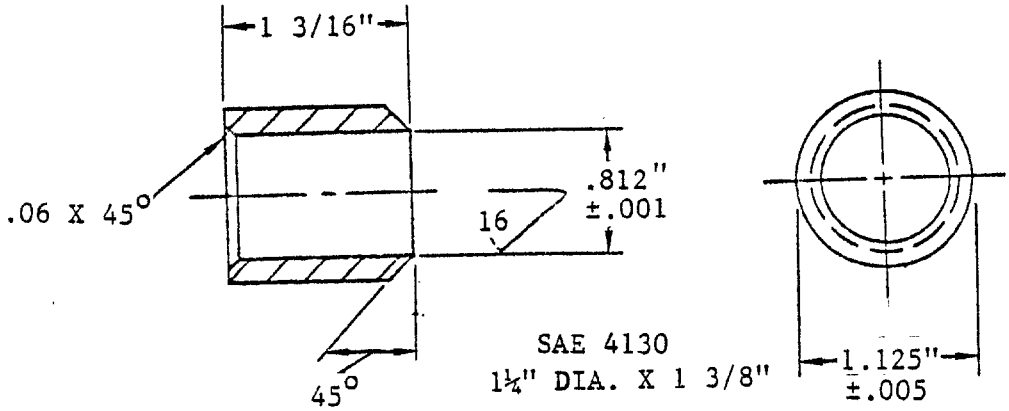
2024 ALUM 2½ DIA X 1-3/4

Entrance Door Bungee Tools (Local Manufacturer)
Figure 52-10-203 (Sheet 2 of 3)

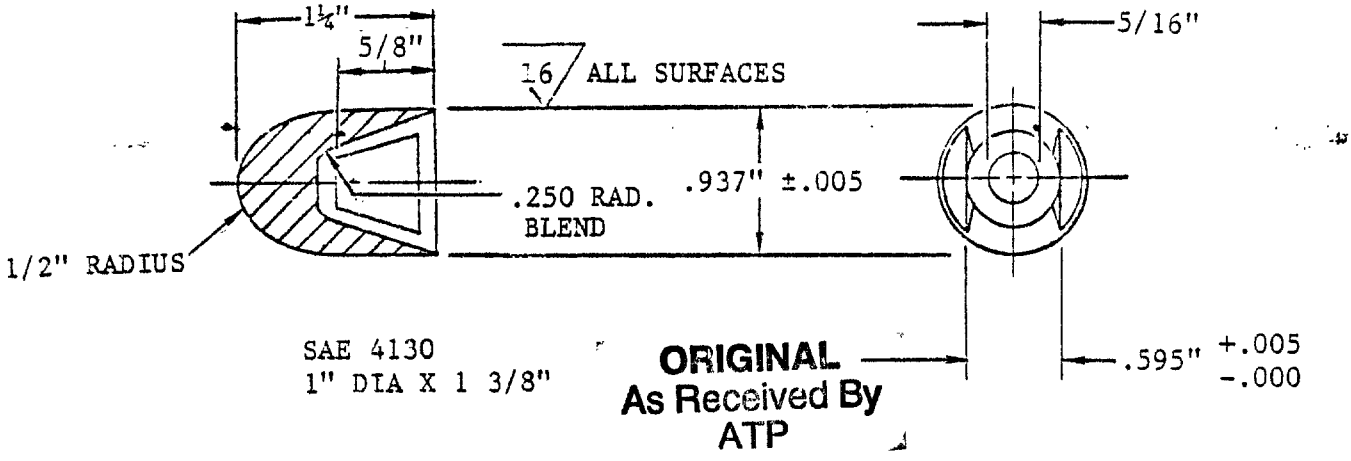
SIZING ROD "E"



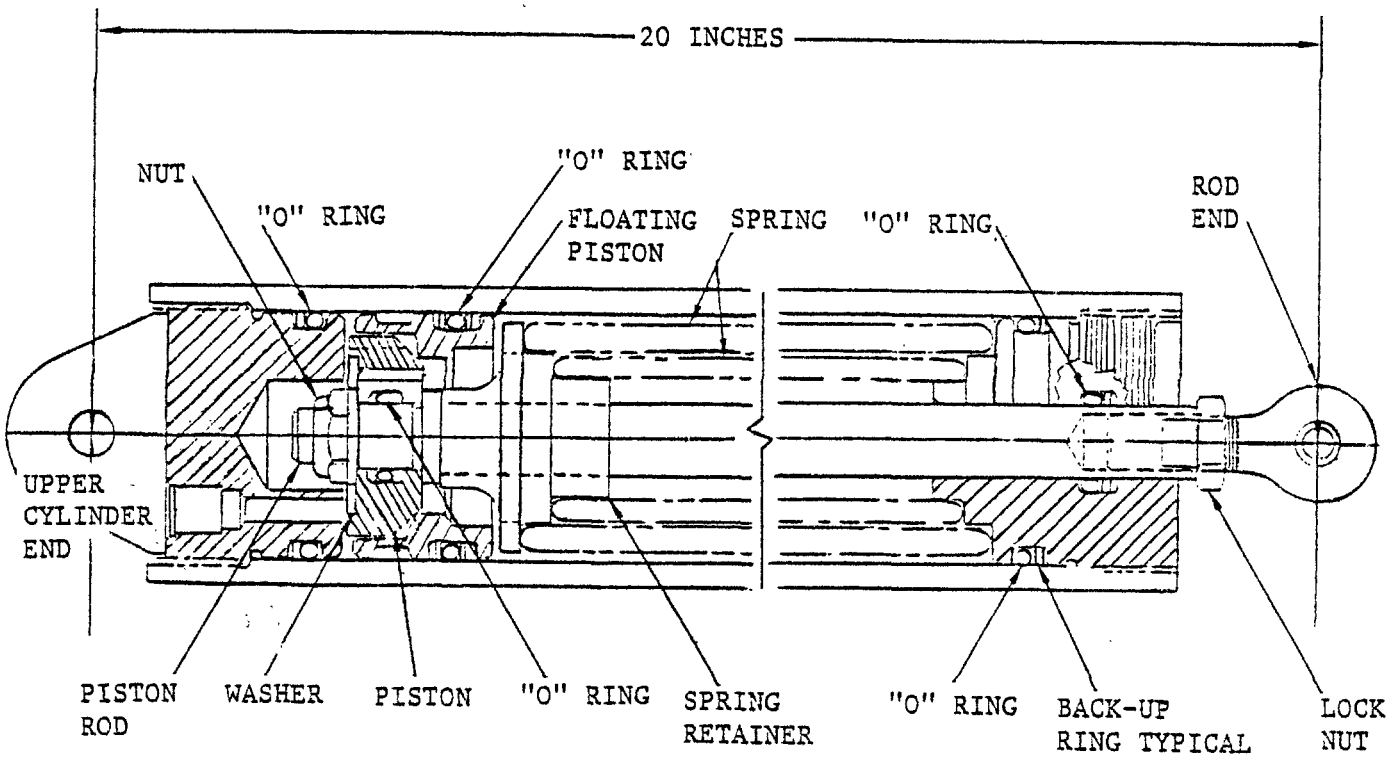
INSTALLATION SLEEVE "F"



INSTALLATION BULLET "G"



Entrance Door Bungee Tools (Local Manufacturer)
Figure 52-10-203 (Sheet 3 of 3)



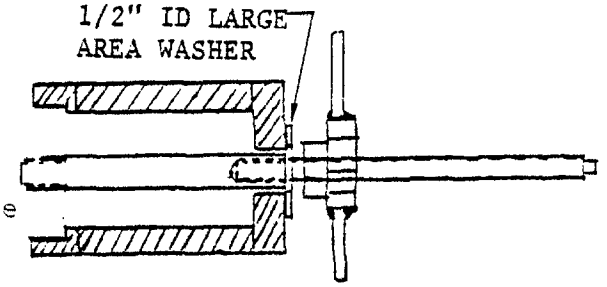
NOTE: Refer to Illustrated Parts Manual for "O" Ring P/N's.

- Applicable to NA-380-1 thru -4, -6 thru -9, -11 thru -13, -15, 1-7, -19, -21, -23, -25, -30 and -32 Not Changed by Service Bulletin 74-27.

ORIGINAL
As Received By
ATP

Main Entrance Door Bungee
Figure 52-10-204

NOTE: Applicable to NA-380-1 thru -4, -6 thru -9, -11 thru -13, -15, -17, -19, -21, -23, -25, -30 and -32 Not Changed by Service Bulletin 74-27.



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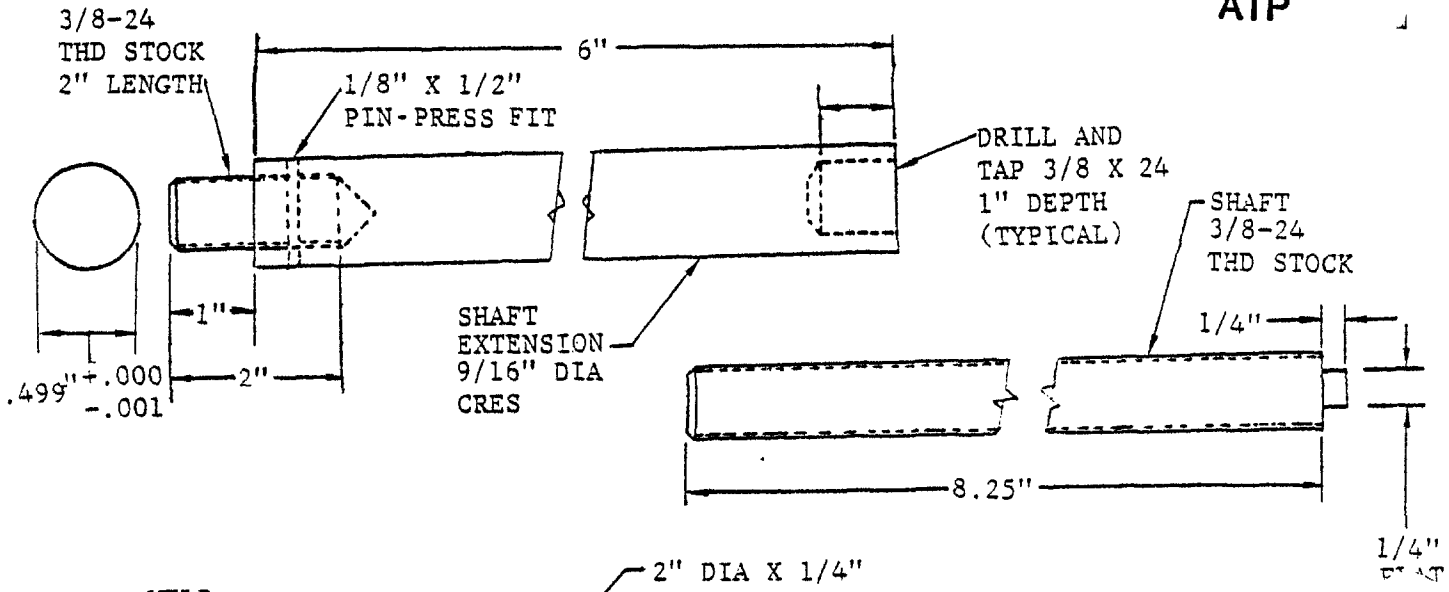

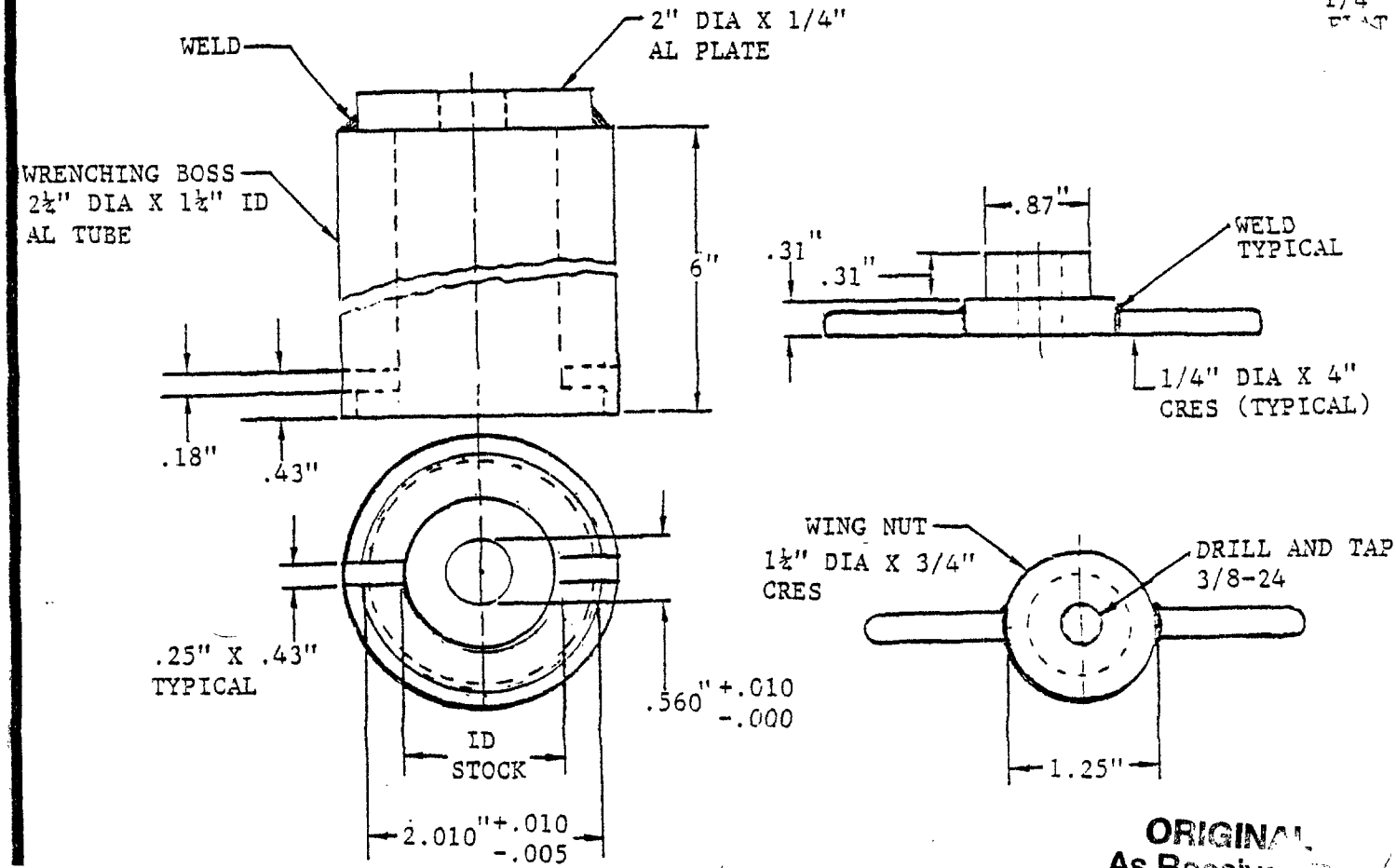

ORIGINAL
As Received By
ATP

Entrance Door Bungee Tool (Local Manufactured)
Figure 52-10-205

DOORS

INSPECTION/CHECK

INSPECTION — MAIN ENTRANCE DOOR FOR CORROSION

EQUIPMENT/TOOLS

- Inspection Light
- Inspection Mirror
- Magnifying Glass

CONSUMABLES

NOTE

The following consumables are necessary only if corrosion is detected during inspection.

- Paint Remover, Non-Acid Type
- Alodine 1201
- Epoxy-Polymide Primer
- Masking Tape

NOTE

Under certain geographical and environmental conditions with high humidity, high temperature, salt water or high industrial contamination, it may be necessary to increase the frequency of this inspection. This determination shall be made by the aircraft Owner and/or Operator.

PROCEDURE

1. Open main entrance door.
2. Carefully lift-up and remove stair tread coverings.
3. Remove access panels (2 ea.) installed during compliance with S/B 86-1 installation (see Figure 52-10-601).
4. Remove screws securing stair treads to structure of main entrance door, then remove stair treads from door. Save screws for reinstallation.
5. Using external light source, inspection mirror, magnifying glass, and any other inspection aids necessary, visually examine all interior areas of the main entrance door. Carefully examine the interior side of the entrance door outer skin for areas of moisture entrapment and the faying surfaces between the door structure and the door outer skin for indications of corrosion between the faying surfaces. Inspect the lower door stop and adjacent door structure for evidence of corrosion. Also inspect the internal door structure (via the access holes) for corrosion in the area adjacent to the lower door stop (see Figure 52-10-601).
6. Inspect interior finish of main entrance door for deterioration primer/paint and for cracked or missing sealant between faying surfaces.
7. If no corrosion or damaged paint films are detected, reassemble door.

NOTE

To accomplish steps 8 and 9 refer to SR-76-023 Corrosion Control Manual or S/B 86-1.

8. If damaged paint films are found, proceed as follows:
   A. Mask off areas adjacent to damaged area.
   B. Remove all paint from damaged area.
   C. Clean area using alkaline waterbase cleaner.
   D. Apply chemical conversion coating to cleaned area.
   E. Apply applicable primer/paint finishes to match surrounding area. (Refer to chapter 20.)
   F. Remove masking paper and tape; then reassemble door.
9. If corrosion is detected, determine extent and depth of corrosion damage, and contact Sabreliner Engineering Department for disposition. If damage is within acceptable limits, proceed as follows:
   A. Mask all components/areas which could be affected by corrosion removal chemicals or corrosion residue.
   B. Strip all paint/primer from corroded area.
   C. Clean corroded areas using alkaline waterbase cleaner.
   D. Remove corrosion using chemical and/or mechanical methods.
   E. Apply chemical conversion coatings to areas where corrosion has been removed.
   F. Apply applicable primer/paint to match surrounding area.
   G. Remove masking paper and tape.
   H. Reassemble main entrance door.

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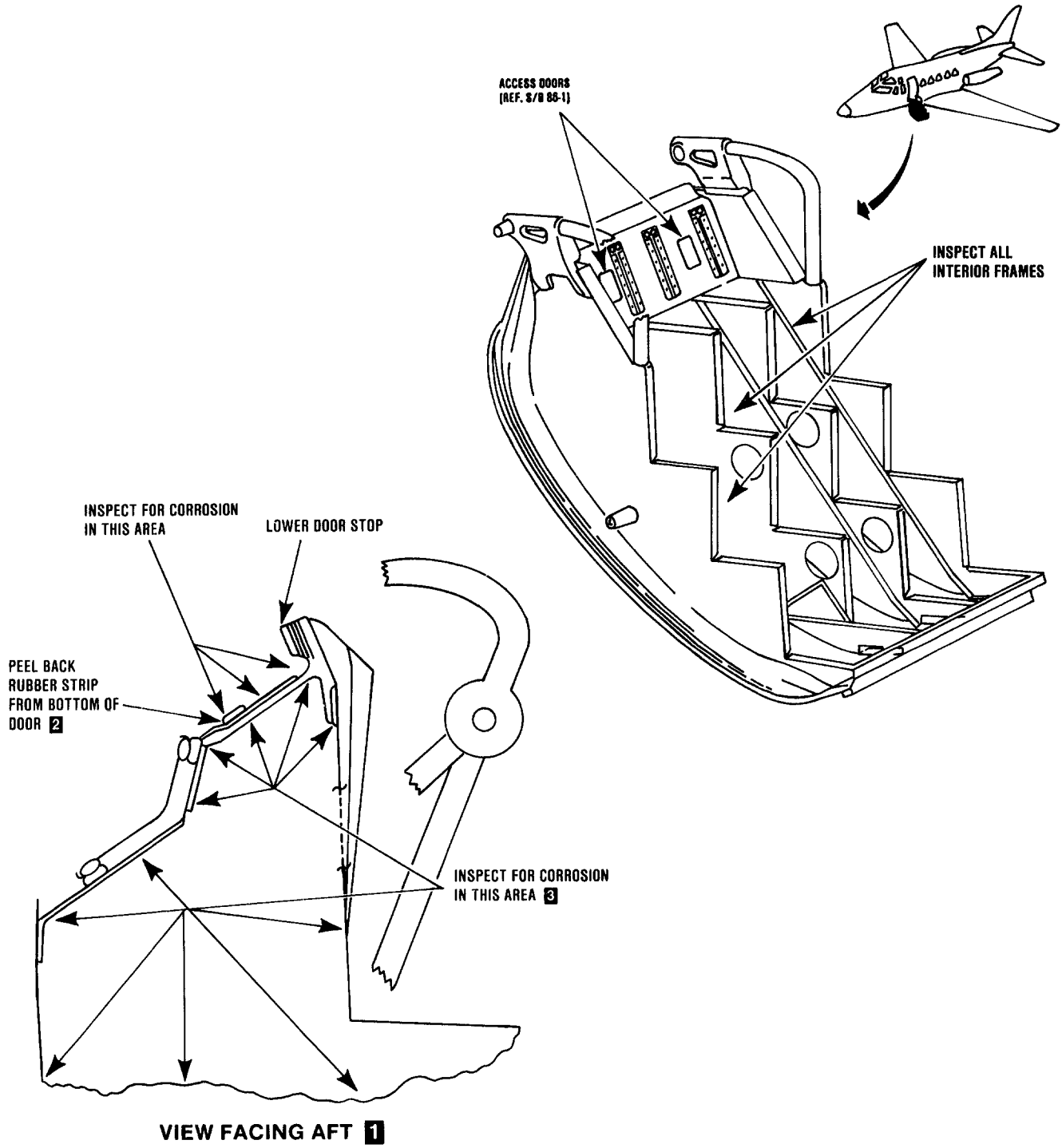
ACCESS DOORS (REF. S/B 86-1)
INSPECT ALL INTERIOR FRAMES
INSPECT FOR CORROSION IN THIS AREA
LOWER DOOR STOP
PEEL BACK RUBBER STRIP FROM BOTTOM OF DOOR 2
INSPECT FOR CORROSION IN THIS AREA 3
VIEW FACING AFT 1

1 ALL DOORS ARE INSPECTED IDENTICALLY, REGARDLESS OF AIRPLANE MODEL OR DOOR CONFIGURATION.

2 RE-CEMENT RUBBER STRIP USING RTV-108 ADHESIVE (GENERAL ELECTRIC) PER MANUFACTURER'S INSTRUCTIONS.

3 REMOVE AND DISCARD ANY INSULATION (IF INSTALLED) FROM DOOR CAVITY PRIOR TO INSPECTION.

NEW

Figure 52-10-601 — Main Entrance Door Inspection Areas

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EMERGENCY EXIT - DESCRIPTION AND OPERATION

1. General (See figure 52-00-1.)

Ground emergency exit doors are retained in place by two fixed pins on the bottom of each door and two sliding pins in the top of each door. The sliding pins may be released, and the door removed, from both outside and inside the airplane. For operation from outside the airplane, a release handle is provided in the upper left corner of the door. A handhold is also provided in the lower center of the door, to facilitate guiding the door into door into the airplane.

Operation of emergency exit doors from inside the airplane is by a single release handle, in the upper center portion of the door.

A seal is provided for emergency exit doors. This pressure seal is mounted on the door and is non-inflatable.

EMERGENCY EXIT - MAINTENANCE PRACTICES

1. Adjusting Emergency Ground Escape Hatch (Door)

A. To perform adjustment of the ground escape door, see figure 52-20-201 and proceed as follows:

NOTE: If a replacement door is being installed, the door may be trimmed as required to obtain the required skin gaps around the hatch opening. See figure 52-20-201 for skin gap tolerances.

- Do not final torque screws or bolts until final adjustments are complete.

(1) Remove any sealant installed around the periphery of the door opening and the door. Remove trim panels as required.

(2) Trim door as required to maintain proper skin gap. (See figure 52-20-201.)

(3) To position door in fuselage opening, proceed as follows:

(a) On the lower intercostal, loosen the four bolts securing the two adjustable, and the two fixed serrated plates.

(b) Position the adjustable serrated plates (forward or aft) so that door is centered fore and aft in the opening between fuselage frames. Inboard and outboard dimensions are .040 inch inside the fuselage mold line. This dimension is obtained by the inboard and outboard movement of the adjustable serrated plates.

(c) Correct an offset or cocked door by adding a shim as shown in figure 52-20-201 between the lower intercostal and the adjustable serrated plate.

NOTE: Shim thickness must not exceed .064 inch.

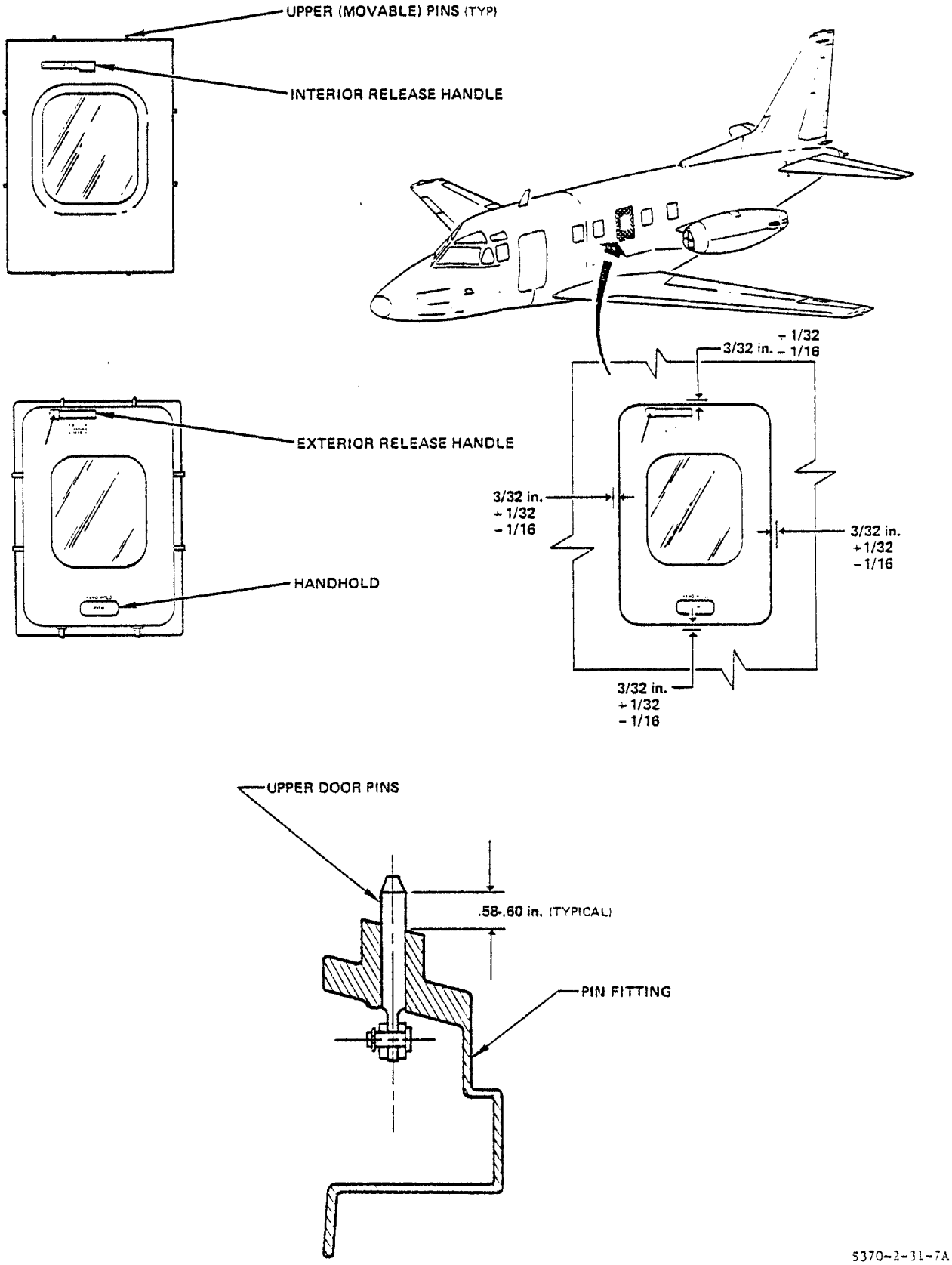
(d) Tighten bolts securing serrated plates.

(e) With the door positioned in the lower sockets and centered (fore and aft), check that the door moves in and out of the fuselage opening with an equal distance between the door side fittings and the fuselage frames.

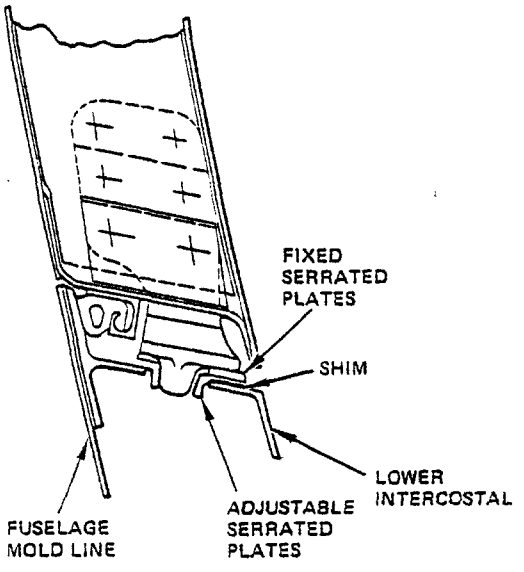
(f) On the upper intercostal, loosen the four bolts securing the two adjustable, and the two fixed serrated plates.

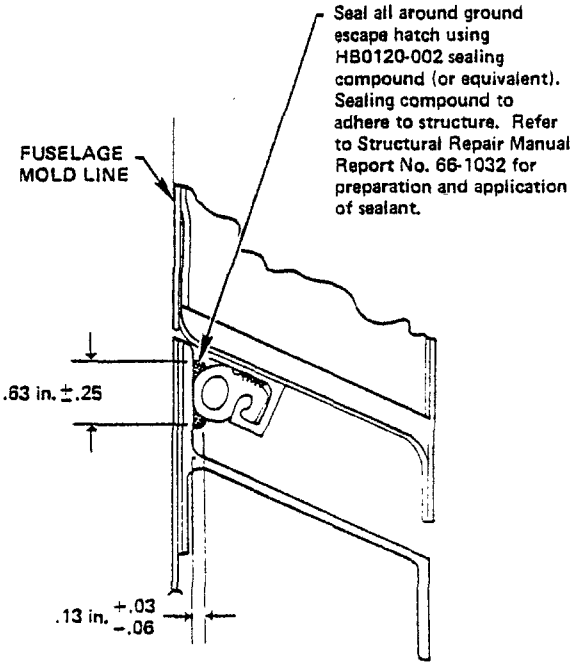
(g) Position the adjustable serrated plates forward or aft so that door is centered in the opening. Inboard and outboard dimensions are .040 inch inside the fuselage mold line. This dimension is obtained by the inboard and outboard movement of the adjustable serrated plates.

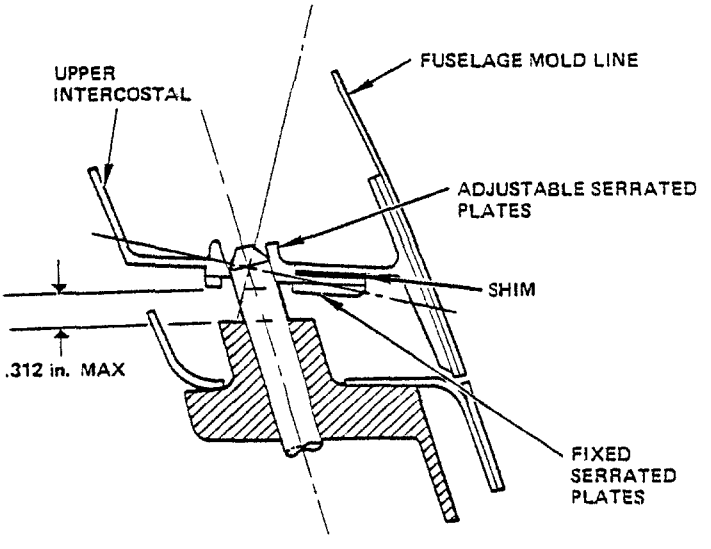
(h) Tighten bolts securing serrated plates.



Adjusting Emergency Ground Escape Hatch (Sheet 1 of 4)
Figure 52-20-201

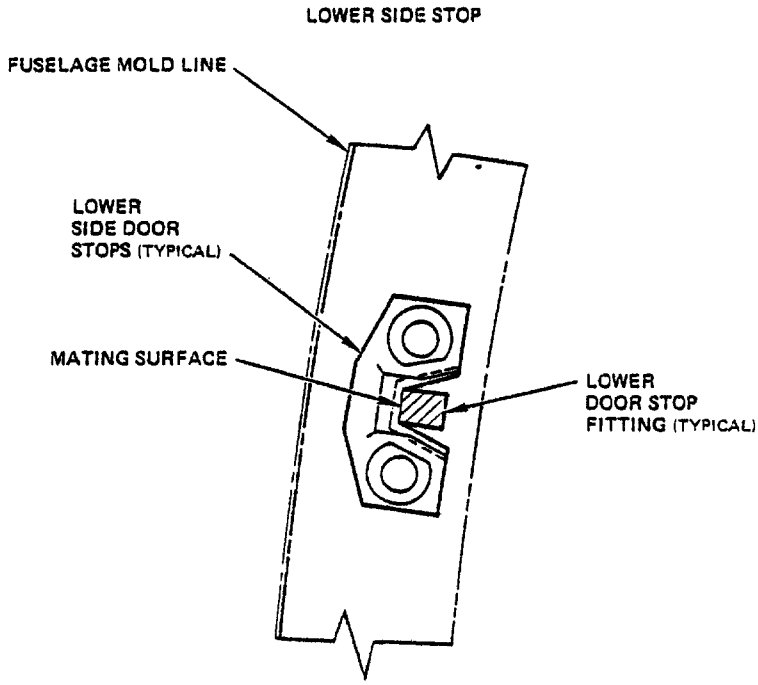



Seal all around ground escape hatch using HB0120-002 sealing compound (or equivalent). Sealing compound to adhere to structure. Refer to Structural Repair Manual Report No. 66-1032 for preparation and application of sealant.
FUSELAGE MOLD LINE
.63 in. ±.25
.13 in. +.03 −.06

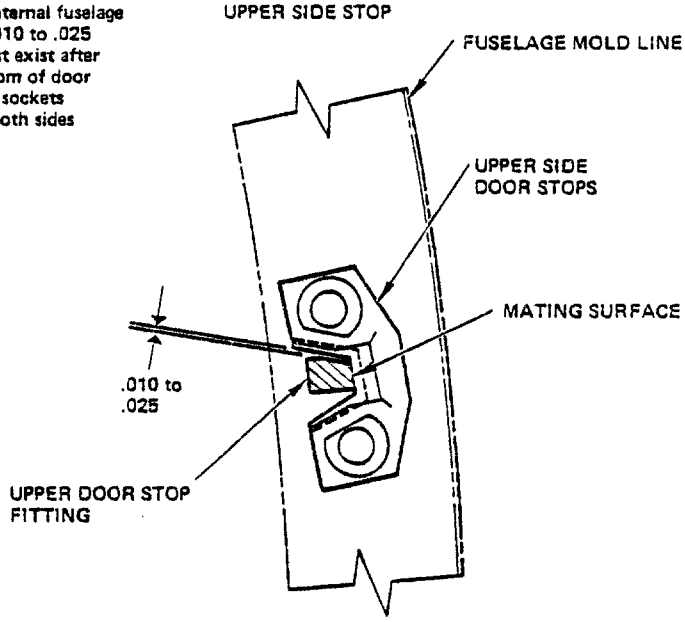


S370-2-31-24

Adjusting Emergency Ground Escape Hatch (Sheet 2 of 4)
Figure 52-20-201

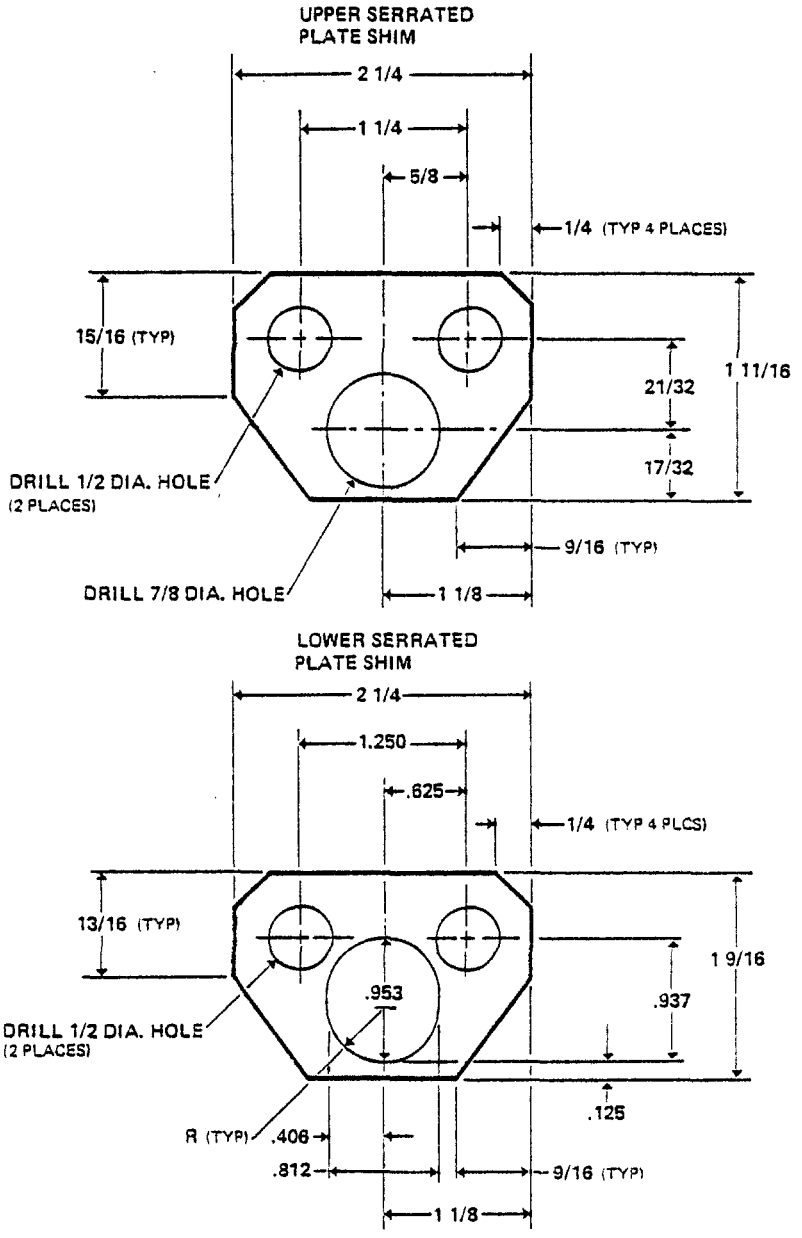


At 3.0 PSI internal fuselage pressure, a .010 to .025 inch gap must exist after pins, at bottom of door are seated in sockets (Typical at both sides of door)



S370-2-31-25

Adjusting Emergency Ground Escape Hatch (Sheet 3 of 4)
Figure 52-20-201



NOTE

1. FABRICATE SHIMS FROM 7075-T6 BARE AL. ALOY. SHT., QQ-A-250/12, OR 2024-T42 BARE AL. ALOY. SHT., QQ-A-250/4.
2. .064 MAXIMUM ALLOWABLE THICKNESS
3. *FINISH PER STRUCTURAL REPAIR MANUAL REPORT NO. 66-1032.*

S370-2-31-26

Adjusting Emergency Ground Escape Hatch (Sheet 4 of 4)
Figure 52-20-201

(4) Adjust door upper pins as follows:

(a) Remove door from fuselage and close and lock door handle.

(b) Check that door pins extend past pin fittings by .58 to .60 inches. Adjust applicable rod end fitting as required. Tighten jam nut on rod end fittings.

(5) Check door upper pin engagement as follows:

(a) Unlock door handle; then install door in fuselage opening. Lock door.

(b) Check that gap between upper surface of pin fitting and lower surface of serrated plate on upper intercostal does not exceed .312 inches. Install shim between upper intercostal and adjustable serrated plate as required. Do not exceed .064 shim thickness.

(6) Position door lower (forward and aft) side stops as follows:

(a) Loosen the four screws securing the two adjustable serrated stops (door side stops) and the two fixed serrated plates.

(b) Position the adjustable serrated stop inboard and outboard so that the two lower door fittings maintain an approximate gap of 1/16 inch to the mating surfaces of the two stops; also center fittings in stops. This gap is to allow movement of door during pressurization to meet mold line flushness requirements.

(c) Tighten screws securing stops.

(7) Position door upper (forward and aft) side stops as follows:

(a) Loosen the four screws securing the two adjustable serrated stops and the two fixed serrated plates.

(b) Position adjustable serrated stops inboard or outboard so that the two upper door fittings maintain an approximate gap of 1/16 inch to the mating surfaces of the two stops. Also position stops up or down to maintain .010 to .025 inch gap between upper surface of fittings and matching surface to stops. The .010 to .025 inch clearance will be checked in a subsequent step while the fuselage is pressurized to 3 psi.

(c) Tighten screws securing stops.

(8) To check mold line flushness, proceed as follows:

(a) Remove door from opening.

(b) Apply a thin coat of vasoline (or equivalent) on each of the four side door fittings; then install tooling putty (or equivalent) on the fittings. This will provide a gage to check clearances while the fuselage is pressurized.

(c) Install and lock door. Pressurize fuselage to 3 psi. (Refer to chapter 21.)

(d) Check that door is flush to 0.042 in. inside mold line at top and bottom edges and flush to 0.047 inside mold line on sides from top to bottom.

(e) If the flush check is not within tolerance, depressurize fuselage and readjust as required; then repressurize fuselage.

(f) Check for proper skin gap around door.

(g) Depressurize fuselage and remove door from fuselage.

(h) Check that depressions in tooling putty at the two upper side stops have a 0.010 to 0.025 in. gap between upper surface of door fittings and matching surface of stops.

(i) Check that depressions in tooling putty at all four side stops have touched (mating surface) all four stops.

(j) Check that depressions in tooling putty at the two lower side stops are centered in the two lower stops.

(k) If steps (h), (i) and (j) are not within tolerances, depressurize fuselage and readjust stops as required.

(l) Depressurize fuselage and remove door. Clean putty from stops. Repeat step (8) (b).

(m) Install and lock door and pressurize fuselage to 1.5 psi.

(n) Depressurize fuselage and remove door.

(o) Check that depressions in tooling putty at all four side stops are within 0.047 in. of touching stops.

(p) If step (o) is not with limits, readjust stops as required and repeat clearance check using new putty.

(9) Final torque all screws and bolts in serrated plates. Check complete installation for proper lockwire, tight jam nuts and security; then lubricate door. (Refer to chapter 12.)

(10) Apply self contouring sealing pad around door seal. Install trim panels as required.

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WARNING: To prevent possible injury to personnel during the following procedures, only medically checked and approved personnel should be allowed to occupy the cabin and cockpit. Pressure changes inside the airplane must not exceed 1000-foot-per-minute altitude changes as shown in a pressure rate-of-change indicators.

(11) Seat weather seal between door and fuselage skin as follows:

NOTE: The weather seal need not be seated if the airplane is sheltered and there is reasonable certainty that the airplane will not be subjected to rain or washing prior to the next flight.

(a) Secure airplane in a suitable position to allow high power settings. Lock both main landing gear wheels and set parking brakes.

(b) Pressurize airplane. (Refer to chapter 21.)

(12) Close and lock door(s).

MAJOR CHANGE

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INSPECTION/CHECK

INSPECT — EMERGENCY GROUND ESCAPE HATCH

1. Remove interior panels as necessary to access door handles; then, remove escape hatch.
2. Inspect pressure seal for cuts, abrasion, distortion, or delamination. Replace pressure seal if damaged, cut, or torn in any way.
3. Inspect rain seal for cuts, abrasion, distortion, or delamination. Repair or replace as necessary.
4. Inspect all contact areas for corrosion.
5. Actuate door release handle to extend and retract pins. Inspect pins for any damage. Pins having chipped or delamination finishes should be replaced. Ensure that handle and pins move freely.
6. Ensure that escape hatch removes and installs freely. If binding occurs, refer to Adjusting Emergency Ground Escape Hatch this chapter.
7. Install escape hatch. Install interior panels removed in step 1.

NOTE

Cabin must be pressurized upon installation of escape hatch. This will properly seat the main pressure seal, preventing moisture from entering cabin.

INSPECT — EMERGENCY GROUND ESCAPE HATCH SEALED AREAS FOR CORROSION (Ref. S/B 87-5)

NOTE

Initial compliance with S/B 87-5 is required prior to this inspection.

1. Remove interior panels as necessary to access door handles; then remove escape hatch. (See Figure 52-20-202.)
2. Using a 90 degree boroscope, inspect for corrosion through each of 16 inspection holes previously complied with in S/B 87-5. Move soundproofing batts as required to allow complete inspection of escape hatch interior areas.

CAUTION

Do not pierce soundproofing batts. See Figure 52-20-202.

3. If no corrosion is found, return emergency ground escape hatch to its original installed configuration.
4. If corrosion is found, it will be necessary to gain access to corroded area by removal of the inner escape hatch skin, P/N 370-300027-33 LH or -34 RH. Refer to Corrosion Control Manual SR-76-023 for removal and treatment of corrosion.

NOTE

Any repairs accomplished should be made in accordance with Sabreliner Structural Repair Manual NA-66-1032 and/or Sabreliner Corrosion Control Manual SR-76-023.

5. Reinstall inner skin using fastener callouts as shown in S/B 87-5 initial compliance and accomplishment requirements.
6. Install ground escape hatch as removed in step 1.

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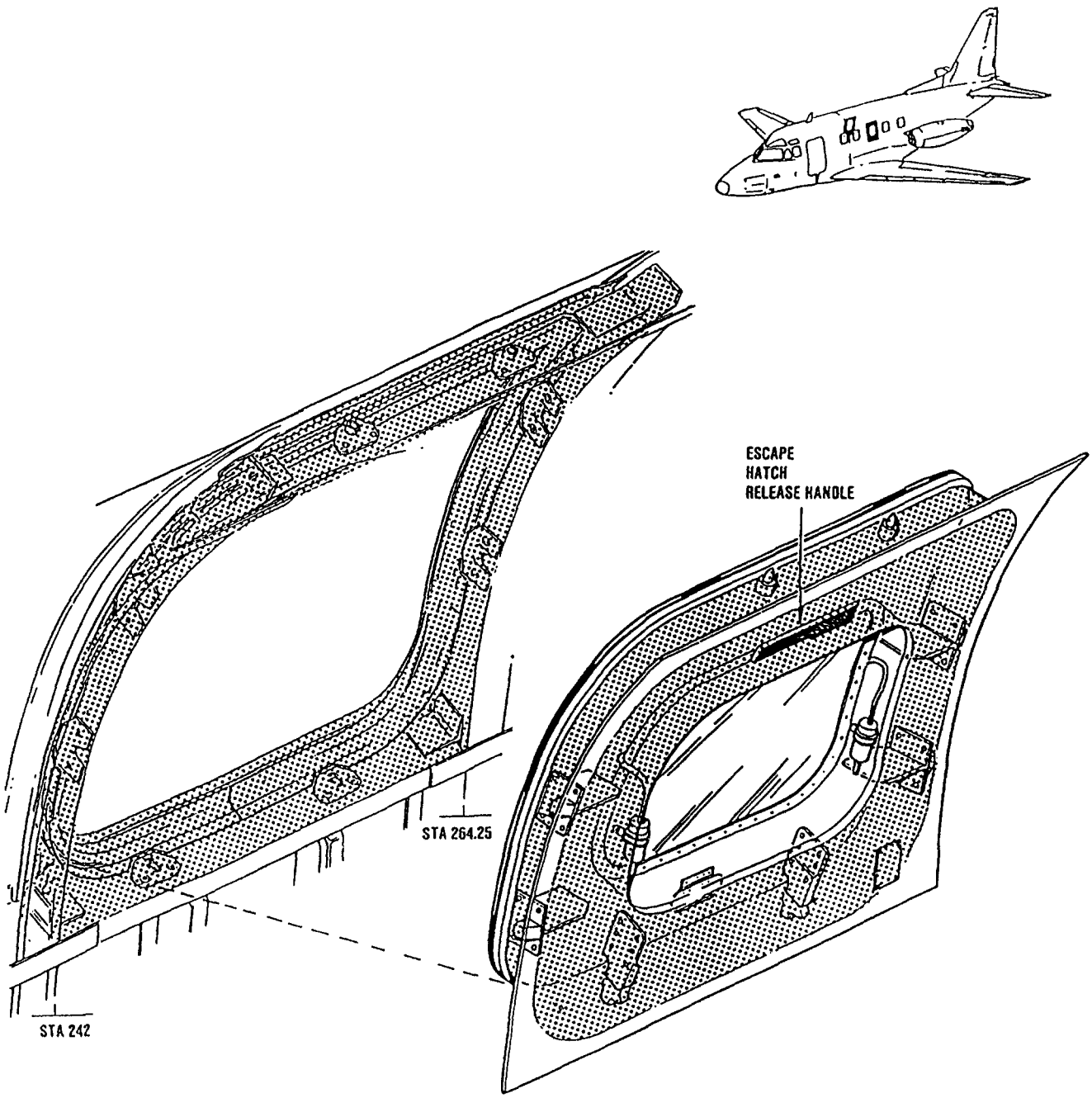

Figure 52-20-202 — Emergency Ground Escape Hatch
Corrosion Inspection (Sheet 1 of 2)

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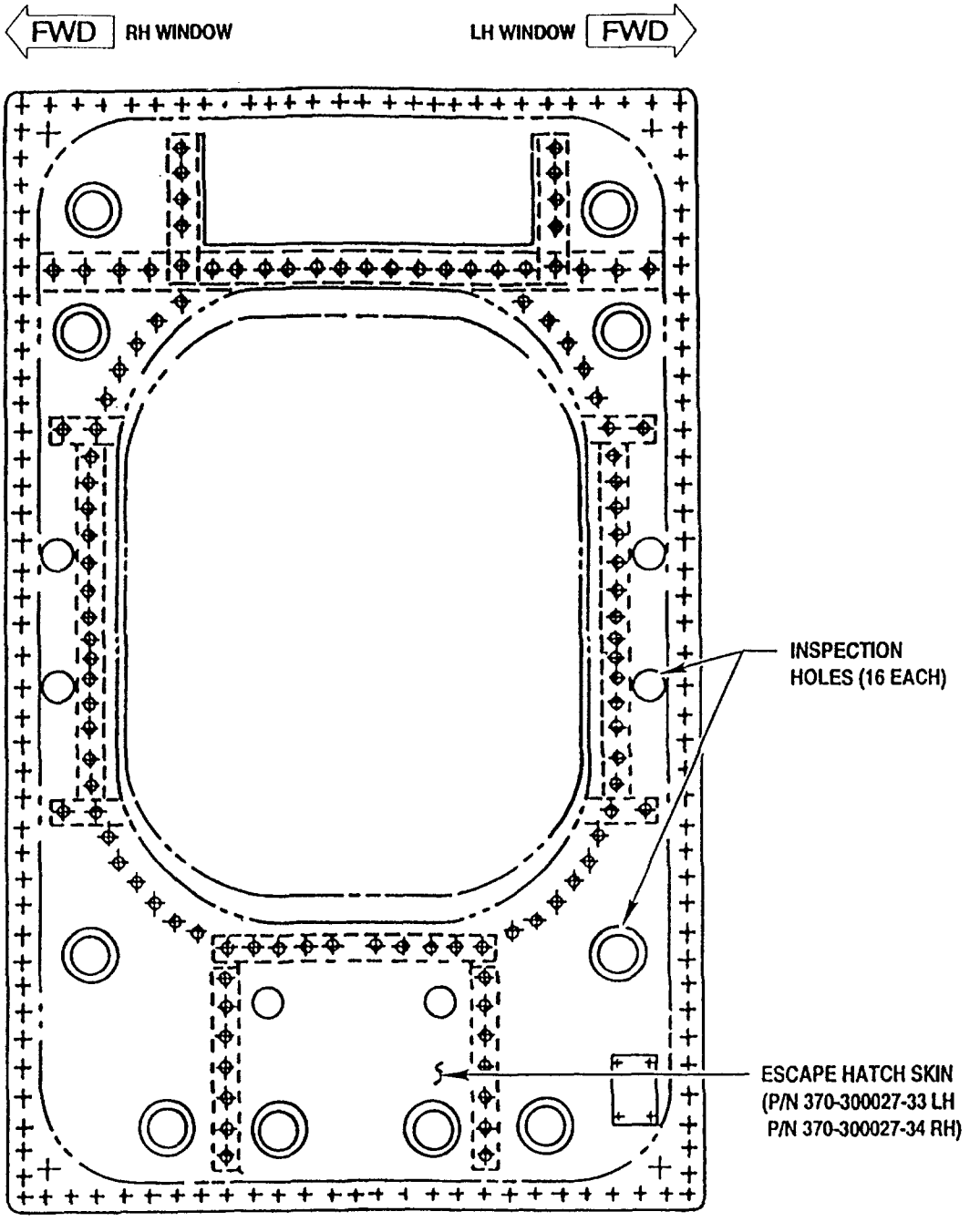

Figure 52-20-202 — Emergency Ground Escape Hatch
Corrosion Inspection (Sheet 2 of 2)

NEW

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ENTRANCE DOOR WARNING SYSTEM - DESCRIPTION AND OPERATION

1. General

The entrance door warning system consists of a microswitch, actuated by the aft door lockpin, and a warning light, on the pilot's instrument panel. The system is powered through the "DOOR SEAL LOCK" circuit breaker from the 28-volt dc essential bus. The door lock switch also controls the door seal solenoid pressure regulator. If the entrance door is not closed and locked, the door seal will not inflate, and the "DOOR OPEN" caution light will illuminate.

LANDING GEAR DOORS - DESCRIPTION

1. Main Landing Gear Doors

The main landing gear doors consists of a pair of hinged doors mounted on the outboard side of the left and right main gear shock struts. The upper strut door is attached to a hinged fitting at the upper end of the door and is connected to the wing structure at the outboard end of the main gear side brace. The lower end of the door is hinged to the lower strut door. Mechanical linkage connecting the two doors folds the lower door upward when the gear is extended to provide ground clearance. The doors are locked in the up position by a hook and roller combination. The hook is connected to, and operated by, the main gear uplock mechanism. The main gear doors are adjusted to close flush with the fuselage and wing surface.

2. Nose Landing Gear Doors

Two nose landing gear fairing doors attach to the fuselage and completely cover the nose gear well upon retraction of the nose gear. The two doors are riveted and resistance-welded sheet-metal assemblies. The principal members are formed inner skins, outer skins, and internal sheet-metal stiffeners. The left door carries a centerline aerodynamic seal.

CHAPTER

53

FUSELAGE

CHAPTER 53

FUSELAGE

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CHAPTER 53

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	622	Oct 26/90
	623	Oct 26/90
	624	Nov 12/93
	625	Nov 12/93
	626	Jun 22/90
	627	Apr 4/92
	628	Apr 4/92
	629	Apr 4/92

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		635	Oct 5/92
		636	Oct 5/92
		637	Oct 5/92
		638	Oct 5/92
		639	Oct 5/92
		640	Oct 5/92
		641	Oct 5/92
		642 *	Feb 29/96
		643 *	Feb 29/96
	Added	644 *	Feb 29/96
	Added	645/646 *	Feb 29/96
53-00-04		801	Mar 1/70
		802	Mar 1/70
53-30-00		1	Aug 6/71

* The asterisk indicates pages revised, added, or deleted by the current revision.

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FUSELAGE

DESCRIPTION AND OPERATION (Continued)

GENERAL

The fuselage is of all-metal, stressed-skin semimonocoque construction, with transverse frames and longerons. A radome, made of dielectric material, forms the forward end of the fuselage from FS 0 to FS 13. The fuselage, for explanation purposes, is separated into three distinct areas. They are the fuselage forward, intermediate, and rear sections. The cockpit and cabin are pressurized and soundproofed for personnel comfort at high altitudes. The nose section, from FS 0 to FS 50, is also pressurized for installation of pressurized equipment.

Chem-Milled aluminum alloy stressed skins, along with formed sheet metal and extruded frames (some of which are Chem-Milled), carry the shear loads. The major portion of the bending loads is carried by the four main longerons. The remainder of the bending loads is carried by the various stringers, intercostals, and secondary longerons. The thinned areas of the Chem-Milled fuselage skins have been reduced to a minimum gauge consistent with the applicable load and fatigue factors and crack propagation resistance.

FORWARD SECTION

The following are contained within the fuselage forward section: batteries and sump jar, space provisions for radio, radar, and electronic equipment installation, and a nose landing gear wheel well. The cockpit enclosure is designed with double-panel overhead and side windows and an electrically heated birdproof windshield. A sliding window, provided for the pilot, permits ventilation and communication when the airplane is on the ground. An antifogging, or dehydration, system is provided between all of the cockpit windows, except the windshields, forward overhead windows, and direct-vision windows, to dehumidify trapped air.

INTERMEDIATE SECTION

The fuselage intermediate section, commonly referred to as the cabin, consists mainly of an entrance door and the passengers' compartment. On S/N 370-1, -7, and 380-1 and subs, an electrical equipment rack is installed on the right side of the fuselage between FS 143 and 158.5. On NA 265-80 airplanes equipped as airway checker airplanes, two avionics racks, each having a controlled environment, and a galley are installed on the right-hand side of the fuselage between FS 160 and 206+9. The passengers' compartment is equipped with triple-paned windows, a dehydrator system, and the oxygen system. Plug-type emergency ground escape hatches are provided on each side of the passengers' compartment. Entrance to the airplane is gained by a plug-type entry door which rotates down, forming steps for entry.

REAR SECTION

The fuselage rear section accommodates a 203-gallon fuselage fuel tank, environmental system equipment hydraulic components and on S/N 370-2 thru -6, -8 and -9, an electrical relay panel. The auxiliary power unit (if installed) is also located in the fuselage rear section. The engine support pylons are mounted on each side as an integral part of the fuselage intermediate and rear sections. Access doors are provided for maintenance of the various systems and components.

CAUTION

On airplanes with the remote sensing unit for the standby compass mounted in the aft fuselage at FS 522.5; a minimum of 36-inches must be maintained between the sensing unit and any steel or magnetic materials added in this area. Non-magnetic materials are the only acceptable materials that may be added in this area.

LIGHTNING PROTECTION

Lightning protection for the fuselage is provided by a conductor along the upper centerline of the radome and two lightning diverter rods attached to the aft end of the fuselage rear section. The strip along the upper centerline of the radome is a 4-mil copper-foil, 1/2-inch wide strip. These strips are not grounded to the airframe. The two lightning diverter rods on the aft end of the fuselage rear section prevent lightning from striking the fuel vent outlet and on S/N 370-2 thru -6, -8 and -9, the fuel jettison outlet. The lightning rods are constructed of 2024-T42 aluminum alloy and provide a mounting base for a static discharger.

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FUSELAGE - TROUBLE SHOOTING

1. Trouble Shooting Radome Lightning Diverter Strip

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. EXCESSIVE STATIC IN COMMUNICATION EQUIPMENT.		
(1) Metal lightning diverter strip damaged by lightning.	Check upper centerline of radome for burn marks, indicating a lightning strike.	Check resistance of lightning diverter strip. Refer to 53-00-3, paragraph 1.
(2) No continuity between lightning diverter strip and airframe.	Excessive burned and/or damaged area around lightning diverter strip.	Replace diverter strip.

FUSELAGE

MAINTENANCE PRACTICES

NOTE

Repairs must be accomplished in accordance with Sabreliner Structural Repair Manual (Report No. NA-66-1032) instructions or by Federal Aviation Agency approved repair procedures for safe life structure.

REMOVING LIGHTNING DIVERTER RODS

1. Drive horizontal Rollpin from diverter rod support.
2. Drive vertical Rollpin from diverter rod support. On lower rod support, this Rollpin will go into the aft end of the fuselage. It is not recoverable.

INSTALLING LIGHTNING DIVERTER RODS

1. Position rod into support, aligning Rollpin holes if rod has been previously installed.

NOTE

When installing new aluminum-type lightning diverter rods, push rod full distance into support; then rotate rod until static discharger can be installed with discharger pointing up in a vertical plane.

2. Drill two No. 12 holes through new lightning diverter rod (using support holes for a template).
3. Install two Rollpins (NAS561-6-14).

REMOVING AND INSTALLING LANDING GEAR WING FAIRING

Removal:

1. Defuel airplane (refer to chapter 28).

NOTE

The airplane must be defueled before removing fairing, as screws coded 6 and 7 on Figure 53-00-1 are threaded into the fuel tank.

2. Remove all screws (94) from fairing, and remove fairing.
3. Re-install screws (finger tight) coded 6 and 7 on Figure 53-00-1, to prevent foreign materials or contaminants entering the fuel tank.

Installation:

1. Check the bottom of tank and top of fairing for cleanliness and loose hardware.
2. Remove screws coded 6 and 7 (if screws are installed).
3. Install washers and new ring seals on screws coded 6 and 7. See Figure 53-00-1.
4. Place fairing on mounting structure and install all screws finger tight. See Figure 53-00-1 for screw pattern.
5. After all screws have been installed, tighten all screws (see Figure 53-00-1 for torque values).
6. Refuel airplane and check fuel tank for leaks.

APPLICATION OF ABRASION RESISTANT COATING

There are several areas on the exterior surface of the airplane to which abrasion resistant coatings have been applied. See chapter 20 for areas and dimensions of abrasion resistant coated areas. The purpose of the abrasion resistant coating is to relieve material abrasion and the resultant surface corrosion which follows.

Whenever an airplane is to be stripped and refinished, the stripping process will remove the abrasion resistant coatings along with the existing finishes. Reapplication of the abrasion resistant coatings must be accomplished during the refinishing of the airplane.

The abrasion resistant coating (Everlube 1329M) is a two part system consisting of a coating and a catalyst (molybdenum disulphide in an epoxy base) which may be applied by spraying, brushing or dipping. The coating when mixed to the manufacturer's specification is of the correct consistency for spray or brush application. If additional thinning is necessary, the addition of not more than 5 percent by volume, of a thinner recommended by the manufacturer, is acceptable. The catalyzed coating will have a pot life of 4 hours at 70-80 degrees. All surfaces to be coated shall be free of oxides, scale, oil, grease or other contaminates. Abrasion resistant coatings may be applied directly onto aluminum surfaces which have been chem film treated, or to CRES steel surfaces coated with a wash primer. All abrasion resistant coated areas will be covered and obscured by the application of the finish top coating.

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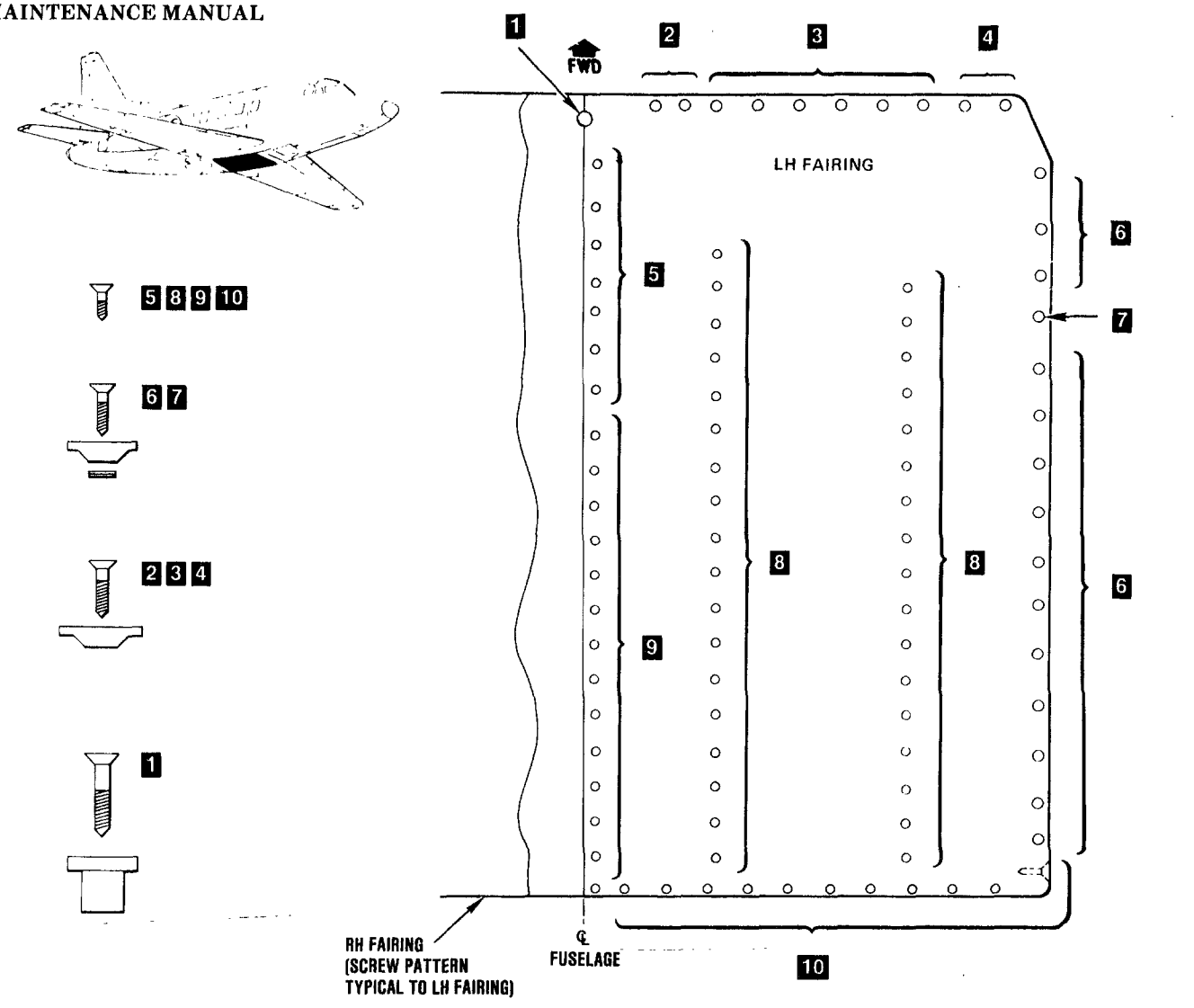

CODE	QUANTITY	SCREWS (SABRELINER PART NO.)	WASHERS (SABRELINER PART NO.)	RING SEAL (SABRELINER PART NO.)	SPACER (SABRELINER PART NO.)	TORQUE (INCH LBS)
1	1	NAS335CPA22	—	—	370-131007-3	140-203
2	2	370-131013-11	370-131014-7	—	—	70-90
3	6	370-131013-9	370-131014-7	—	—	70-90
4	2	370-131013-7	370-131014-7	—	—	70-90
5	7	7S34-3A2	—	—	—	—
6	14	370-13013-19	370-131014-9	5R21-11	—	140-203
7	1	370-13013-13	370-131014-7	5R21-10	—	70-90
8	35	7S34-3A4	—	—	—	—
9	14	7S34-3A7	—	—	—	—
10	12	7S34-3A3	—	—	—	—

NOTES

- *All screws have torque-set recess heads except code 1 screw (Phillips)*
- *Part numbers are marked on 370-131013 screws (screw head)*

S370-2-31-22

Figure 53-00-1 — Removing and Installing Landing Gear Wing Fairing

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FUSELAGE

INSPECTION/CHECK

CHECKING RADOME LIGHTNING DIVERTER STRIP FOR RESISTANCE

Measure resistance between forward end of metal strip and first radome mounting screw outboard (left or right) of upper centerline screw. There should be continuity and 3 ohms or less resistance.

NOTE

The protective coating on the metal strip should not be scratched or damaged when the resistance check is being made. The metal strip should have the resistance check made after radome repair and before refinishing.

CHECKING RADOME NOSE BOOT

1. The radome nose boot is the focal point of the airstream during flight and protects the fiber glass radome. The boot must be replaced when it shows signs of excessive wear or damage.
   A. Check nose boot for cuts tears, weather erosion, and any other signs of excessive damage.

INSPECT — FUSELAGE INTERIOR (Canted Fus. Sta. 457.4 to 485)

Equipment/Tools

- None

Consumables

- None

Inspection

1. Remove appropriate access panels and inspect the following:
   A. Stabilizer support fittings for corrosion, security and cracks.
   B. Structural components visually for cracks, distortion and loose or missing fasteners.
   C. Accessible electrical wiring and connections externally for cleanliness, security, corrosion and proper safetying where required. Electrical leads for chafing and damage.
   D. On S/N 370-2 through -6, -8 and -9 airplanes, fuel jettison and vent tubes for security condition, evidence of chafing against structural frames.
   E. Close access panels.

INSPECT — AFT FUSELAGE INTERIOR

Equipment/Tools

- None

Consumables

- None

Inspection

1. Inspect the following:
   A. Horizontal stabilizer trim actuators for security, interconnecting flexible shaft disconnects for proper safety.
   B. Hydraulic pressure transmitters for evidence of leakage, transmitter electrical connectors for security and safetying.
   C. Accessible hydraulic components and line connections for evidence of leakage. Motor and pump for evidence of overheating and damage.
   D. Accessible environmental components, ducts and lines for evidence of external damage and clearance from surrounding structure; couplings for cuts and tears, and clamps secure.
   E. Fire detector control boxes for security on S/N 370-2 through -6, -8 and -9 airplanes.
   F. Ram-air emergency air-source door for freedom of operation.
   G. Electrical relays and associated wiring for condition, security and evidence of damage.
   H. Horizontal stabilizer bungee (P/N 265-528210) for security and evidence of damage.
   I. Radio altimeter and voice recorder (if installed) for security of mounting.

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INSPECTION/CHECK (Continued)

INSPECT — FLIGHT AND PASSENGER COMPARTMENT

Equipment/Tools

- None

Consumables

- None

Inspection

1. Inspect the following:

FLIGHT COMPARTMENT

A. Instrument panels for security and placards for legibility.

B. Thunderstorm and console floodlights for cracks, broken lenses and security.

C. Utility light housings for damage, security, attachment and placards for legibility.

D. Circuit breakers for damage, security, attachment and placards for legibility.

E. Instrument cover glasses for cracks, slippage marks for alignment, limit and operating range marks for legibility.

F. Standby compass fluid for discoloration, leakage and legibility of markings. Correction cards for legibility.

G. Panel floodlights, caution, warning and mode lights for cracked lenses and broken covers.

H. Center pedestal for security, legibility of markings and cleanliness.

I. Overhead switch panels for security and legibility of markings.

J. Pilot's and co-pilot's seats for security and freedom of operation. Seat belts for general condition and pull test date. Arm rests for damage.

K. Shoulder harness for general condition and inertia reel for proper operation.

L. Inspect oxygen regulators for damage, security and cleanliness.

M. Oxygen masks for general condition, kinked hoses and cleanliness.

N. Portable fire extinguisher for security and proper charge. Check last weight date of bottle.

O. Flight compartment for general cleanliness condition.

P. Ventilation nozzles for proper operation and for obstruction.

Q. Sliding window for proper operation and seating.

R. Decals, placards and markings for legibility.

S. Instrument lines, wire bundles, and flex hoses forward of instrument panel for condition, security and interference.

ENTRANCE WAY

A. Window and escape hatches dehydrator capsules — desiccant for evidence of moisture (deep blue — dry condition, opaloid or bright pink — moisture saturated).

B. On S/N 380-1 and Subs, the electrical equipment rack components for security, cleanliness, broken or corroded connectors, and accumulations of dirt, lint and other foreign material.

C. Equipment and furnishings for condition and security. Placards and markings for legibility.

D. Forward and aft avionics equipment rack components (if installed) for security of mounting, damaged or corroded electrical connectors or wires, and accumulations of lint or dirt at fan inlets and outlets.

E. Avionics compartment seals for evidence of damage and/or improper sealing.

F. Galley area for security of components and cleanliness of area.

G. Lavatory compartment for cleanliness and adequate supplies.

PASSENGER COMPARTMENT

A. Interior doors for binding and defective latch mechanism.

B. Passenger seats and seat belts for condition and operation.

C. Fire extinguishers (if installed) for condition, security, specified pressure and seal intact. Check last weight date of bottle.

D. Emergency escape hatch dehydrator capsules for dry condition (deep blue). Service as required. Check escape hatch soundproofing for moisture and area for corrosion.

E. Emergency exits secured.

F. Cabin and personnel warning signs for condition and operation.

G. Equipment and furnishings for condition, security and operation.

H. Markings and required placards for condition and legibility.

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INSPECTION/CHECK (Continued)

INSPECT — LOWER FORWARD FUSELAGE INTERIOR — Fus. Sta. 182 to 242

Equipment/Tools

- None

Consumables

- None

Inspection

1. Open appropriate access panels and inspect the following:
   - A. Wing attach fittings and structures, cap angles and intercostals, for cracks, deformation, sheared or pulled rivets.
   - B. Accessible wire runs in areas for overheating, chafing, proper support, and clearance from structure.
   - C. Speed brake dump valve for evidence of leakage and security.
   - D. Speed brake control valve for evidence of leakage and security, flex hoses for evidence of leakage.
   - E. Water separator bowl drain hole and line drain hole for obstructions (if applicable).
   - F. On S/N 370 airplanes, lower batteries; battery tray lift mechanism for ease of operation.
     - a. Battery cables for corrosion, broken strands; latches for damage and security.
     - b. Batteries for excessive heat; cases and covers for damage and distortion; external electrical connectors for burned or damaged contacts.
     - c. Battery and sump jar vents for obstructions, binding and corrosion.
     - d. Batteries and sump jar for evidence of leakage or overflow of electrolyte.
     - e. Batteries and sump jar for corrosion, pad for proper saturation.
     - f. Raise and secure batteries.
   - G. Close access panels.

INSPECT — FUSELAGE EXTERIOR

Equipment/Tools

- None

Consumables

- None

Inspection

1. Inspect the following:
   - A. Fuselage skin for cracks, distortion, corrosion and defective rivets or screws.
   - B. Radome for cracks, dents, cleanliness, and condition of rain erosion coating.
   - C. Inspect pitot heads for obstructions, condition and security.
   - D. Windshields for evidence of overheating, burned spots and cracking of inner or outer panels.
   - E. Cockpit side windows and overhead windows for cracking of inner or outer panels; moisture between panels.
   - F. Windshield wiper blades for deterioration and arms for security.
   - G. Antennas for erosion, cracks, dents and security.
   - H. Undersurface of fuselage for evidence of fuel or hydraulic leakage.
   - I. Entrance door skin and structure for cracks, distortion and defective rivets or screws, and check fairing door for loose hinge points.
   - J. Entrance door for free movement through range of travel, positive locking and proper alignment when closed and secondary seal for damage. Door strip seals for security.
   - K. Anti-collision lights for broken, cracked or fogged lenses and for loose or missing screws.
   - L. Speed brake panel and well for cracks, distortion and defective rivets or screws.
   - M. Speed brake panel attach points for looseness.
   - N. Speed brake actuators for evidence of leakage, piston rods for scoring and pitting and hoses for chafing and twisting.
   - O. Speed brake limit switch for corrosion and security.
   - P. Engine start external power receptacle for cleanliness, cracks or bent pins. Check microswitch arm for free operation, bending and excessive wear.
   - Q. Cabin windows for cracking of inner or outer panels; moisture between panels.
   - R. Cabin emergency exits secured.
   - S. Utility external power receptacle for corrosion, cracks or bent pins.

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INSPECTION/CHECK (Continued)

INSPECT — FUSELAGE EXTERIOR (Cont'd)

Inspection (Cont'd)

T. Hinged access doors for damage and hinge wear. Check camloc fasteners for worn studs and grommets and snap rings for security.

U. Lightning diverter rods for security and bent or broken rods.

V. Acrylic tail cone for cracks and crazing.

W. Fuel vent outlet screen and on NA-370-2 through -6, -8 and -9 airplanes, the jettison line for obstruction.

X. Painted surfaces for scratches, chipping and cleanliness.

Y. Inspect APU exhaust duct and surrounding area for cracks, deep scratches and distortion.

Z. Required markings for legibility.

INSPECT — FUSELAGE NOSE SECTION INTERIOR

Equipment/Tools

- None

Consumables

- None

Inspection

1. Open appropriate access panels and inspect the following:
   A. Communication equipment for security, evidence of overheating, chafing and applicable safeties.
   B. Communication equipment supporting racks for distroted mounts, cracks and applicable safeties.
   C. Nose steering and windshield temperature control boxes for security and proper safeties.
   D. Electrical wiring terminal boards, transformers, relays and electrical connectors for evidence of overheating, chafing, evidence of overheating and security.
   E. Inventors for security, broken mounts, exhaust opening for obstructions and electrical cables for chafing, evidence of overheating and security.
   F. Ground cooling blower for security, inlet and exhaust opening for obstructions.
   G. All accessible structure and skin for distortion, cracks, corrosion and missing fasteners.
   H. Access panel seals for distortion, cracks, corrosion, tears and security.
   I. On S/N 380-1 and Subs, batteries for security and overheat damage. Battery sump jar for proper servicing, security and cracks. Sump jar hoses and vents for obstructions.
   J. Battery cables for corrosion or broken strands. Battery external electrical connectors for burned or damaged contacts.
   K. Install access panels.

INSPECT — LOWER FUSELAGE INTERIOR — Fus. Sta. 264 to 357

Equipment/Tools

- None

Consumables

- None

Inspection

1. Open appropriate access panels and inspect the following:
   A. Accessible fire extinguisher lines for chafing, security of connections; agent containers for security and specified pressure.
   B. Wire runs for chafing, proper support, and clearance from structure.
   C. Fuel and hydraulic lines in areas for chafing, evidence of leakage and security.
   D. On S/N 370-2 through -6, -8 and -9 airplanes, fuel jettison valves for evidence of leakage and security.
   E. Hydraulic system relief valve, system return filter, auxiliary shut-off valve for evidence of leakage and security.
   F. Accessible structure and skins for cracks, corrosion, and evidence of failure.
   G. Close access panels.

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INSPECTION/CHECK (Continued)

INSPECT — UPPER LONGERON SPLICE F.S. 206 LH AND RH.

Equipment/Tools

- None

Consumables

- Sealant PR 1422 GB

Inspection

1. Remove interior to gain access to FS 206 upper longeron splices LH and RH. See Figure 53-00-601.
2. Remove sealant on FS 206 upper longeron splice LH and RH .
3. Inspect for cracks and/or corrosion.

NOTE
Any repairs accomplished should be made in accordance with Sabreliner Structural Repair Manual (NA-66-1032) and/or Sabreliner Corrosion Control Manual (SR-76-23).

4. Apply (PR 1422 GB) Sealant per Sabreliner Corrosion Control Manual (SR-76-23), chapter 20.

NOTE
Recommend a review of pending additional interior compliance items before installing interior.

5. Perform Cabin Pressurization Control System Check per Sabreliner Maintenance Manual (NA-69-420), chapter 21.
6. Install Interior.

INSPECT — ALL CABIN FLOORS, FLOOR SUPPORT BEAMS, AND LOWER MAIN INTERMEDIATE LONGERONS LH AND RH.

Equipment/Tools

- None

Consumables

- Sealant PR 1422 GB
- Tape P32-2

Inspection

1. Remove cabin seats, carpet, insulation, heater pads, etc. to gain access to cabin floor.
2. Remove LH and RH flight control tunnels (See Figure 53-00-602).
3. Remove cabin floor panels as necessary and LH and RH close out fillet skin at FS 206 to 257.56 (P/N 265-300501-5 and -6 LH and RH) to gain access to floor support beams and lower main intermediate longerons.

NOTE
Do not remove sealant unless sealant is cracked, split, bubbled, or corrosion damage is evident in the area.

4. Inspect longerons and floor support beams for cracks and/or corrosion.
5. Apply sealant (PR 1422 GB) per Sabreliner Corrosion Control Manual (SR-76-23), chapter 20.
6. Inspect cabin floor panels for cracks and/or corrosion.

NOTE
Any repairs accomplished should be made in accordance with Sabreliner Structural Repair Manual (NA-66-1032) and/or Sabreliner Corrosion Control Manual (SR-76-23).

7. Install floor panels and tape all joints with P32-2 Tape.
8. Install LH and RH flight control tunnels.
9. Install LH and RH close out fillet skin.

NOTE
Recommend a review of pending additional interior compliance items before installing interior.

10. Perform Cabin Pressurization Control System Check per Sabreliner Maintenance Manual (NA-69-420), chapter 21.
11. Install cabin interior.

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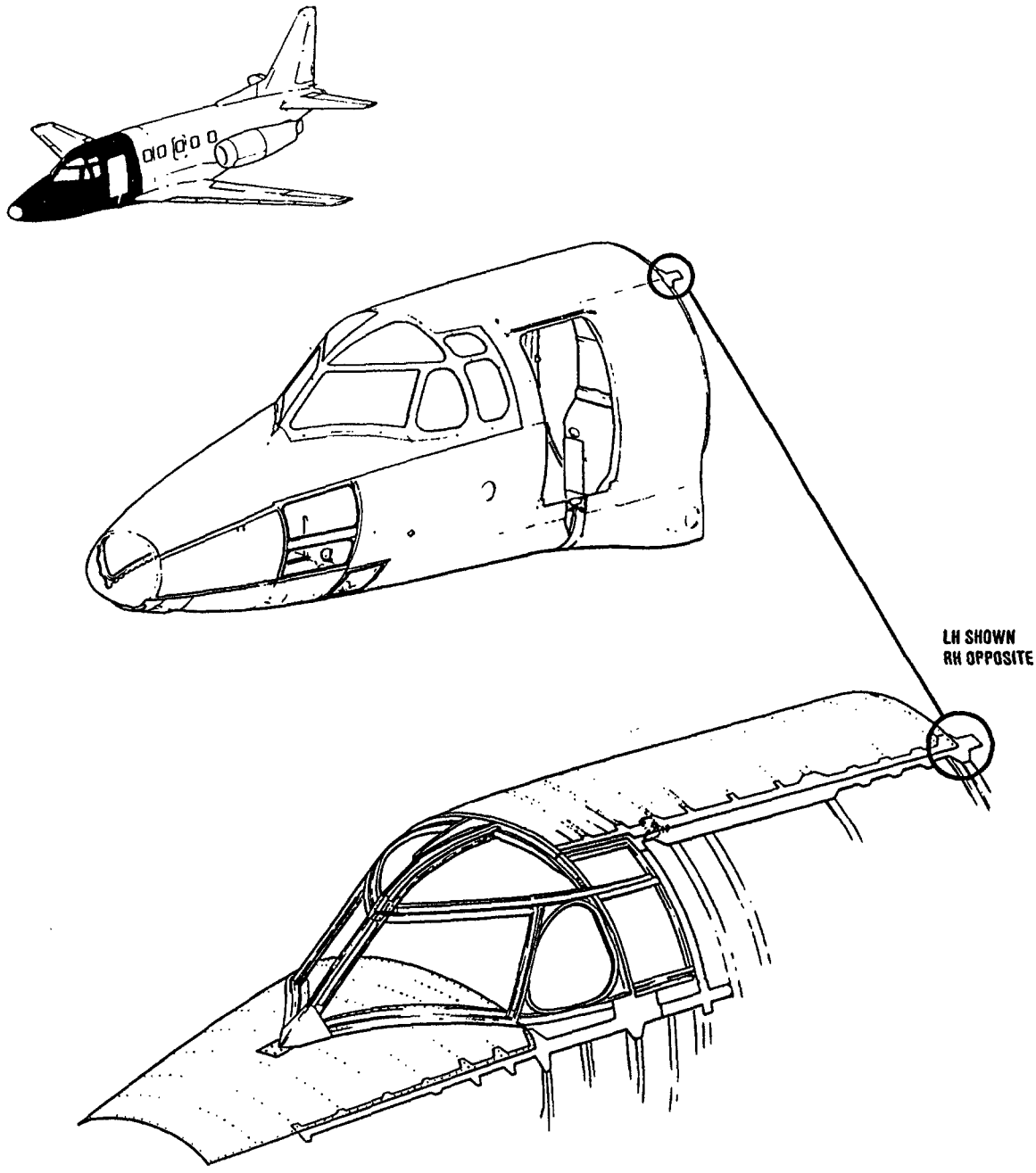

NOTE
Look for flaws in the paint, skin and fasteners in all areas as an indication of corrosion.

Figure 53-00-601 — Inspect — Upper Longeron Splices F.S. 206, LH and RH

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NOTE

Remove fillets above wing and center floor panels.

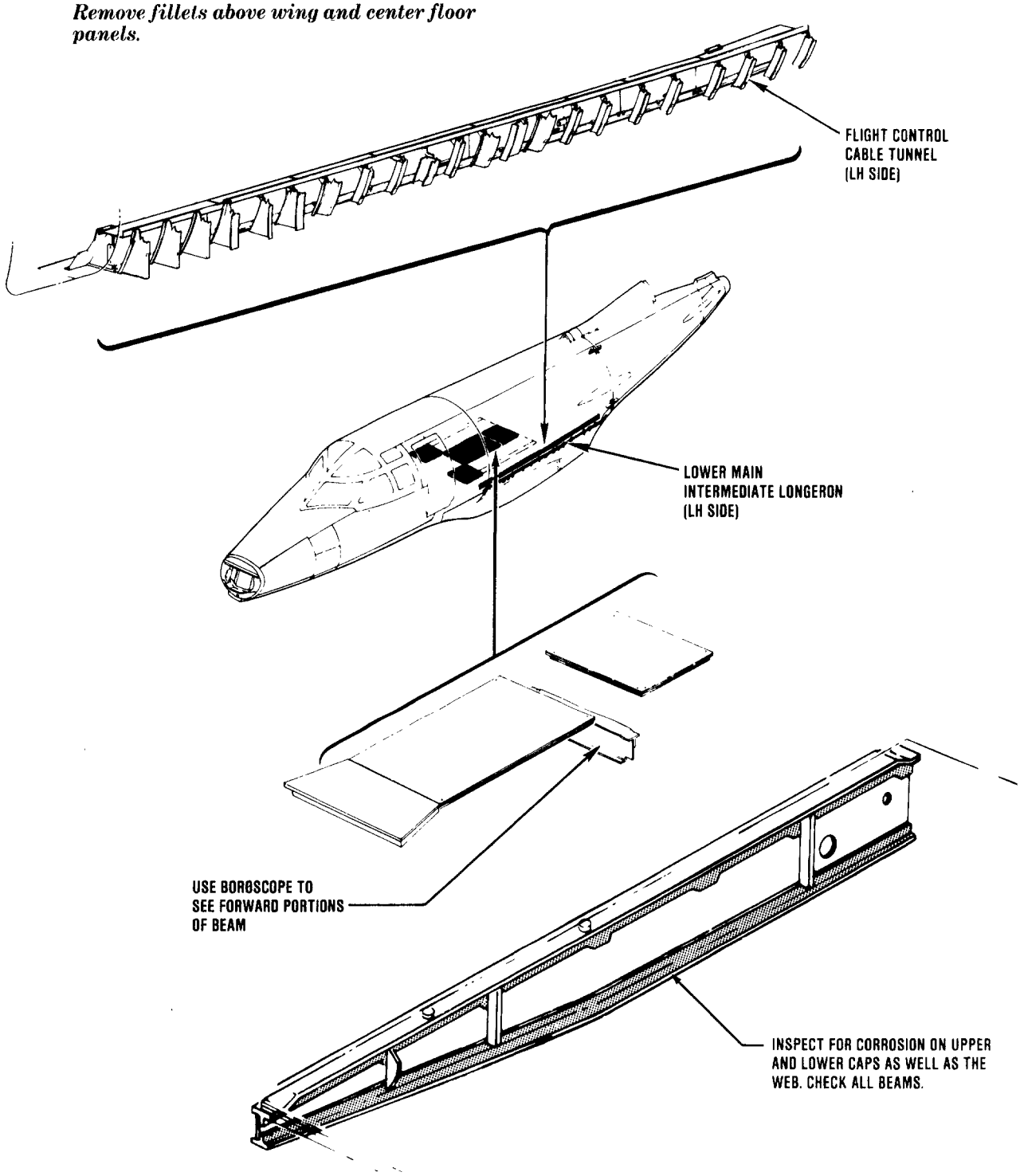
FLIGHT CONTROL CABLE TUNNEL (LH SIDE)
LOWER MAIN INTERMEDIATE LONGERON (LH SIDE)
USE BORESCOPE TO SEE FORWARD PORTIONS OF BEAM
INSPECT FOR CORROSION ON UPPER AND LOWER CAPS AS WELL AS THE WEB. CHECK ALL BEAMS.

Figure 53-00-602 — Inspect — All Cabin Floors, Floor Support Beams, and Lower Main Intermediate Longerons

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INSPECTION/CHECK (Continued)

INSPECT — STRAP BELOW ESCAPE HATCHES

Equipment/Tools

- None

Consumables

- Sealant PR 1422 GB

Inspection

1. Remove cabin interior.

NOTE

Do not remove sealant unless sealant is cracked, split, bubbled, or corrosion damage is evident in the area.

2. Inspect strap for corrosion. (See Figure 53-00-603.)

NOTE

Any repairs accomplished should be made in accordance with Sabreliner Structural Repair Manual (NA-66-1032) and/or Sabreliner Corrosion Control Manual (SR-76-23).

3. Apply sealant (PR 1422 GB) per Sabreliner Corrosion Control Manual (SR-76-23), chapter 20.

NOTE

Recommend a review of pending additional interior compliance items before installing interior.

4. Perform cabin pressurization control system check per Sabreliner Maintenance Manual (NA-69-420), chapter 21.
5. Install Interior.

INSPECT — AILERON BELLCRANK SUPPORT AND PULLEY SUPPORT — LANDING GEAR EMERGENCY RELEASE

Equipment/Tools

- None

Consumables

- None

Inspection

1. Remove cabin interior.
2. Inspect bellcrank support and emergency release pulley support for cracks and/or corrosion. (See Figure 53-00-604.)

NOTE

Any repairs accomplished should be made in accordance with Sabreliner Structural Repair Manual (NA-66-1032) and/or Sabreliner Corrosion Control Manual (SR-76-23).

NOTE

Recommend a review of pending additional interior compliance items before installing interior.

INSPECT — ADF LOOP AND ADF SENSE ANTENNA SUPPORT STRUCTURE AND UPPER MAIN INTERMEDIATE LONGERONS

Equipment/Tools

- None

Consumables

- Sealant PR 1422 GB

Inspection

1. Remove upper cabin interior.

NOTE

Do not remove sealant unless sealant is cracked, split, bubbled, or corrosion damage is evident in the area.

2. Remove sealant as required around ADF loop antenna support structure, ADF sense antenna support structure, upper main intermediate longerons and surrounding areas. (See Figure 53-00-605.)
3. Inspect for cracks and/or corrosion.

NOTE

Any repairs accomplished should be made in accordance with Sabreliner Structural Repair Manual (NA-66-1032) and/or Sabreliner Corrosion Control Manual (SR-76-23).

4. Apply sealant (PR 1422 GB) per Sabreliner Corrosion Control Manual (SR-76-23), chapter 20.

NOTE

Recommend a review of pending additional interior compliance items before installing interior.

5. Perform cabin pressurization control system check per Sabreliner Maintenance Manual (NA-69-420), chapter 21.
6. Install interior.

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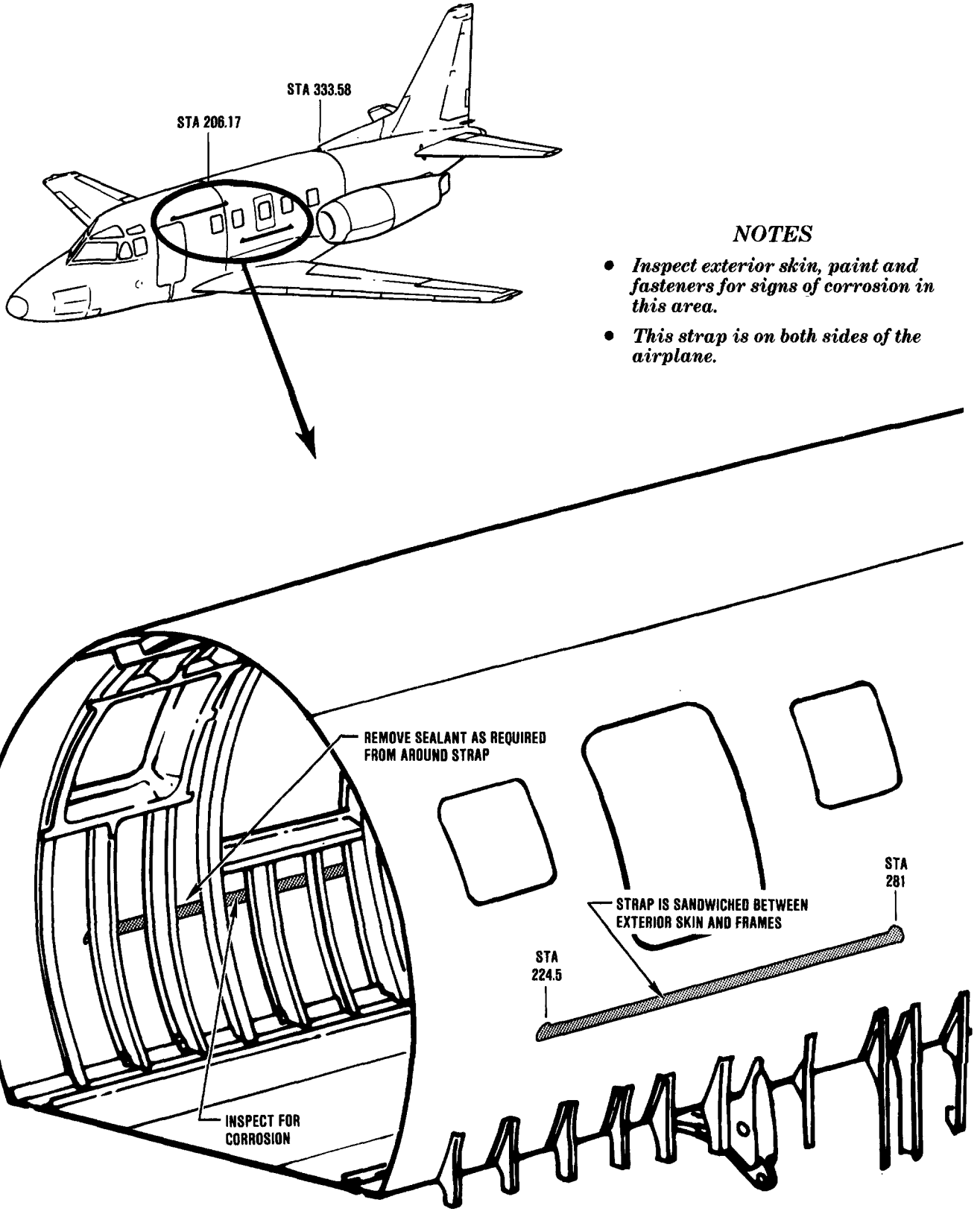
STA 333.58
STA 206.17
NOTES
• Inspect exterior skin, paint and fasteners for signs of corrosion in this area.
• This strap is on both sides of the airplane.
REMOVE SEALANT AS REQUIRED FROM AROUND STRAP
STA 281
STRAP IS SANDWICHED BETWEEN EXTERIOR SKIN AND FRAMES
STA 224.5
INSPECT FOR CORROSION

Figure 53-00-603 — Inspect — Strap Below the Escape Hatches

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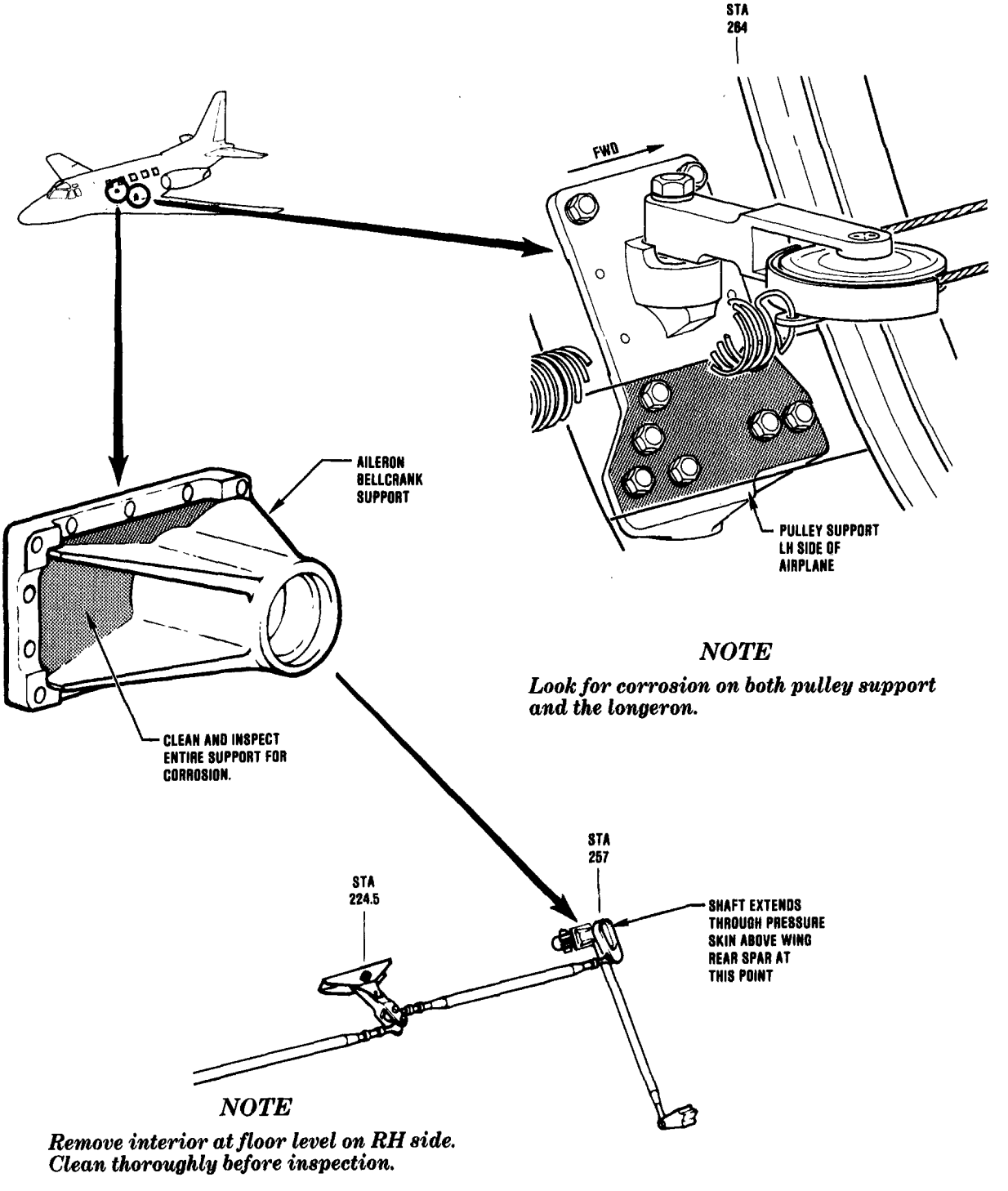

NOTE

Look for corrosion on both pulley support and the longeron.

NOTE

Remove interior at floor level on RH side. Clean thoroughly before inspection.

Figure 53-00-604 — Inspect — Aileron Bellcrank Support and Pulley Support — Landing Gear Emergency Release

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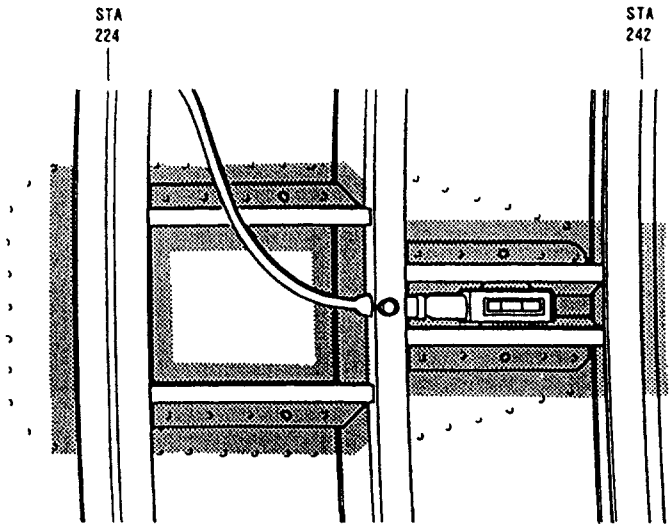

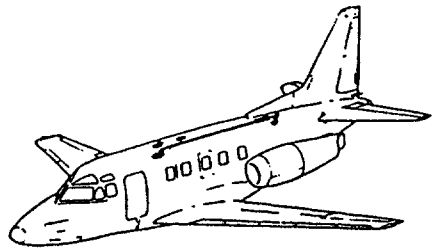

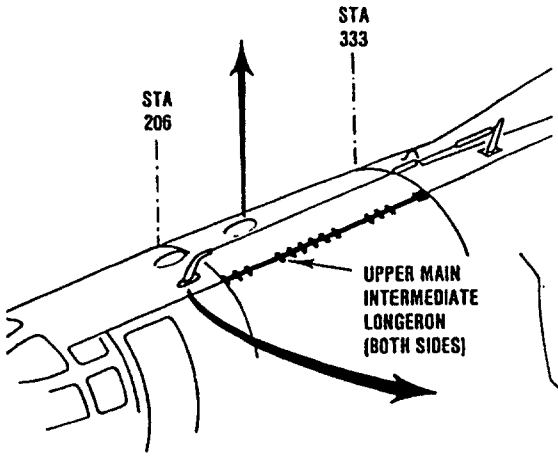

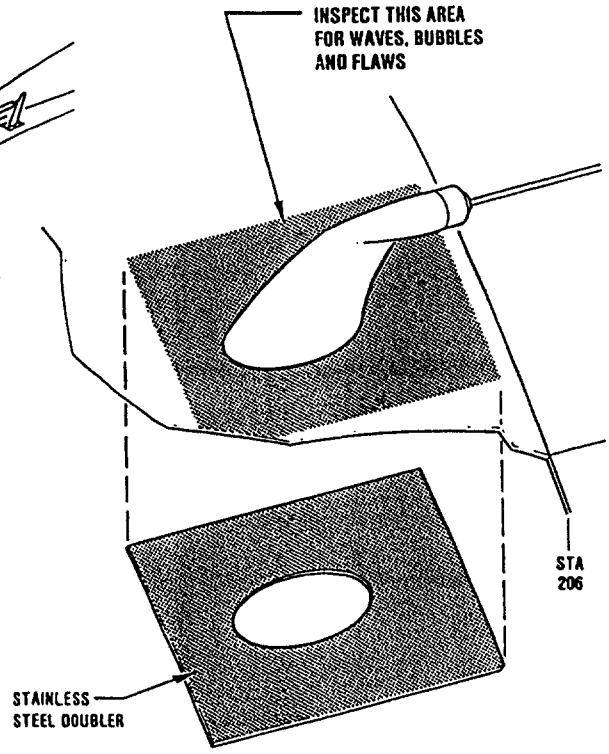

NOTES

- *Remove interior in area of antenna structure.*
- *Do not remove sealant unless sealant is cracked, split, bubbled, or corrosion damage is evident in the area.*

Figure 53-00-605 — Inspect — ADF Loop and ADF Sense Antenna Support Structure and Upper Main Intermediate Longerons

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INSPECTION/CHECK (Continued)

INSPECT — HORIZONTAL STIFFENERS ON FWD SIDE OF F.S. 333 AND BULKHEAD TO SIDE SKIN ATTACH ANGLE

Equipment/Tools

- None

Consumables

- Sealant PR 1422 GB

Inspection

1. Remove interior from in front of the F.S. 333 bulkhead.
2. Remove sealant and inspect for cracks and/or corrosion. (See Figure 53-00-606.)

NOTE

Any repairs may require Aux Fuel Cell removal.

NOTE

Any repairs accomplished should be made in accordance with Sabreliner Structural Repair Manual (NA-66-1032) and/or Sabreliner Corrosion Control Manual (SR-76-23).

3. Apply sealant (PR 1422 GB) per Sabreliner Corrosion Control Manual (SR-76-23), chapter 20.

NOTE

Recommend a review of pending additional interior compliance items before installing interior.

4. Perform cabin pressurization system check per Sabreliner Maintenance Manual (NA-69-420), chapter 21.
5. Install interior.

INSPECT — WEB OF BULKHEAD F.S. 298.5

Equipment/Tools

- None

Consumables

- As required per Sabreliner Corrosion Control Manual (SR-76-23), chapter 20

Inspection

1. Remove Main Landing Gear wheel well tubs. (See Figure 53-00-607.)
2. Clean bulkhead per Sabreliner Corrosion Control Manual (SR-76-23), chapter 20.
3. Inspect bulkhead at FS 298.5 for cracks and/or corrosion. (See Figure 53-00-607.)

NOTE

Any repairs accomplished should be made in accordance with Sabreliner Structural Repair Manual (NA-66-1032) and/or Sabreliner Corrosion Control Manual (SR-76-23).

4. Install Main Landing Gear wheel well tubs.

INSPECT — LOWER FLANGE AND SKIN F.S. 13 TO 76, LOWER CAP ANGLE OF NOSE GEAR BEAM B.P. 9 AND LOWER ELECTRONICS BAY

Equipment/Tools

- Adjustable Flex Shaft Boroscope and Power Supply

Consumables

- None

Inspection

1. Remove doors for inspection access. (See Figure 53-00-608.)
2. Using a boroscope, inspect the bottom flange, both sides of the support structure, and both bottom sides of the electronics bay and the fwd longerons for cracks and/or corrosion. (See Figure 53-00-608.)

NOTE

Any repairs accomplished should be made in accordance with Sabreliner Structural Repair Manual (NA-66-1032) and/or Sabreliner Corrosion Control Manual (SR-76-23).

3. Install doors removed in step 1.

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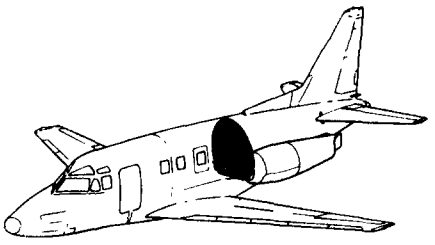

NOTES

- *Remove interior from in front of the bulkhead.*
- *Remove sealant from shaded areas and inspect for corrosion around horizontal channels, cap angle and skin.*

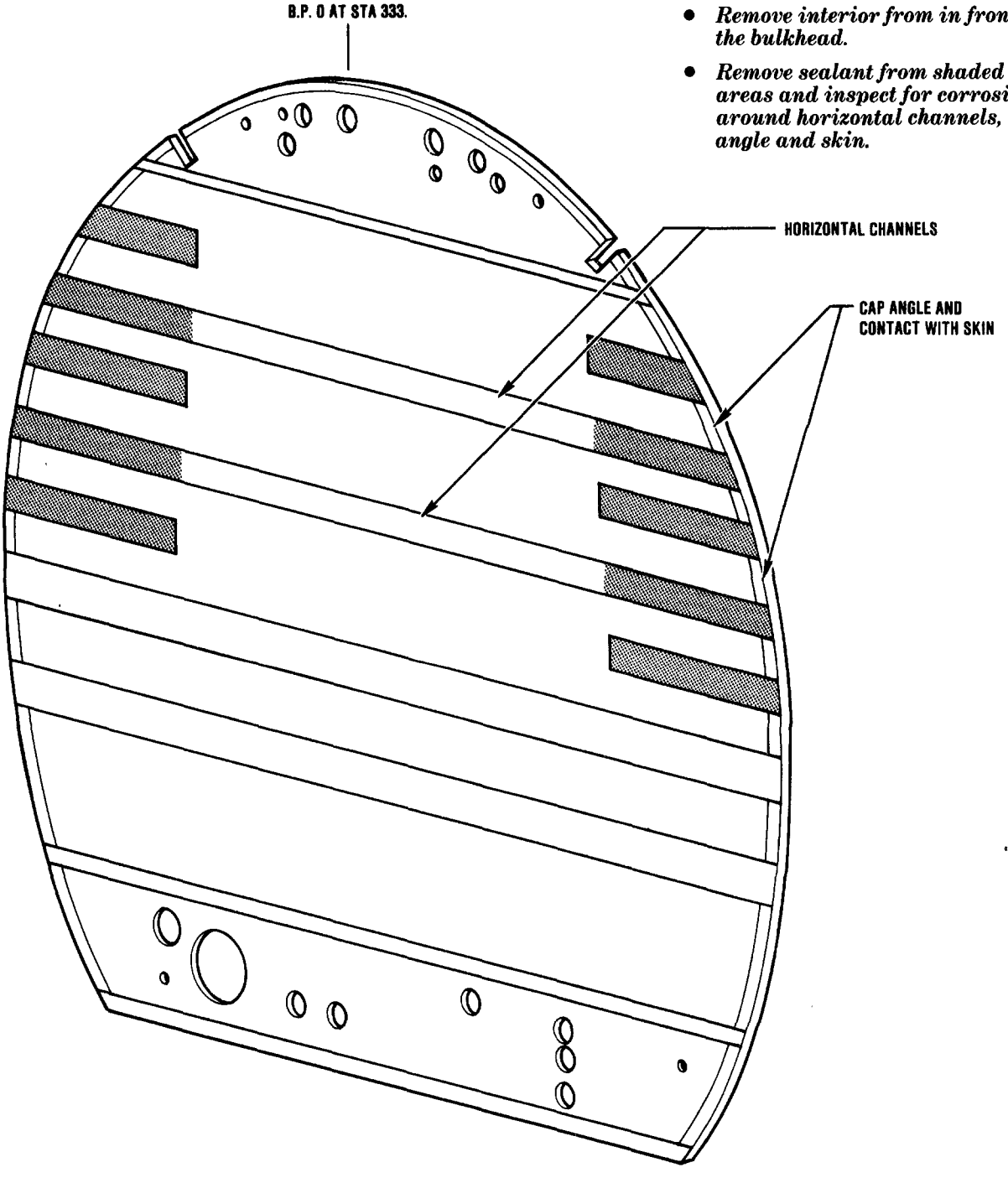

Figure 53-00-606 — Inspect — Horizontal Stiffeners on Fwd Side of F.S. 333 and Bulkhead to Side Skin Attach Angle

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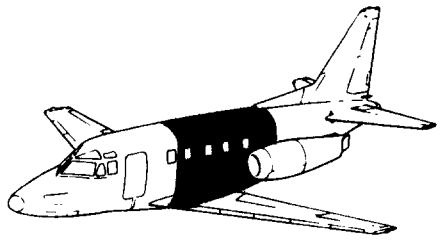

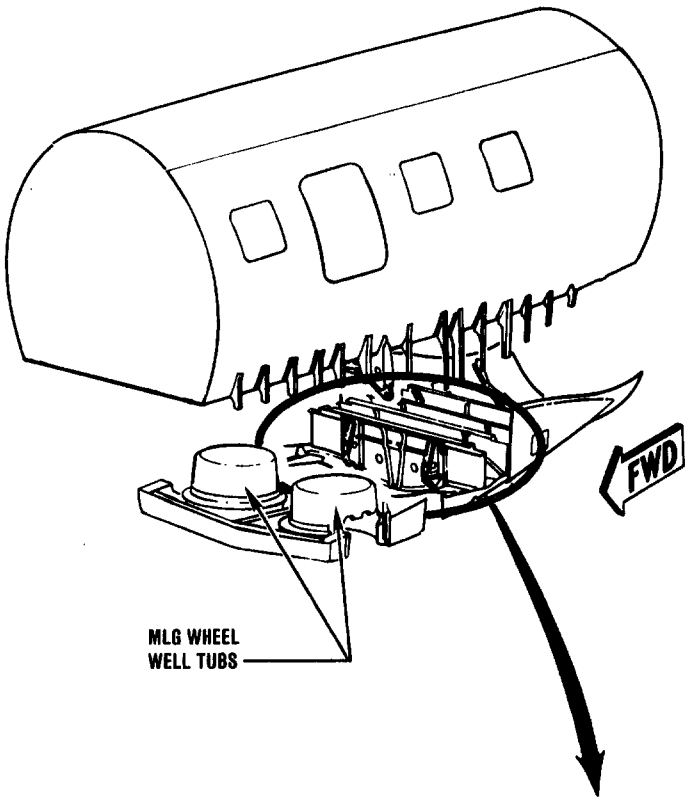

NOTES

- *The bulkhead must be cleaned before inspection.*
- *Inspect forward and aft sides of entire assembly for corrosion.*

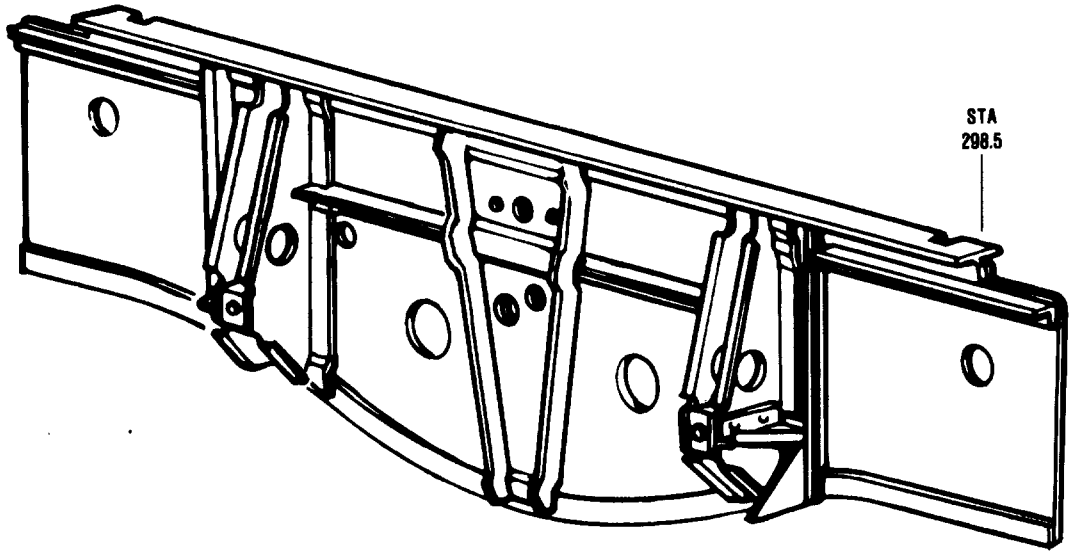

Figure 53-00-607 — Inspect — Web of Bulkhead F.S. 298.5

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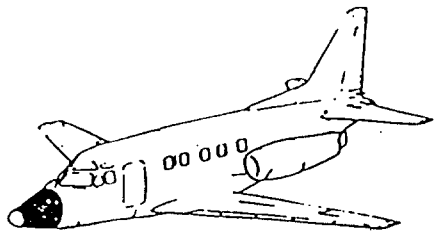
NOTES

- *Clean areas before inspection.*
- *Inspect for cracks and corrosion especially in shaded areas.*



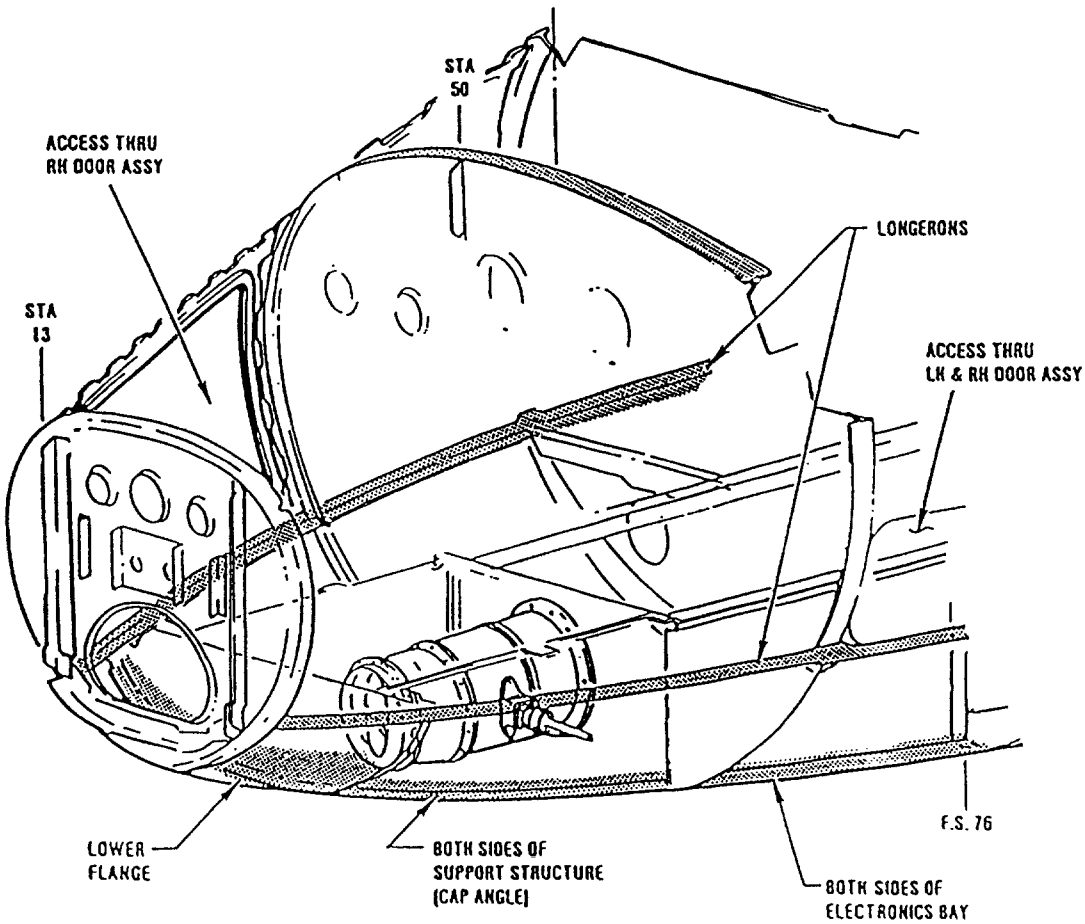


Figure 53-00-608 — Inspect — Lower Flange and Skin F.S. 13 to 76, Lower Cap Angle of Nose Gear Beam B.P. 9 and Lower Electronics Bay

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INSPECTION/CHECK (Continued)

INSPECT — SPLICE BETWEEN LOWER CENTER SKIN AND SIDE SKIN F.S. 99 TO 156.25 LH AND RH

Equipment/Tools

- None

Consumables

- None

Inspection

1. Visually inspect splice area for signs of cracks, corrosion, waves, bubbles, loose or missing fasteners. (See figure 53-00-609.)

NOTE

Any repairs accomplished should be made in accordance with Sabreliner Structural Repair Manual (NA-66-1032) and/or Sabreliner Corrosion Control Manual (SR-76-23).

INSPECT — SKIN SPLICE ON INTERMEDIATE LONGERON LH AND RH SIDE. INSPECT LONGERON ABOVE MAIN ENTRANCE DOOR. INSPECT LH AND RH LOWER LONGERON.

Equipment/Tools

- None

Consumables

- None

Inspection

1. Visually inspect skins against longerons for signs of cracks and corrosion, waves, bubbles, loose or missing fasteners. (See Figure 53-00-610.) Inspect exterior seams.

NOTE

Any repairs accomplished should be made in accordance with Sabreliner Structural Repair Manual (NA-66-1032) and/or Sabreliner Corrosion Control Manual (SR-76-23).

INSPECT — SKIN SPLICE AT F.S. 143 LOWER SIDE SKIN LH AND RH SIDE, COCKPIT GLASS SUPPORT STRUCTURE, WINDSHIELD BOWS, INTERMEDIATE LONGERONS AND SKIN SPLICE LH AND RH AND COCKPIT FLOORS AND LOWER LONGERONS LH AND RH SIDE.

Equipment/Tools

- Adjustable Flex Shaft Boroscope and Power Supply

Consumables

- Sealant PR 1422 GB

Inspection

1. Remove both cockpit seats, carpet, heater pads and insulation.
2. Remove access panels, F.S. 132 to F.S. 140 LH and RH.
3. Remove all window reveals.
4. Remove closeout panels between windows and side circuit breaker panels, left and right sides.
5. Remove cockpit floors.
6. Inspect drains in cockpit area.
7. Inspect Lower Longeron interiors surfaces.
8. Inspect cockpit floor panels for corrosion.
9. Inspect F.S. 143 bulkhead LH and RH for cracks and corrosion.
10. Visually inspect the inside cockpit window frame structure for signs of loose or missing fasteners, bubbles in the paint, cracks and corrosion. Inspection of the window frame outer structure should be accomplished during window repair or replacement (See Figure 53-00-611.)

NOTE

Do not remove sealant unless sealant is cracked, split, bubbled, or corrosion damage is evident in the area.

11. Visually inspect entire aft surface of windshield bow for cracks, using 4x magnifying glass. See Figure 53-00-621 (Sheet 2).
12. If crack damage is present, remove and replace support with new part.
13. Install new support in accordance with Figure 53-00-621 (Sheet 3) criteria.
14. Inspect intermediate longeron LH and RH for cracks and corrosion.
15. Inspect stainless steel skin splice at F.S. 143, both sides of fuselage.

NOTE

Any repairs accomplished should be made in accordance with Sabreliner Structural Repair Manual (NA-66-1032) and/or Sabreliner Corrosion Control Manual (SR-76-23).

16. Apply sealant (PR 1422 GB) per Sabreliner Corrosion Control Manual (SR-76-23), chapter 20.
17. Install cockpit floors, access panels, window reveals, all closeout panels.
18. Perform cabin pressurization control system check per Sabreliner Maintenance Manual (NA-69-420), chapter 21.
19. Install cockpit interior and seats.

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NOTES

- *Each splice is sandwiched between the exterior skin and the inboard structure.*
- *Inspect for waves, bubbles and flaws in skin and paint system. Check for loose, missing and corroded fasteners.*

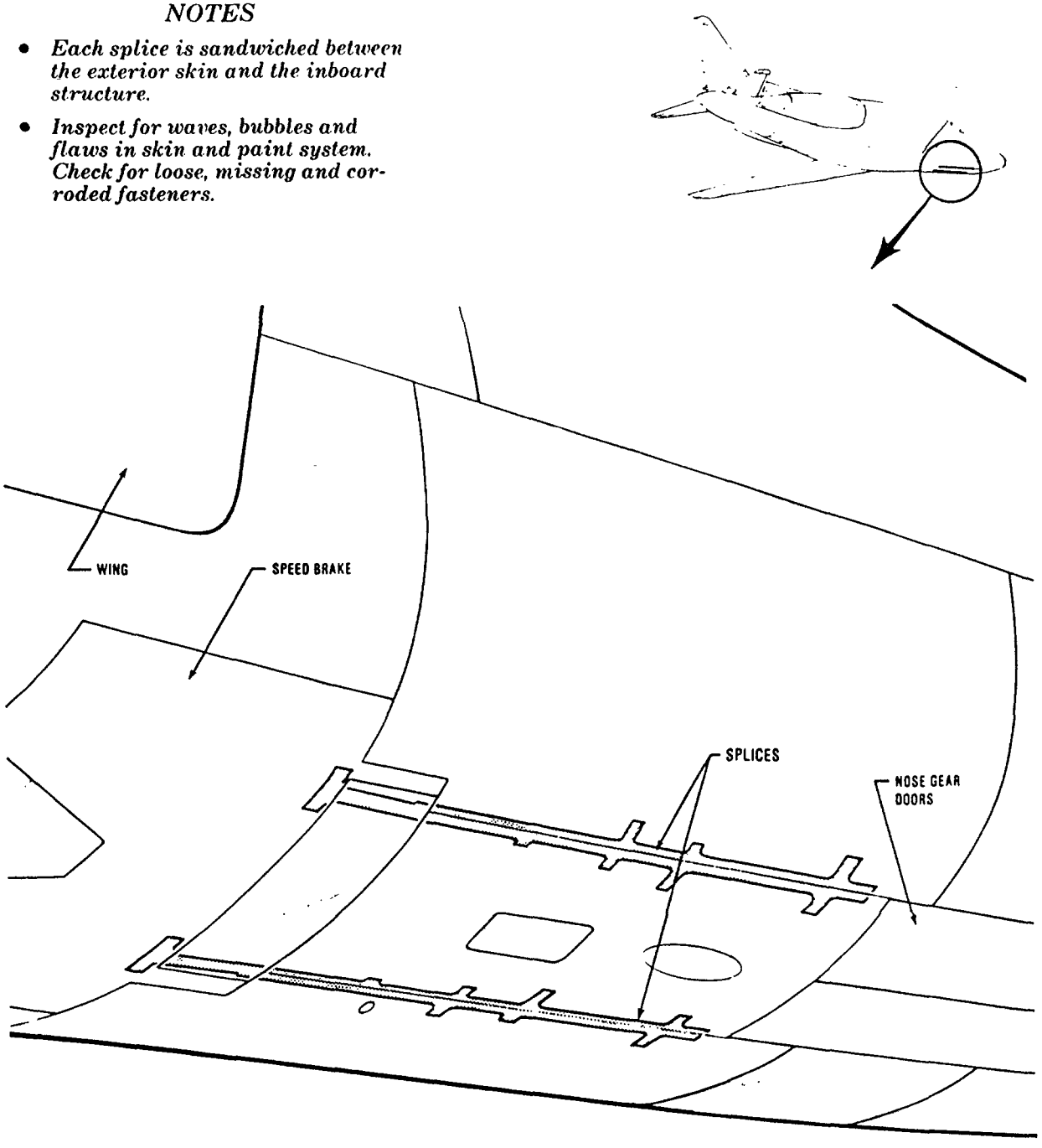

Figure 53-00-609 — Inspect — Splices Between Lower Center Skin and Side Skin F.S. 99 to 156.25 LH and RH

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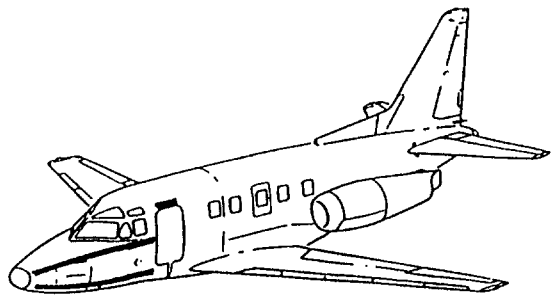
NOTE

Look for flaws in paint, skin and fasteners in all these areas as an indication of corrosion..



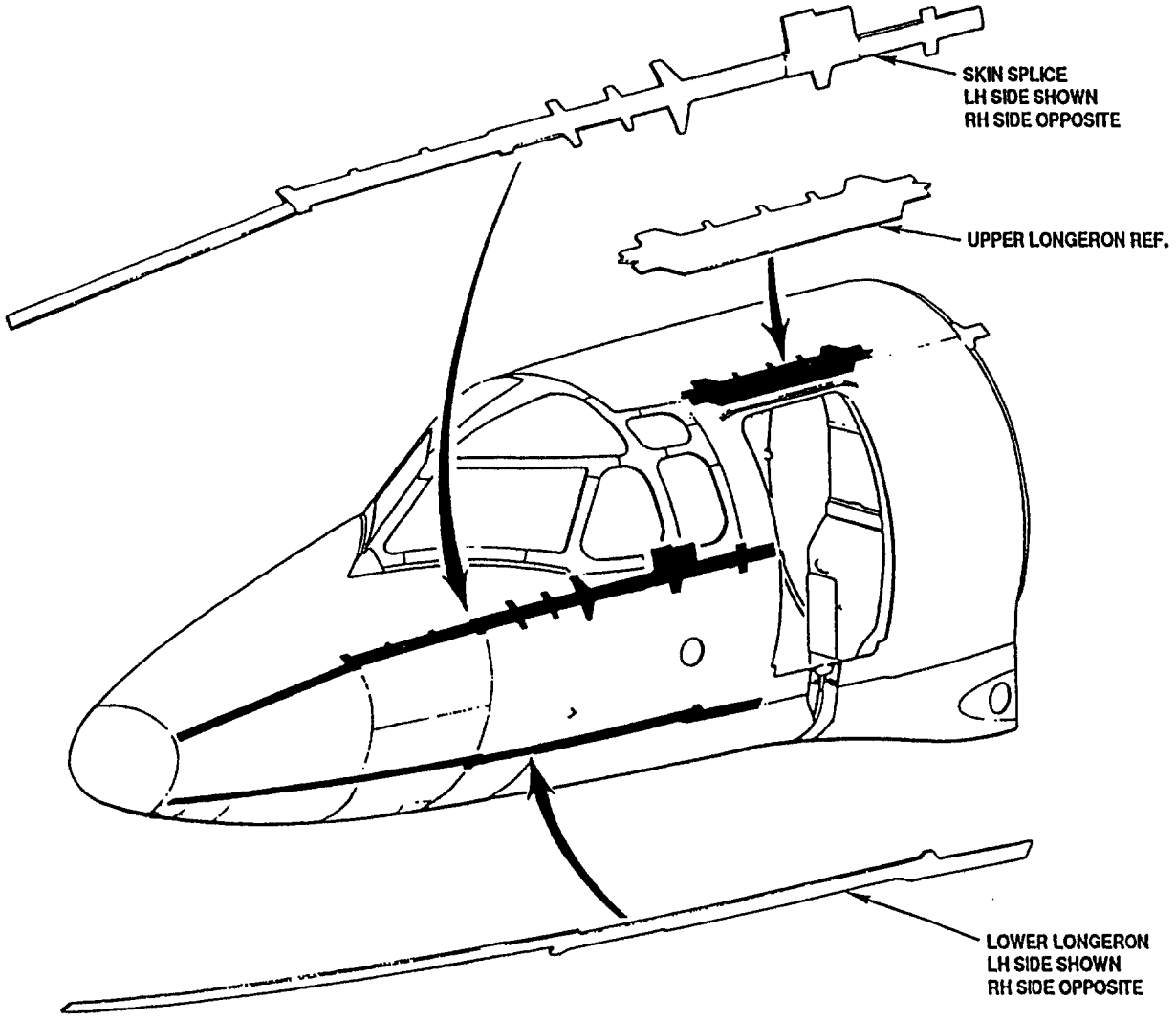


Figure 53-00-610 — Inspect — Skin Splice on Intermediate Longeron LH and RH Side. Inspect Longeron Above Main Entrance Door. Inspect Lower Longeron LH and RH Side.

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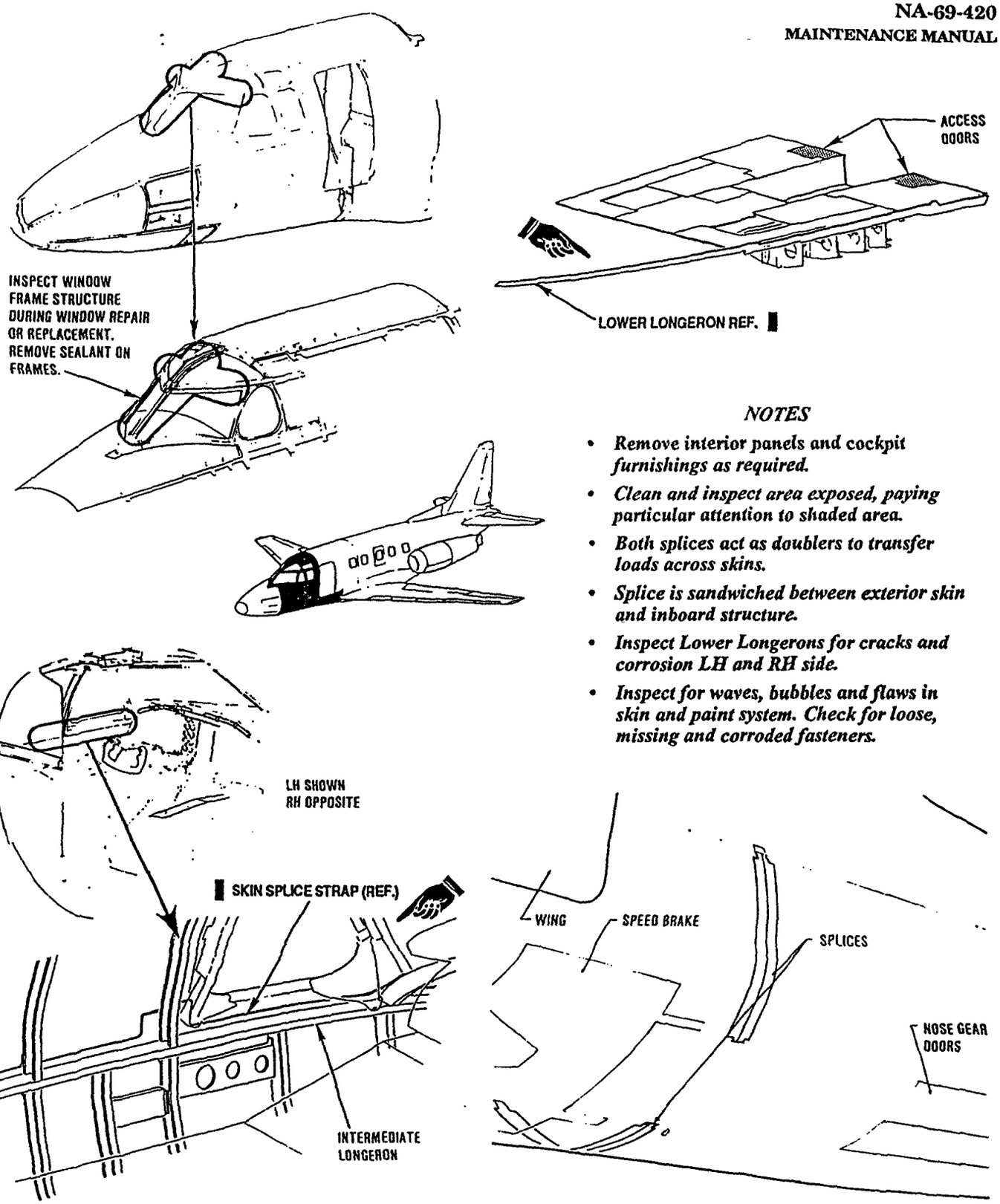

NOTES

- *Remove interior panels and cockpit furnishings as required.*
- *Clean and inspect area exposed, paying particular attention to shaded area.*
- *Both splices act as doublers to transfer loads across skins.*
- *Splice is sandwiched between exterior skin and inboard structure.*
- *Inspect Lower Longerons for cracks and corrosion LH and RH side.*
- *Inspect for waves, bubbles and flaws in skin and paint system. Check for loose, missing and corroded fasteners.*

Figure 53-00-611 — Inspect — Skin Splice at F.S. 143 Lower Side Skin, Cockpit Glass Support Structure, Intermediate Longerons LH and RH, Skin Splice LH and RH Side, Cockpit Floors, Lower Longerons LH and RH Side

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INSPECTION/CHECK (Continued)

INSPECT — LOWER LONGERON ALONG THRESHOLD OF MAIN ENTRANCE DOOR AND LOWER DOOR STOP

Equipment/Tools

- None

Consumables

- Sealant PR 1422 GB

Inspection

1. Open main entrance door.

NOTE

Do not remove sealant unless sealant is cracked, split, bubbled, or corrosion damage is evident in the area.

2. Clean area per Sabreliner Corrosion Control Manual (SR-76-23), chapter 20.
3. Inspect for cracks and corrosion. (See Figure 53-00-612.)

NOTE

Any repairs accomplished should be made in accordance with Sabreliner Structural Repair Manual (NA-66-1032) and/or Sabreliner Corrosion Control Manual (SR-76-23).

4. Apply Sealant (PR 1422 GB) per Sabreliner Corrosion Control Manual (SR-76-23), chapter 20.

INSPECT — FLOOR SUPPORT BEAM F.S. 264 AND WING ATTACH FITTINGS LH AND RH ON FUSELAGE

Equipment/Tools

- None

Consumables

- As required per Sabreliner Corrosion Control Manual (SR-76-23), chapter 20

Inspection

1. Remove main landing gear wheel well tubs. (See Figure 53-00-613.)
2. Clean floor support beam and wing attach fittings, per Sabreliner Corrosion Control Manual (SR-76-23), chapter 20.
3. Visually inspect beam, and wing attach fittings for cracks and corrosion. (See Figure 53-00-613.)

NOTE

Any repairs accomplished should be made in accordance with Sabreliner Structural Repair Manual (NA-66-1032) and/or Sabreliner Corrosion Control Manual (SR-76-23).

4. Install MLG wheel well tubs removed in step 1.

INSPECT — INTERMEDIATE TO AFT UPPER LONGERON SPLICE F.S. 357 LH AND RH

Equipment/Tools

- None

Consumables

- Sealant PR 1422 GB

Inspection

1. Open door and aft equipment bay doors. (See Figure 53-00-614.)

NOTE

Do not remove sealant unless sealant is cracked, split, bubbled, or corrosion damage is evident in the area.

2. Visually inspect splices and surrounding skin for loose bolts, cracks, and corrosion. (See Figure 53-00-614.)

NOTE

Any repairs accomplished should be made in accordance with Sabreliner Structural Repair Manual (NA-66-1032) and/or Sabreliner Corrosion Control Manual (SR-76-23).

3. Apply sealant (PR 1422 GB) per Sabreliner Corrosion Control Manual (SR-76-23), chapter 20.
4. Close doors opened in step 1.

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NOTES

- *Clean and inspect for corrosion on the lower surface of the structure.*
- *If evidence of corrosion is found, remove fiberglass, clean and inspect upper surface of longeron and stop.*

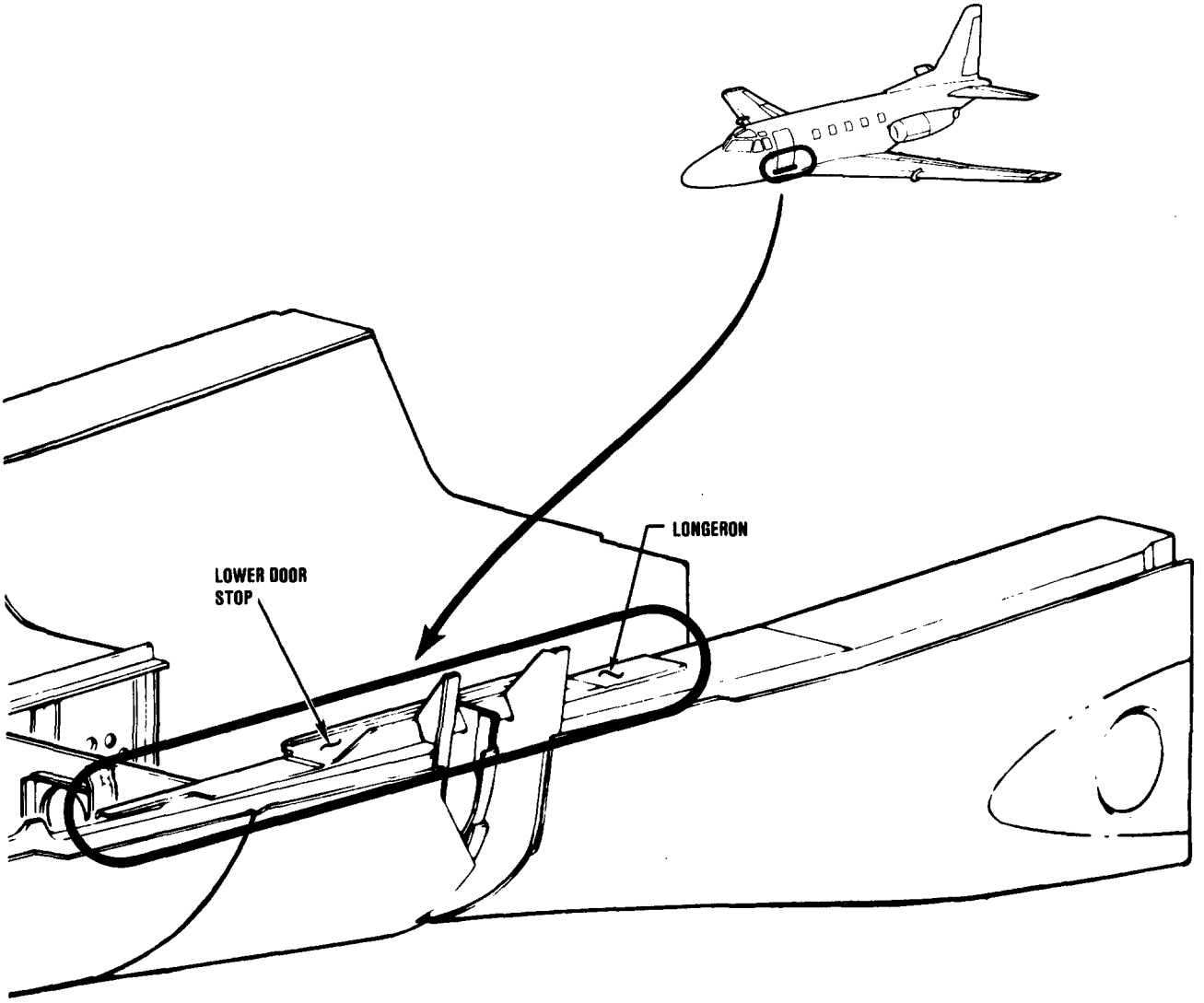

Figure 53-00-612 — Inspect — Lower Longeron Along Threshold of Main Entrance Door and Lower Door Stop

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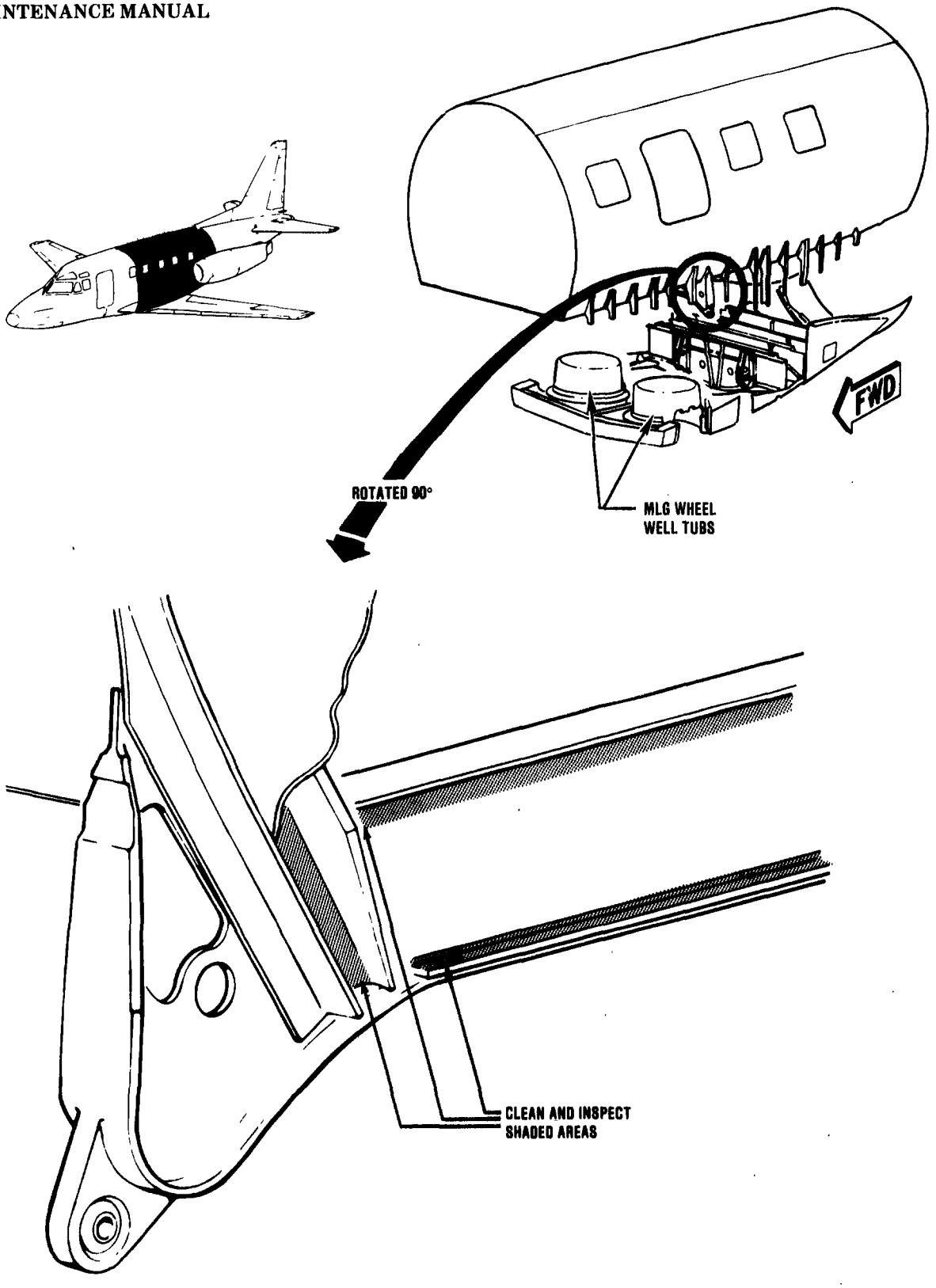

Figure 53-00-613 — Inspect — Floor Support Beam F.S. 264 and Wing Attach Fittings LH and RH on Fuselage

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NOTES

- *Remove upper fuselage door and inspect fwd portion of splice.*
- *Inspect aft portion of splice from aft equipment bay.*
- *Do not remove sealant unless sealant is cracked, bubbled, or corrosion damage is evident in this area.*

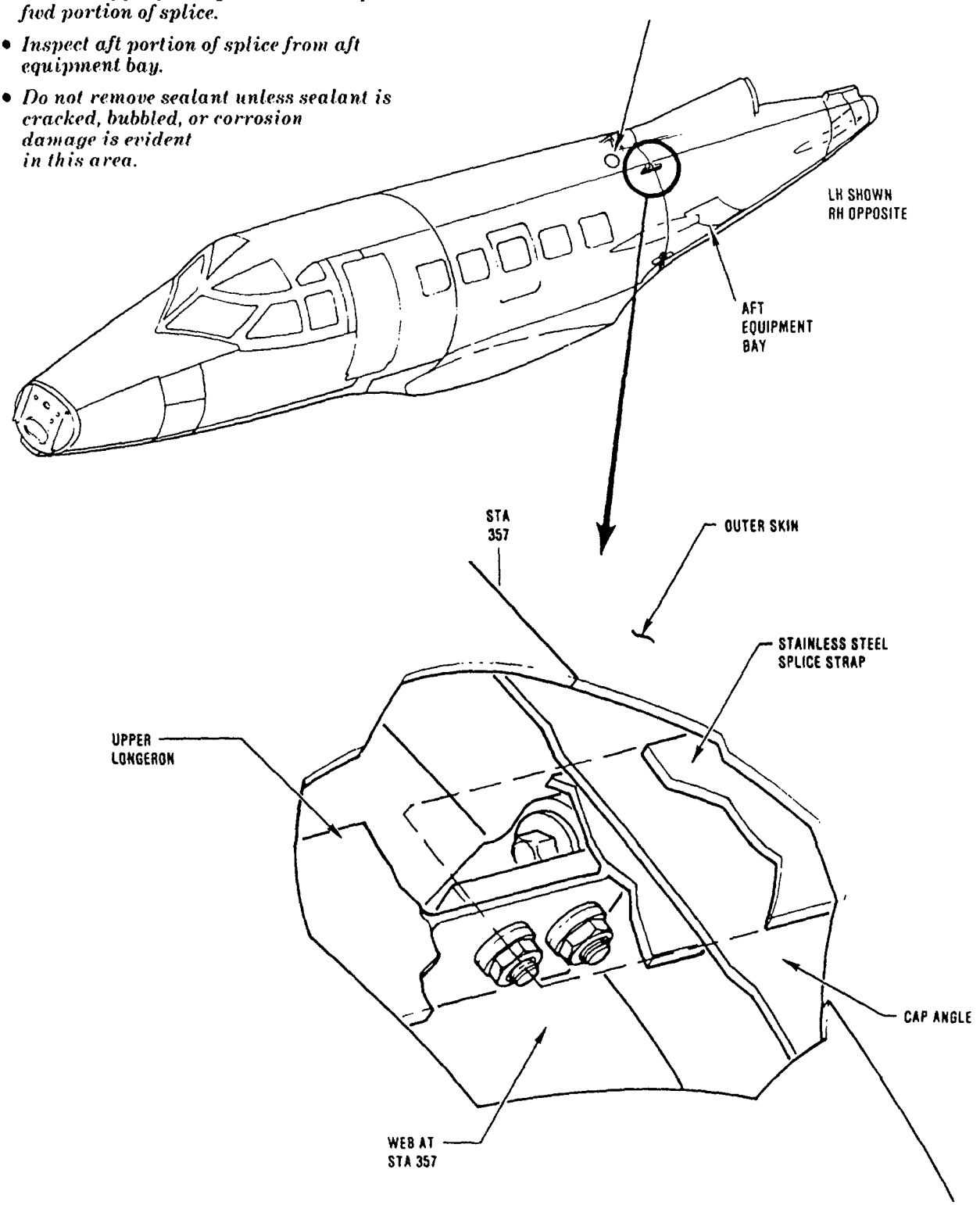

Figure 53-00-614 — Inspect — Intermediate to Aft Upper Longeron Splice F.S. 357 LH and RH

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INSPECTION/CHECK (Continued)

INSPECT — VERTICAL HAT SECTIONS ON AFT SIDE OF F.S. 333, FLOOR OF AUX FUEL BAY, PYLON CASTINGS AND UPPER INTERMEDIATE LONGERONS LH AND RH F.S. 333 TO 357 (ESPECIALLY NEAR THE CENTER OF THE BULKHEAD)

Equipment/Tools

- None

Consumables

- Sealant PR 1422 GB

Inspection

1. Remove aft aux fuel cell per applicable Sabreliner Maintenance Manual, chapter 28.
2. Remove sealant per Figure 53-00-615.
3. Inspect entire surface of vertical hat sections, top aux skin seams, and two inches of floor aft of hat sections for cracks and corrosion.
4. Inspect the rest of the aux fuel cell floor for cracked, split, bubbled sealant or evident corrosion damage.
5. Inspect upper intermediate longerons for cracked, split, bubbled sealant, cracks and corrosion.
6. Inspect the Pylon Casting Fingers at F.S. 333 for cracks and corrosion.

   NOTE

   Any repairs may require F.S. 333 interior removal.

   NOTE

   Any repairs accomplished should be made in accordance with Sabreliner Structural Repair Manual (NA-66-1032) and/or Sabreliner Corrosion Control Manual (SR-76-23).

7. Apply Sealant (PR 1422 GB) per Sabreliner Corrosion Control Manual (SR-76-23), chapter 20.
8. Perform auxiliary fuel cell bay pressurization and leakage test, per Sabreliner Maintenance Manual, chapter 21.
9. Install aft aux fuel cell per applicable Sabreliner Maintenance Manual, chapter 28.

INSPECT — AFT AUX FUEL CELL FLOOR SUPPORT STRUCTURE

Equipment/Tools

- Adjustable Flex Shaft Borescope and Power Supply

Consumables

- None

Inspection

1. Remove access panel. (See Figure 53-00-616.)
2. With a borescope inspect floor support structure for cracks and corrosion per Figure 53-00-616.

   NOTE

   Any repairs may require Aux Fuel Cell removal.

   NOTE

   Any repairs accomplished should be made in accordance with Sabreliner Structural Repair Manual (NA-66-1032) and/or Sabreliner Corrosion Control Manual (SR-76-23).

3. Install access panel removed in step 1.

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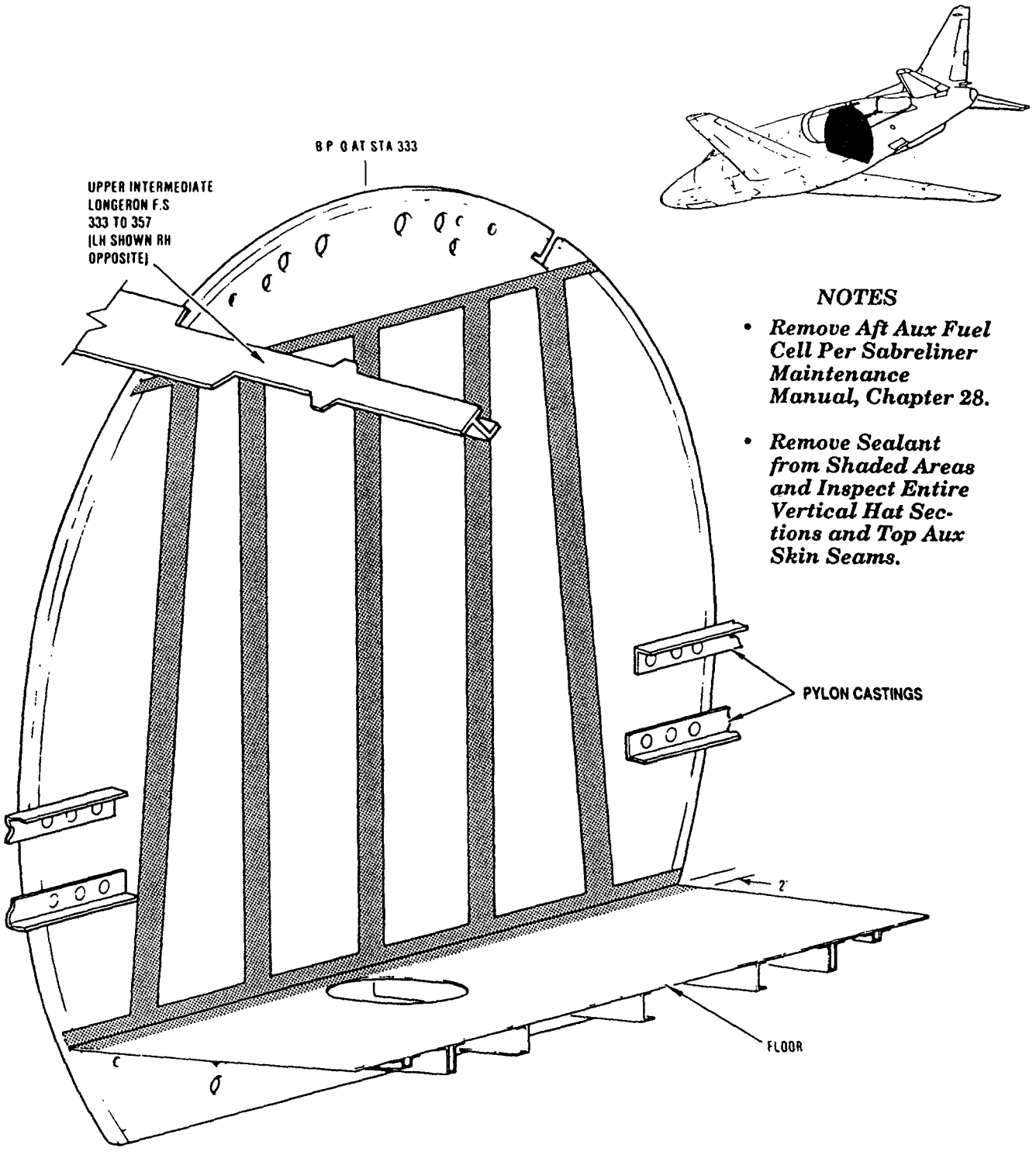
BP 0 AT STA 333
UPPER INTERMEDIATE LONGERON F.S 333 TO 357 (LH SHOWN RH OPPOSITE)
NOTES
• Remove Aft Aux Fuel Cell Per Sabreliner Maintenance Manual, Chapter 28.
• Remove Sealant from Shaded Areas and Inspect Entire Vertical Hat Sections and Top Aux Skin Seams.
PYLON CASTINGS
2"
FLOOR

Figure 53-00-615 — Inspect — Vertical Hat Sections on Aft Side of F.S. 333, Floor of Aux Fuel Bay, Pylon Castings and Upper Intermediate Longerons LH and RH F.S. 333 to 357 (Especially Near the Center of the Bulkhead)

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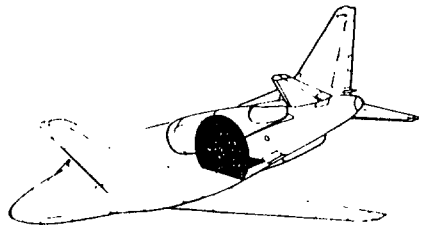

NOTES

- *Remove access panel below floor structure and inspect.*
- *Use boroscope for detailed inspection of shaded areas.*

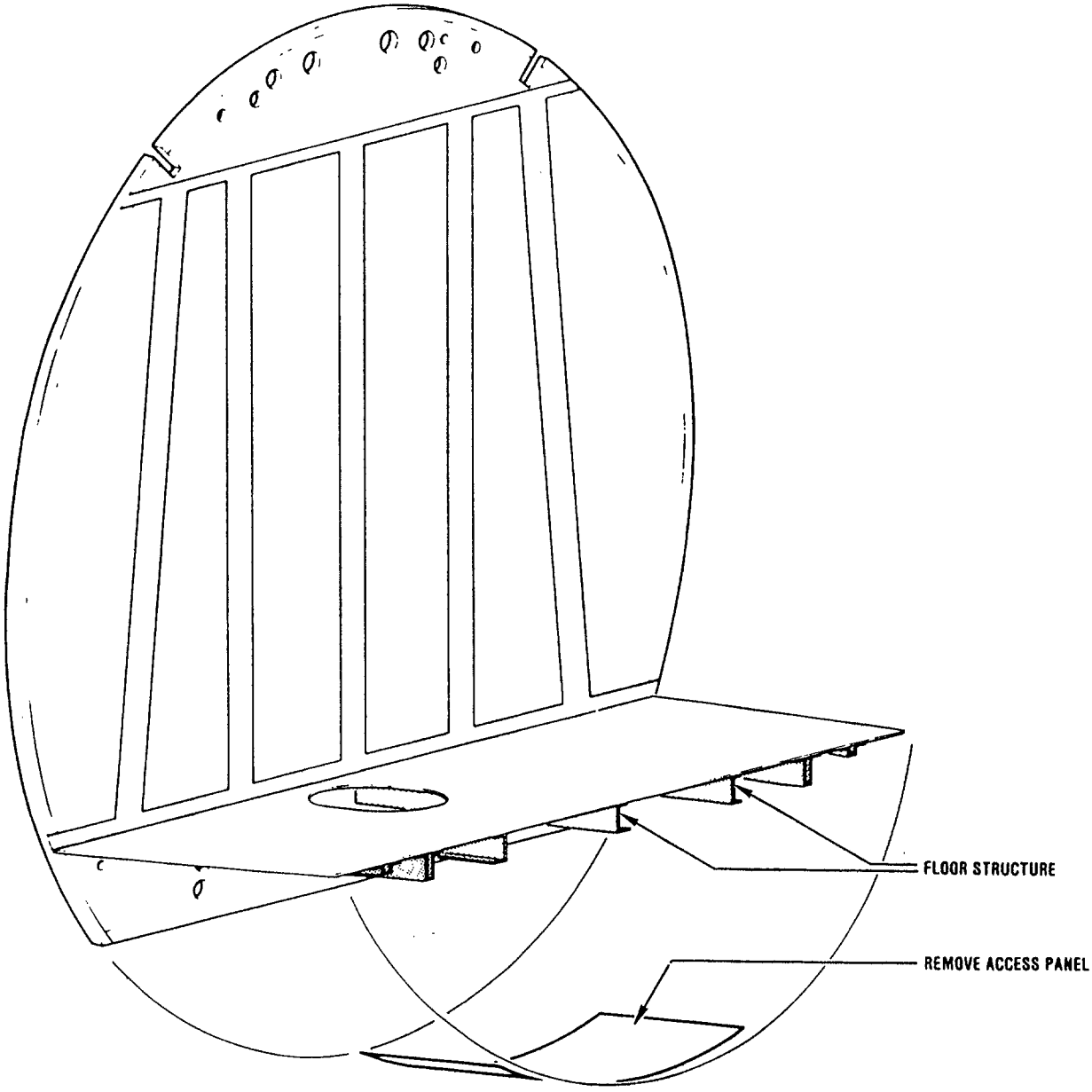

Figure 53-00-616 — Inspect — Aft Aux Fuel Cell Floor Support Structure

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INSPECTION CHECK (Continued)

INSPECT — INACTIVE ESCAPE HATCH

Equipment/Tools

- None

Consumables

- None

Inspection

1. To inspect inactive escape hatch, proceed as follows:
   A. Remove flooring as necessary to gain access to inactive escape hatch.
   B. Check for foreign material in escape hatch drain holes.
   C. Inspect area for corrosion and moisture. Remove and/or treat as necessary.
   D. Replace flooring over escape hatch.

INSPECT — INACTIVE ESCAPE HATCH HINGE FITTINGS, AND MOUNTING PIN RECEPTACLES — S/N 370-1 AND SUBS AND 380-1 THRU -68

WARNING

Clear wheel well area of equipment and personnel. Do not enter speed brake area when electrical and hydraulic power is applied. Anyone struck by a moving speed brake panel can be seriously or fatally injured.

1. Lower speed brake.
2. Install ground safety locks.
3. Remove access panels. (See Figure 53-00-618.)

Remove inactive escape hatch as follows:

   A. Remove floor covering over escape hatch access in companion way area.
   B. Remove all sound proofing and insulation from hatch area.
   C. Drill out rivets around periphery of inflight escape hatch securing pressure seal to hatch.
   D. Remove safetywire securing hatch mounting pin in place.
   E. Remove mounting pins from receptacles in FS 174.
   F. Allow forward end of hatch to drop down to clear fuselage structure, than, move hatch forward to remove from airplane.

5. With hatch removed, carefully examine the inflight escape hatch opening in the fuselage and the hatch itself for indications of cracks and corrosion. (See Figure 53-00-619.)
6. If cracks or corrosion are found, repair per instructions in Corrosion Control Manual (SR-76-023).
7. If no cracks or corrosion are found, install escape hatch as follows:
   A. Lubricate mounting pins with Lubri-bond A.
   B. Slide escape hatch aft onto hinge fittings. Then, push up and install mounting pins.
   C. Seal around pressure seal and escape hatch using PR1422 GB sealant.
   D. Install rubstrips and retainers. Reference Corrosion Control Manual, SR-76-023.
   E. Secure door over escape hatch in companionway floor.
8. Remove ground safety locks.
9. Close speed brake.

INSPECT — LOWER CAP ANGLE IN AFT WELL — F.S. 298 + 21 — S/N 370-1 AND SUBS AND 380-1 AND SUBS

1. Remove floor covering around the forward portion of the aft well at F.S. 298 + 21.
2. Expose vertical flange of P/N 306-312300-007 and remove sealant from radius. See Figure 53-00-620.
3. Inspect vertical flange for cracks and corrosion.
4. Remove inspection panel and inspect flange P/N 306-312300-007 from below for cracks and corrosion. See Figure 53-00-620.
5. If evidence of cracks and corrosion is present, remove sealant, finish, and structure as required to determine full extent of damage.

NOTE

Any repairs accomplished should be made in accordance with Sabreliner Structural Repair Manual (NA-66-1032) and/or Sabreliner Corrosion Control Manual (SR-76-23).

6. Seal in accordance with Sabreliner Corrosion Control Manual, SR-76-23.
7. Install items removed in Steps 1 and 4.

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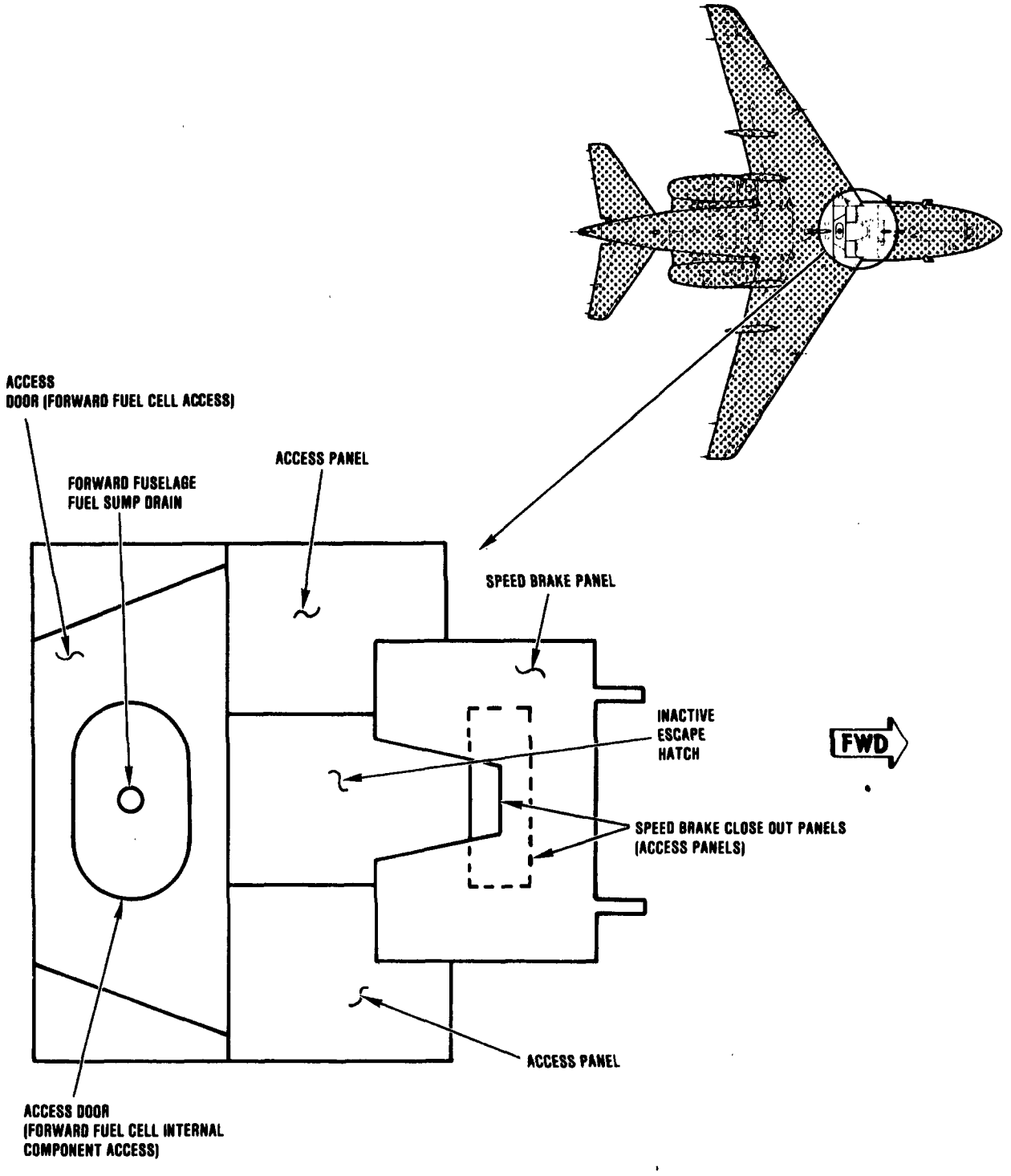

Figure 53-00-617 — Inactive Escape Hatch Access — S/N 380-70 Only

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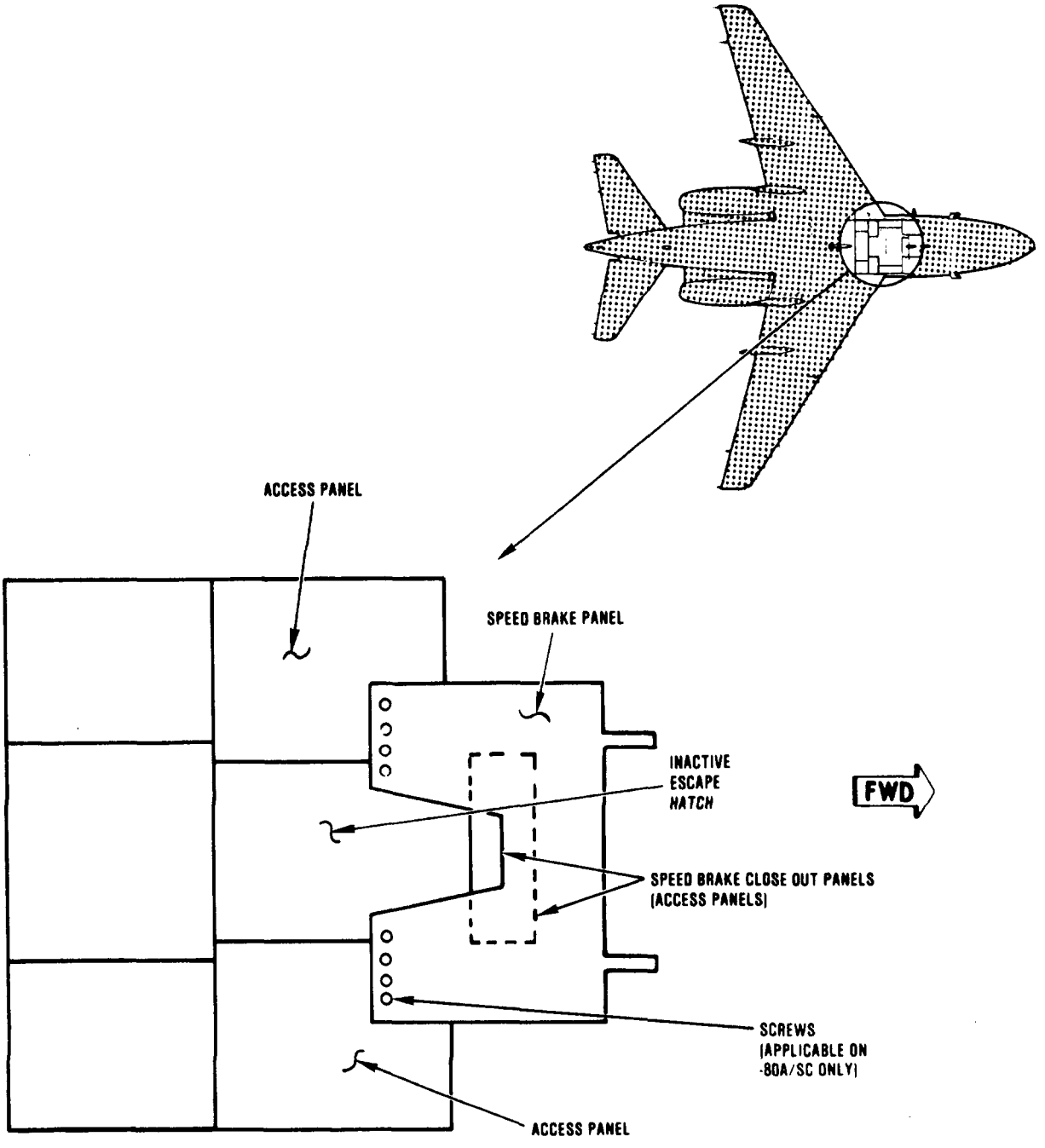

Figure 53-00-618 — Inactive Escape Hatch Access — S/N 370-1 thru -9 and 380-1 thru -69, -71 and Subs

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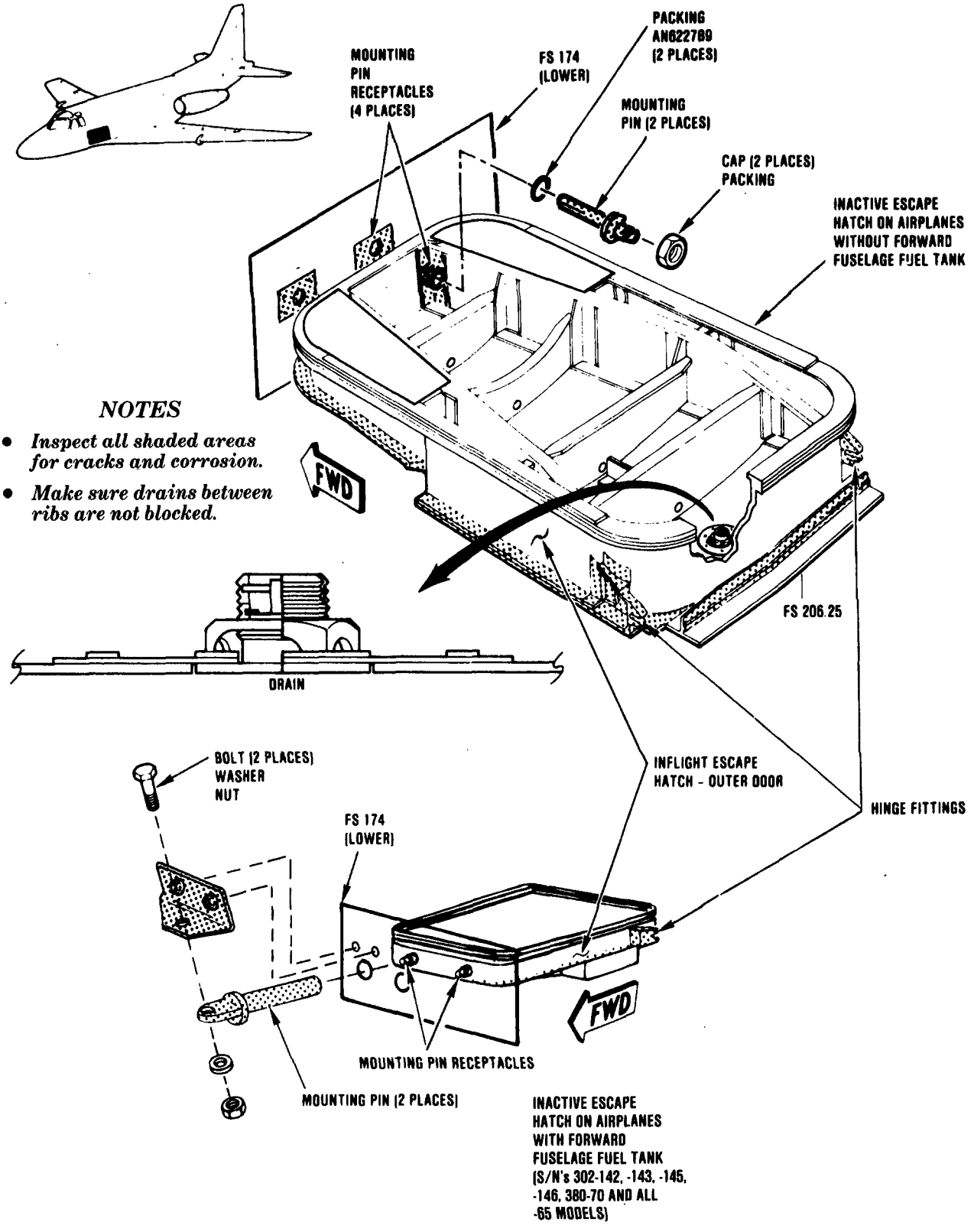
MOUNTING PIN RECEPTACLES (4 PLACES)
FS 174 (LOWER)
PACKING AN622789 (2 PLACES)
MOUNTING PIN (2 PLACES)
CAP (2 PLACES) PACKING
INACTIVE ESCAPE HATCH ON AIRPLANES WITHOUT FORWARD FUSELAGE FUEL TANK
NOTES
• Inspect all shaded areas for cracks and corrosion.
• Make sure drains between ribs are not blocked.
FWD
FS 206.25
DRAIN
BOLT (2 PLACES)
WASHER
NUT
FS 174 (LOWER)
INFLIGHT ESCAPE HATCH - OUTER DOOR
HINGE FITTINGS
FWD
MOUNTING PIN RECEPTACLES
MOUNTING PIN (2 PLACES)
INACTIVE ESCAPE HATCH ON AIRPLANES WITH FORWARD FUSELAGE FUEL TANK (S/N's 302-142, -143, -145, -146, 380-70 AND ALL -65 MODELS)

Figure 53-00-619 — Inactive Escape Hatch Inspection

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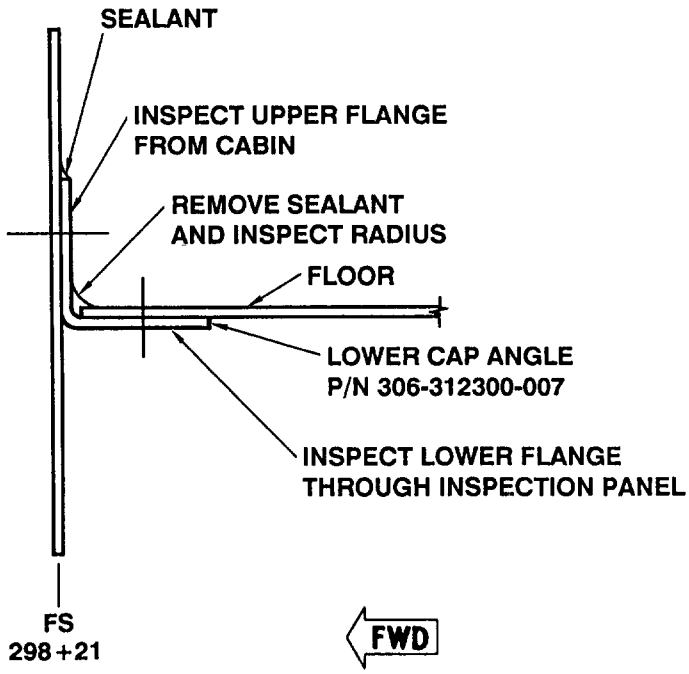

SECTION AT A/C CENTERLINE
VIEW LKG OUTBD

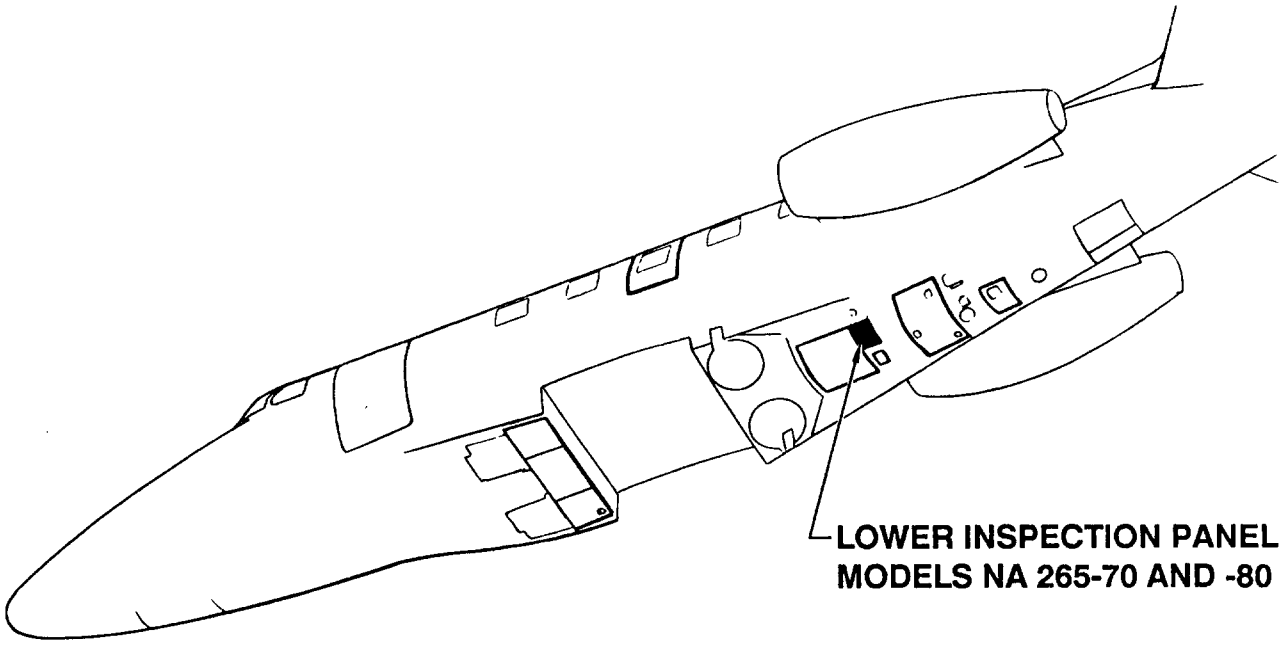

Figure 53-00-620 — Inspect Lower Cap Angle In Aft Well
F.S. 298 + 21 — S/N 370-1 and Subs and 380-1 and Subs

NEW

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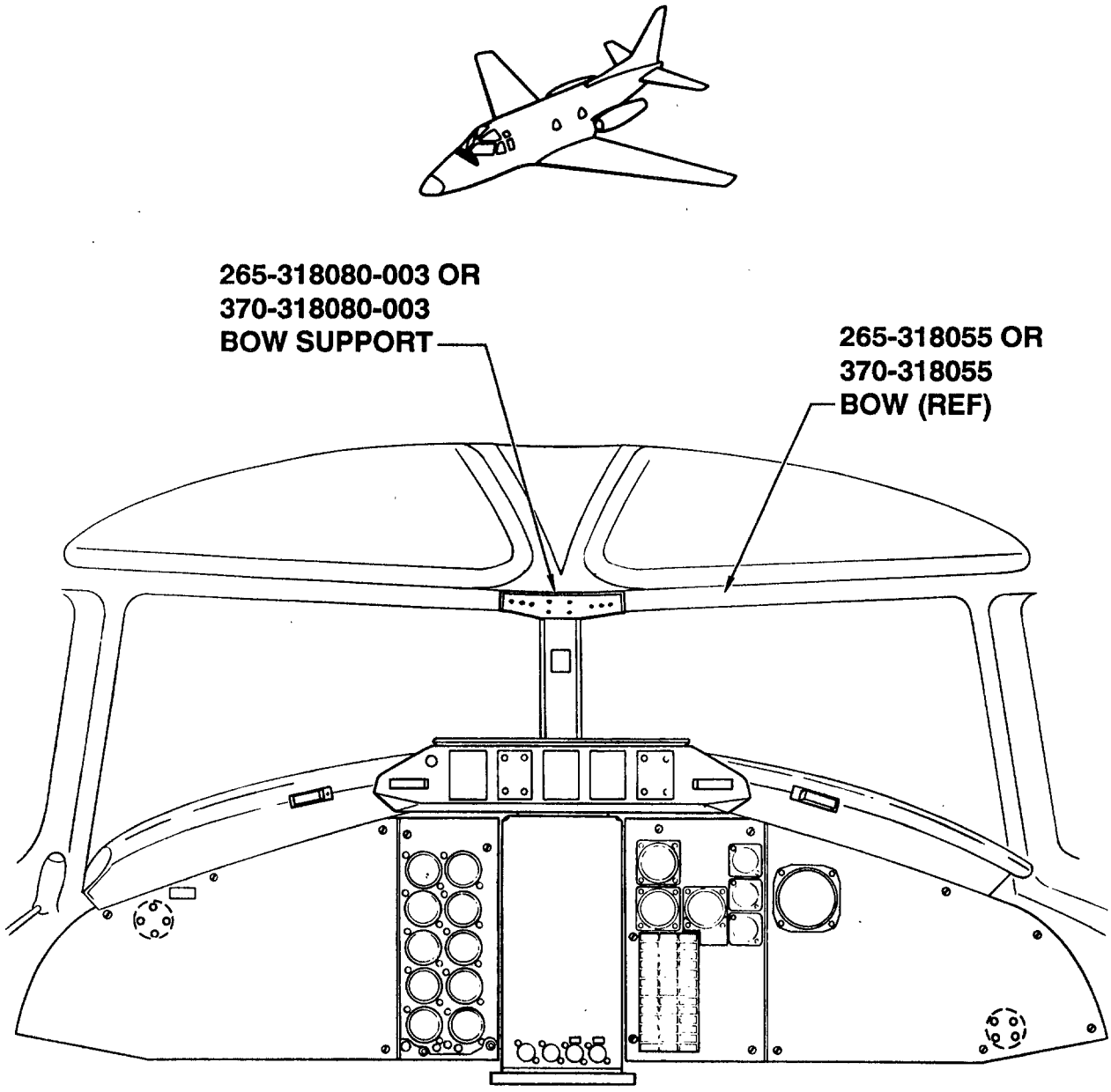

Figure 53-00-621 — Inspect Windshield Bow Support (Sheet 1 of 3)

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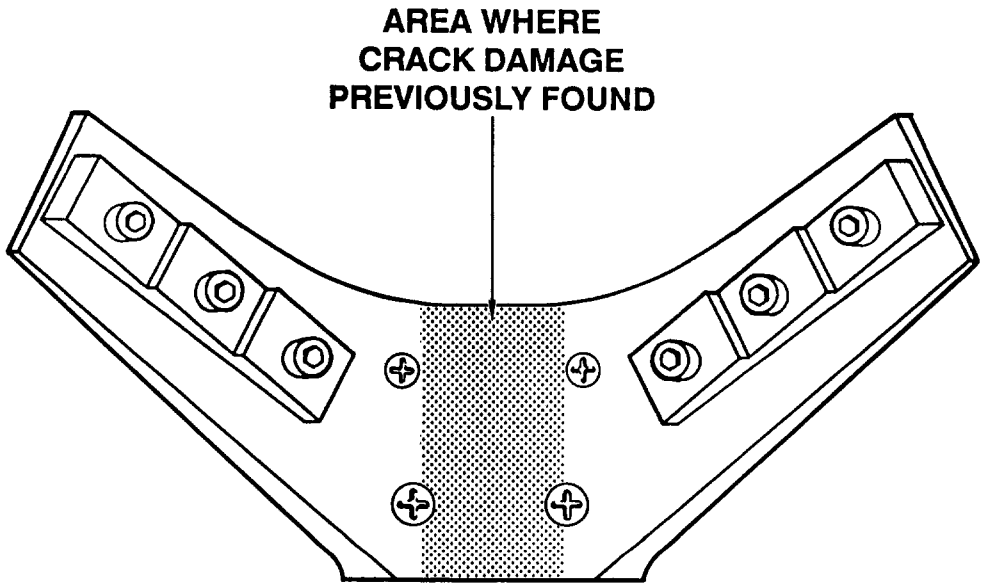

Figure 53-00-621 — Inspect Windshield Bow Support (Sheet 2 of 3)

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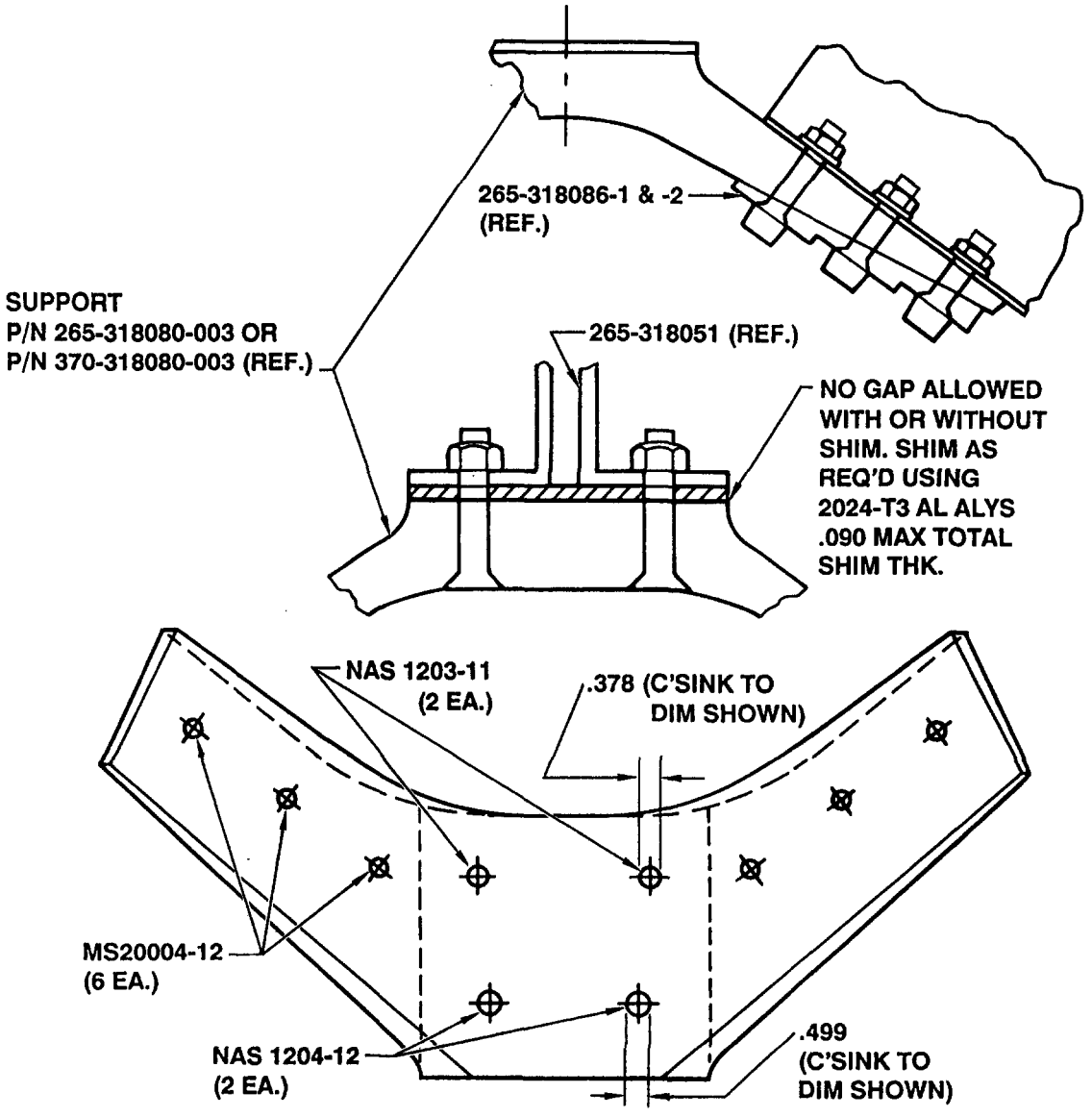
1. VERIFY AN UNBROKEN LAYER OF EPOXY POLYAMIDE PRIMER IS PRESENT ON MATING SURFACES.
2. INSTALL SUPPORT WITHIN TOLERANCES SHOWN BELOW:



NOTE

.190-.194 dia. for #10 screws and .250-.254 for 1/4" screws.

Figure 53-00-621 — Inspect Windshield Bow Support (Sheet 3 of 3)

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INSPECTION CHECK (Continued)

INSPECT — UPPER MAIN ENTRANCE DOOR STOP FLANGE P/N 370-311286-5 — S/B 92-4 S/N 370-1 AND SUBS AND 380-1 AND SUBS

Equipment/Tools

- Nortec 23ST Mini Eddy Scope (or equivalent) Reference Crack Standard P/N 370-311286-005 (Figure 53-00-623)

Consumables

- Sealant PR1422GB2 (or equivalent)
- 13"x6" sheet Stainless Steel 0.020 thick
- Methyl Ethyl Ketone (MEK) (or equivalent)

Inspection

1. Remove upper interior trim panel on F.S. 143 to F.S. 172 entryway bulkhead.
2. Remove upper center interior trim panel above entrance door opening.
3. Remove sealant along top edge of door seal retainer (Figure 53-00-622).
4. Insert a 0.020 stainless steel sheet between structure and door seal retainer.
5. Mark cut line in accordance with dimensions shown on Figure 53-00-622.

CAUTION

Make sure stainless steel sheet is inserted below cut line to protect structure during cutting of fiberglass door seal retainer. (No more than .10 inch below cut line.)

6. Cut fiberglass door seal retainer. Retain removed fairing portion for reinstallation.
7. Clean 370-311285-17 support with MEK to remove all sealant in preparation for the Eddy Current test.
8. Inspect upper door stop flange P/N 370-311286-5 per steps A thru I.
   A. Make sure area to be inspected is free of sealant and other surface related materials that could affect the sensitivity of the test.
   B. Set the Eddy Scope as follows:

      Frequency — 1 khz
      Vertical Gain — 92
      Horizontal Gain — 78
      Vertical Position — 65
      Horizontal Position — 14
      Rotation — 306

   C. Null the probe.
   D. Calibrate the Eddy Scope utilizing the setup standard P/N 370-311286-005 (Figure 53-00-623).
   E. Make sure the frequency setting is adequate to penetrate the .050 thick doubler on the setup standard (typical subsurface crack inspection).

NOTE

The setup parameters may vary depending on the equipment used, and may differ from those described in step B.

   F. The crack response from the setup standard should produce a vertical deflection as shown on Figure 53-00-624 (View A). Scan at 90° to the potential crack direction. Once the crack response can be produced from the reference standard, proceed to step G.
   G. In reference to the area to be inspected, the suspected crack runs in a forward and aft direction as shown on Figure 53-00-624 (View B). Scan at 90° to the potential crack direction with multiple passes.
   H. To ensure test sensitivity is being obtained, periodically recalibrate to the reference standard during testing.
   I. Mark all relevant indications with a commercial type grease pencil.
9. If cracks are detected — contact Sabreliner Corporation Engineering — Spirit Facility (No cracks are acceptable). If no cracks are detected, proceed with step 10.
10. Bond fiberglass fairing strips back onto P/N 370-311285-17 support using PR1422GB2. Fill cutting cap with sealant and seal edges of fairing strips. Let cure and touch up with matching interior paint as required.
11. Reinstall interior trim panels (3 each).

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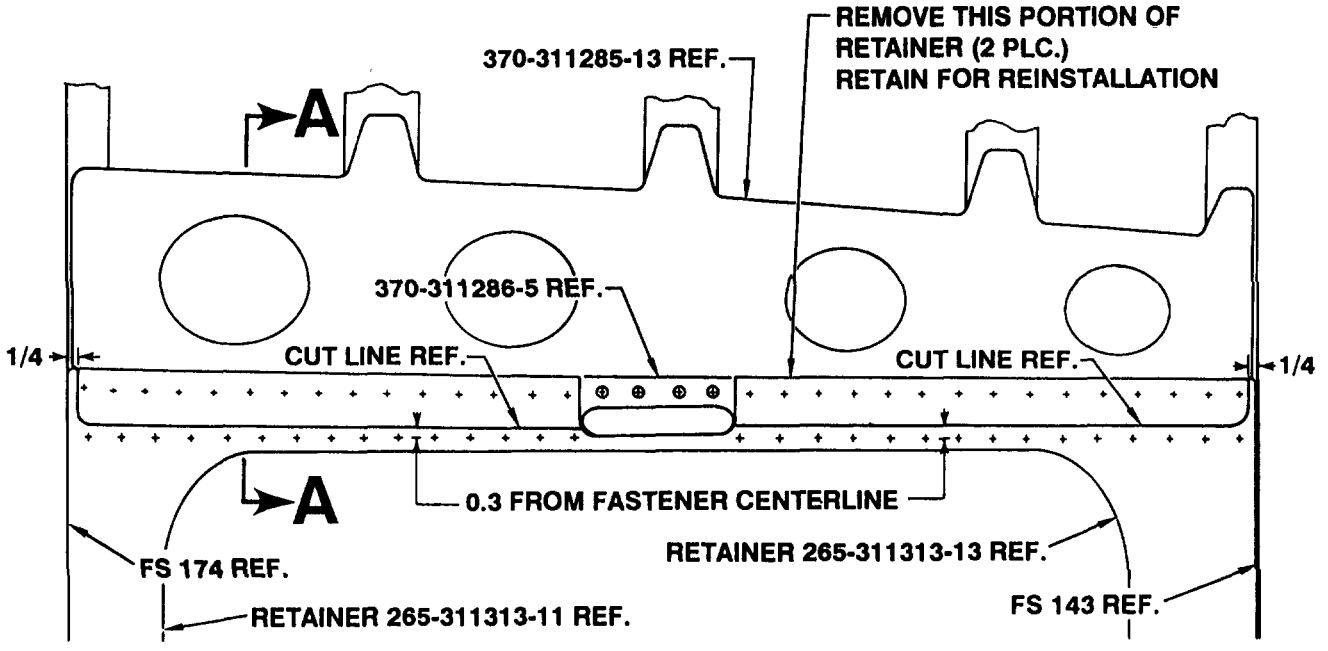
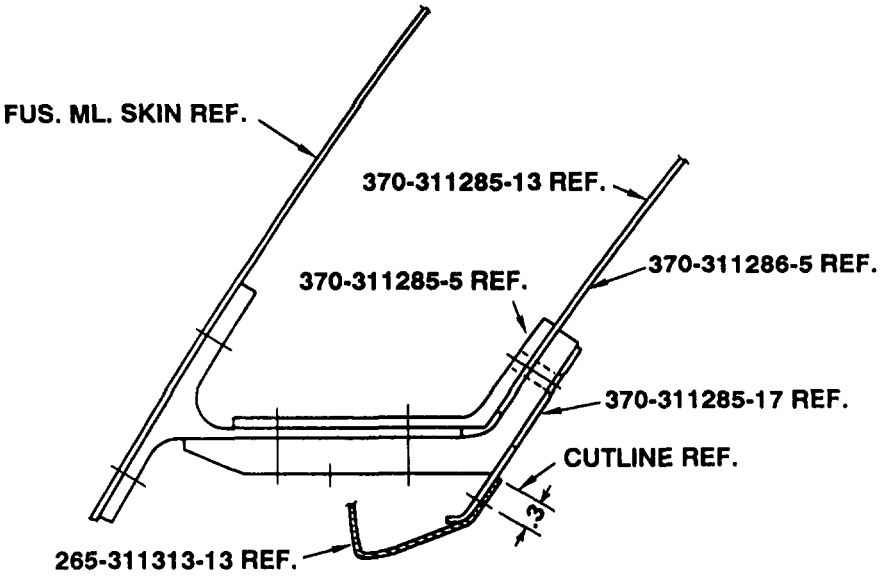


VIEW LOOKING OUTBOARD LH SIDE



A-A

Figure 53-00-622 — Cut Line Dimensions and Inspection Area (Sheet 1 of 2)

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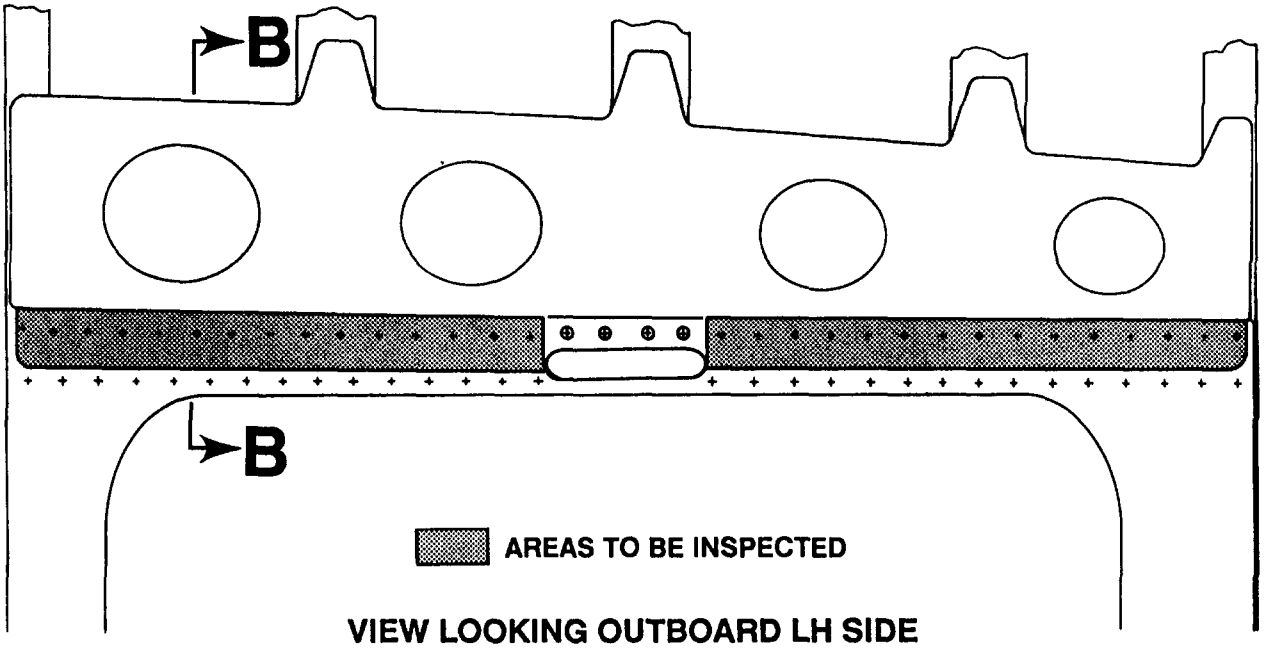

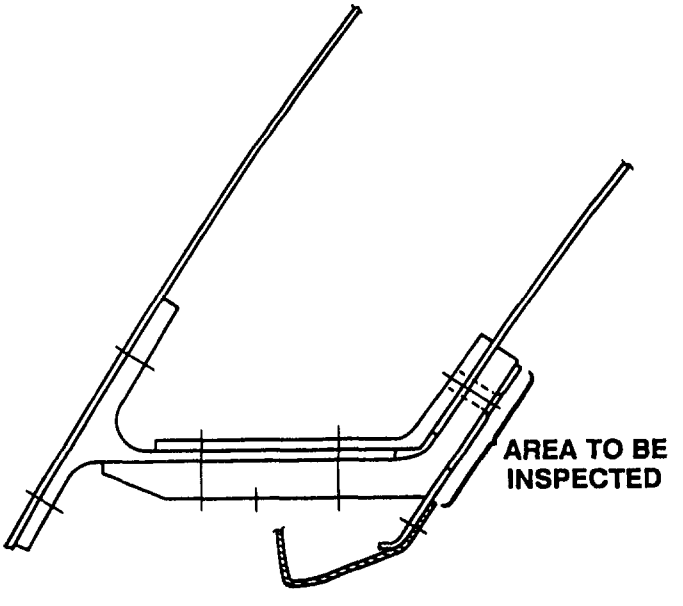

B-B

Figure 53-00-622 — Cut Line Dimensions and Inspection Area (Sheet 2 of 2)

NEW

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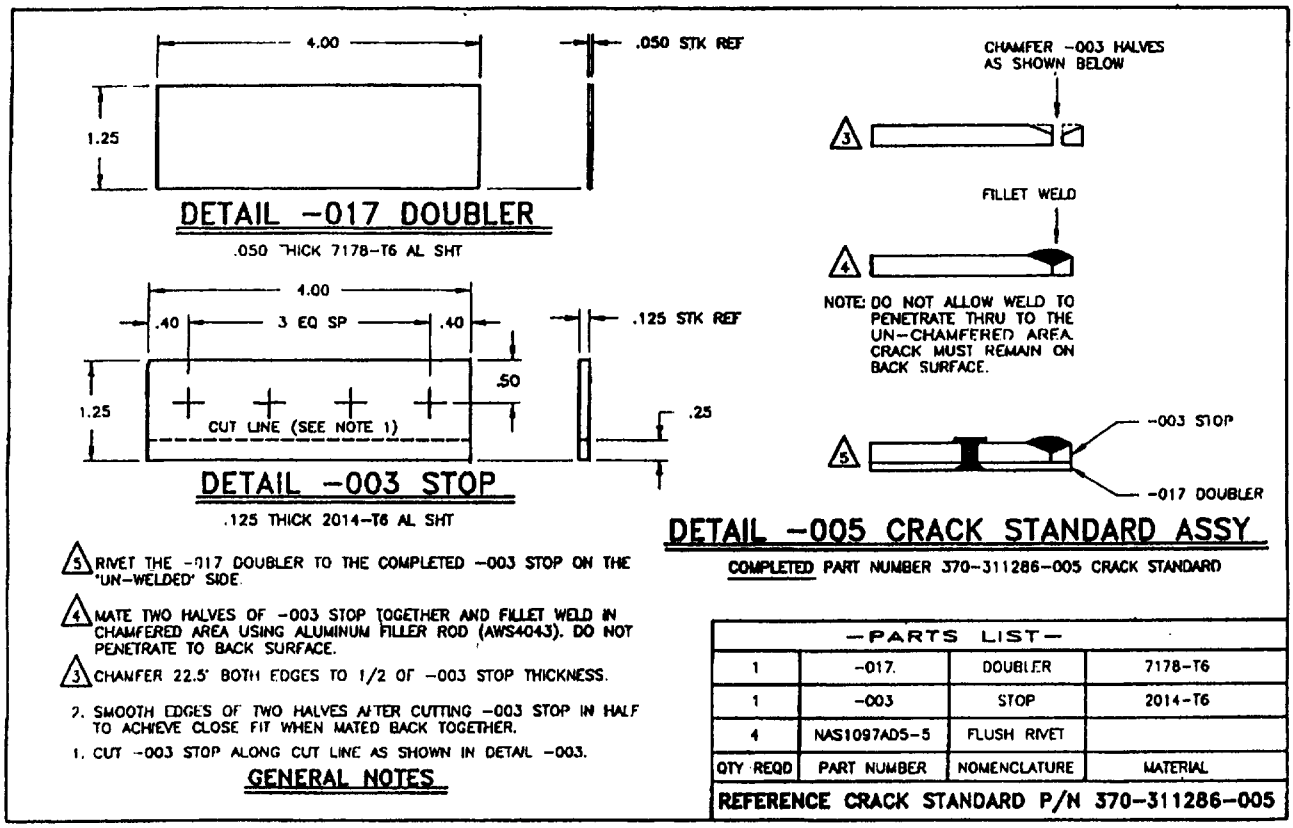
4.00
.050 STK REF
1.25
DETAIL -017 DOUBLER
.050 THICK 7178-T6 AL SHT
4.00
.40
3 EQ SP
.40
.125 STK REF
.50
1.25
.25
CUT LINE (SEE NOTE 1)
DETAIL -003 STOP
.125 THICK 2014-T6 AL SHT
CHAMFER -003 HALVES AS SHOWN BELOW
3
FILLET WELD
4
NOTE: DO NOT ALLOW WELD TO PENETRATE THRU TO THE UN-CHAMFERED AREA. CRACK MUST REMAIN ON BACK SURFACE.
5
-003 STOP
-017 DOUBLER
DETAIL -005 CRACK STANDARD ASSY
COMPLETED PART NUMBER 370-311286-005 CRACK STANDARD
5 RIVET THE -017 DOUBLER TO THE COMPLETED -003 STOP ON THE 'UN-WELDED' SIDE.
4 MATE TWO HALVES OF -003 STOP TOGETHER AND FILLET WELD IN CHAMFERED AREA USING ALUMINUM FILLER ROD (AWS4043). DO NOT PENETRATE TO BACK SURFACE.
3 CHAMFER 22.5° BOTH EDGES TO 1/2 OF -003 STOP THICKNESS.
2. SMOOTH EDGES OF TWO HALVES AFTER CUTTING -003 STOP IN HALF TO ACHIEVE CLOSE FIT WHEN MATED BACK TOGETHER.
1. CUT -003 STOP ALONG CUT LINE AS SHOWN IN DETAIL -003.
GENERAL NOTES
-PARTS LIST-
1 -017. DOUBLER 7178-T6
1 -003 STOP 2014-T6
4 NAS1097AD5-5 FLUSH RIVET
QTY REQD PART NUMBER NOMENCLATURE MATERIAL
REFERENCE CRACK STANDARD P/N 370-311286-005

Figure 53-00-623 — Reference Crack Standard P/N 370-311286-5

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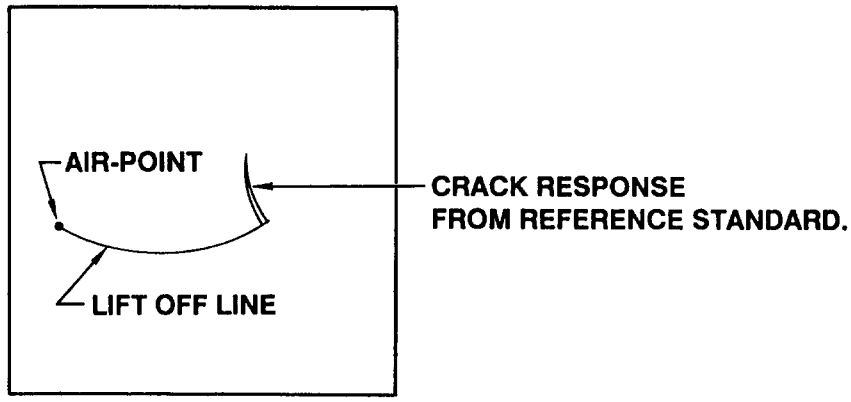
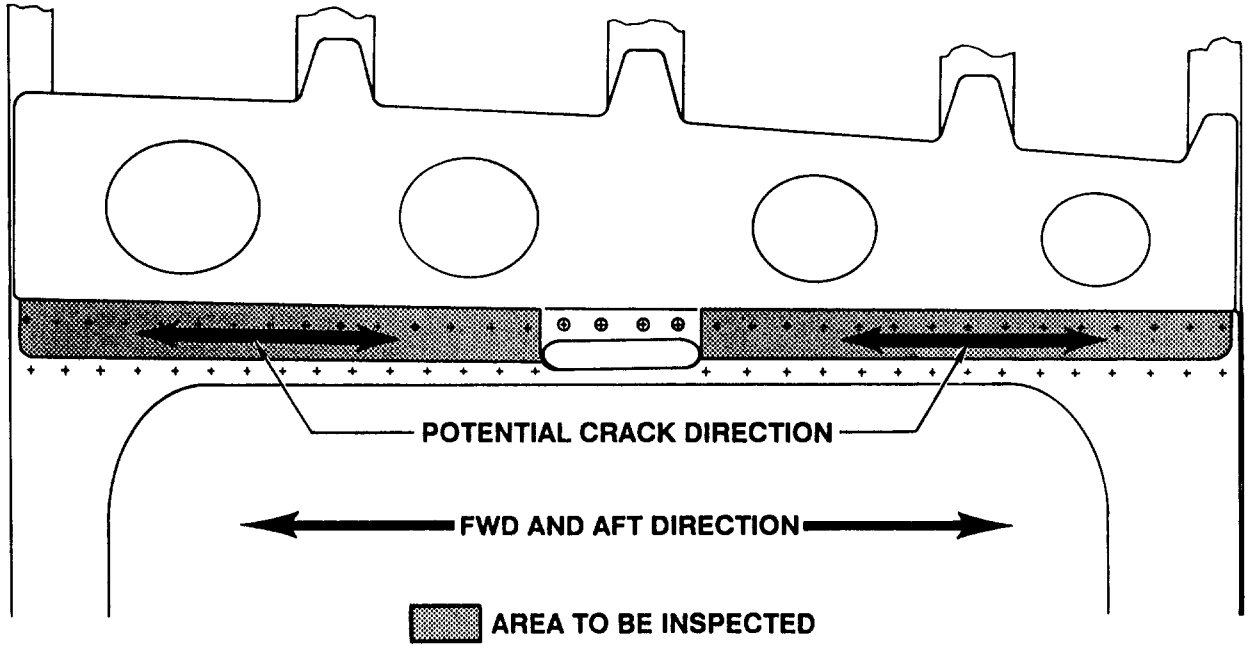


CRT (VIEW A)



(VIEW B)

Figure 53-00-624 — CRT Crack Response and Potential Crack Direction

NEW

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INSPECTION CHECK (Continued)

INSPECT — CLIPS — FS 76

1. Remove instruments at upper center section of instrument panel to gain access to FS 76.
2. Using borescope with right angle viewing, inspect radius of clips attached from angle braces to top of FS 76 bulkhead for cracks. If any cracks are found, contact Engineering Department — Sabreliner Corporation, for disposition.
3. Reinstall instruments removed in step 1.

INSPECT — PYLON MATCH ANGLES (RH and LH) AND FUSELAGE SKIN — S/N 370-2 thru -6, -8 and -9

Inspection

1. Inspect match angles between pylon and fuselage skin (see Figure 53-00-622) for evident corrosion damage migrating from angles, upper and lower surfaces of pylon.
2. Check for cracked or split sealer, bubbles in sealer, or any evidence of corrosion beneath sealer.
3. An inspection of the stainless steel heat shield doublers, P/N 265-321250-5 LH and -6 RH (see Figure 53-00-622) must be accomplished from the aircraft aft equipment bay. Check for cracked or bubbled paint, or any visible signs of corrosion.

NOTE

An inspection of the forward stainless steel heat shield doublers, P/N 265-321350-3 LH and -4 RH, can be accomplished only when the Aux. Fuel Cell has been removed.

4. If corrosion is evident through paint, the paint must be stripped from an area around the upper and lower match angles sufficient to determine the extent of the corrosion. If evidence of corrosion between mating surfaces if found, the suspect pylon must be removed from the fuselage, so that further damage inspections can be performed. (Reference Structural Repair Manual, NA-66-1032 and Corrosion Control Manual, SR-76-23.)

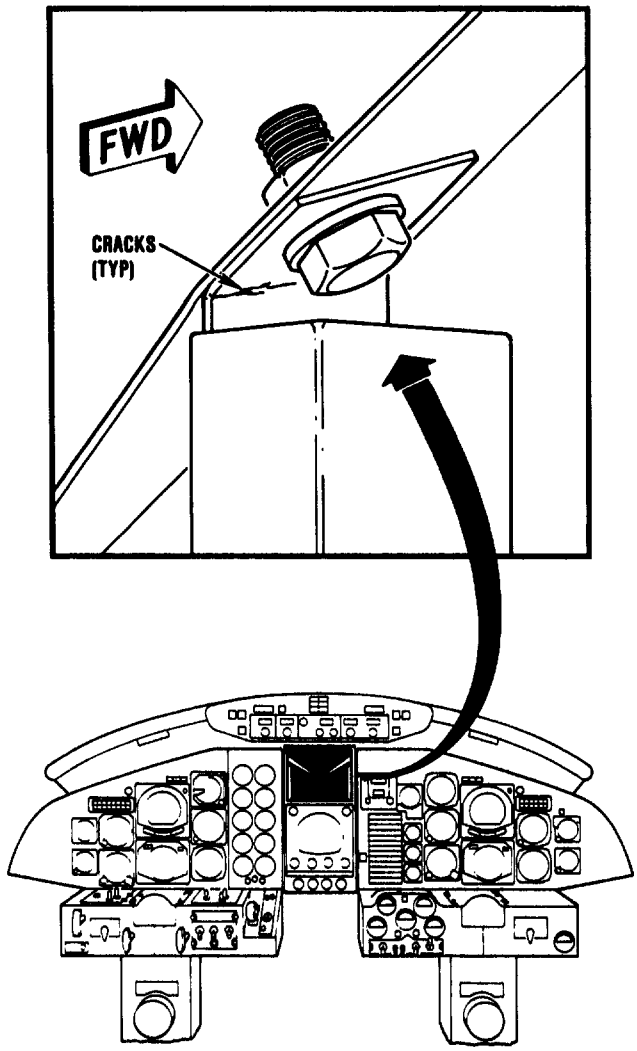

Figure 53-00-625 — Inspect — Clips, F.S. 76

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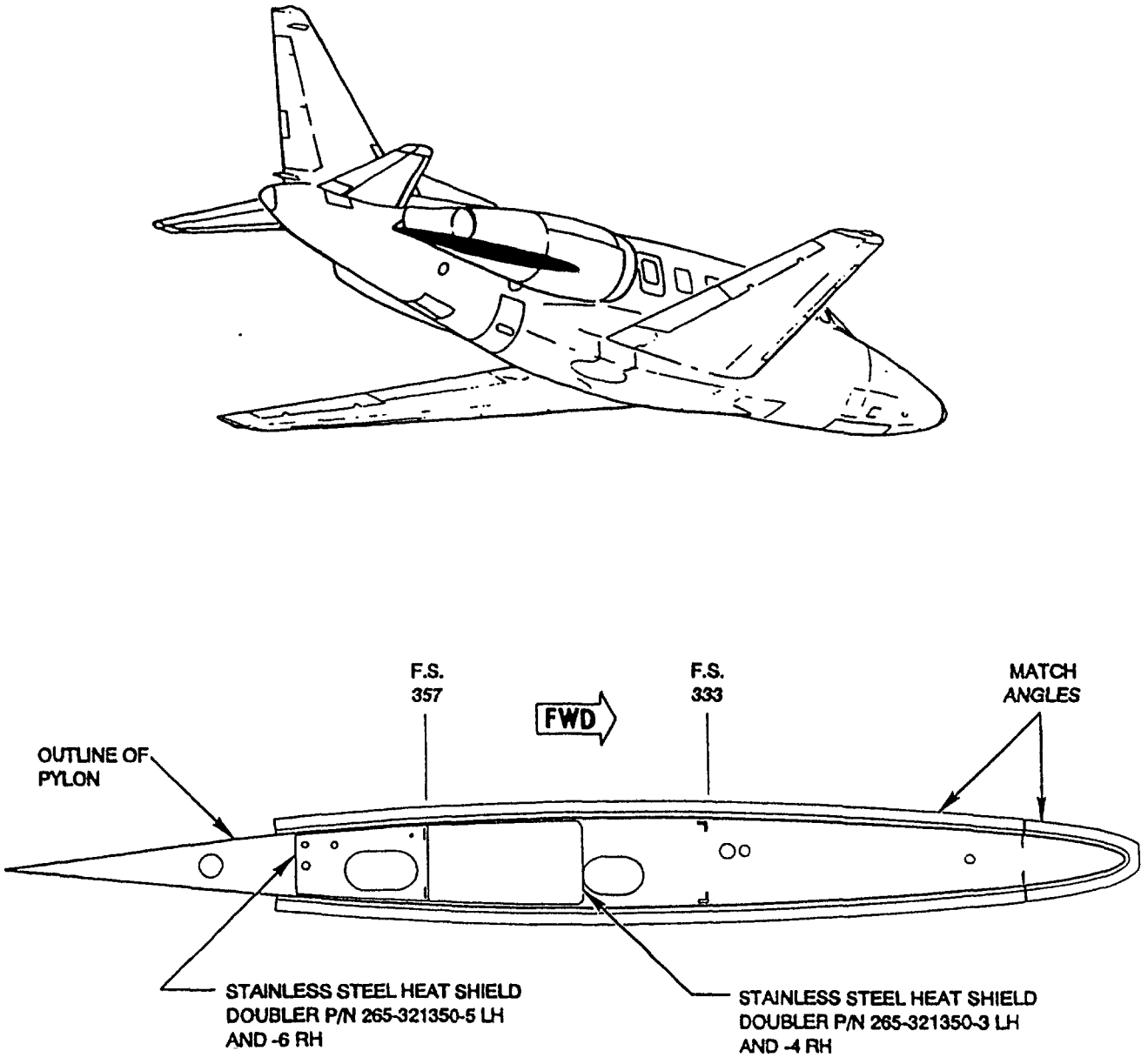

Figure 53-00-626 — Inspect Pylon Match Angles and Fuselage Skin

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INSPECTION/CHECK (Continued)

INSPECT RADIO ALTIMETER ANTENNA INTERIOR AND EXTERIOR DOUBLERS, AND FUSELAGE SKIN IN THE AREA

1. Inspect fuselage skin around Radio Altimeter Antennas for paint cracks, bubbles, loose rivets, or any visible sign of corrosion.

NOTE

The Radio Altimeter Antennas are normally located F.S. 333 to F.S. 390 on the belly of the aircraft. Various types and shapes of antennas are being used in the Sabreliner fleet.

2. Gain access to the aft fuselage through access panels and doors. See Figure 53-00-627 for reference.
3. Inspect the stainless steel skin doublers for both the forward and aft antennas. Check for sealer cracks, bubbles, loose rivets, or any sign of corrosion in between the doubler and the fuselage skin.

NOTE

Any repairs accomplished should be made in accordance with Sabreliner Structural Repair Manual (NA-66-1032) and/or Sabreliner Corrosion Control Manual (SR-76-23).

INSPECT FUSELAGE STRUCTURE BELOW LUGGAGE, GALLEY AND FORWARD LAVATORY/STORAGE AREA, FS 143 TO 206 RH SIDE AND FS 174 TO 206 LH SIDE, WL -21 TO WL -29 (Ref. S/B 94-1)

Equipment/Tools

- Boroscope with 90° viewing probe

Inspection

1. Remove interior covering (carpet) from BL 10 web, FS 143 to FS 206, WL -21 to WL -29 RH side and FS 172 to FS 206, BL 10, WL -21 to WL -29 LH side. See Figure 53-00-628 (Sheet 1 of 2).
2. Install 4 each boroscope inspection holes as shown in Figure 53-00-628 (Sheet 2 of 2).
3. Inspect close-out area FS 143 to FS 182 RH side, BL 10 to BL 23, WL -21 to WL -29 through existing access opening of FS 162 and FS 178 using flashlight, mirror and boroscope. Inspect close-out area FS 182 to FS 206 RH side, BL 10 to BL 23, WL -21 to WL -29 using boroscope through added inspection holes. Inspect for cracks, corrosion and liquid stains.
4. Inspect close-out area FS 172 to FS 206 LH side, BL 10 to BL 23, WL -21 to WL -29 through added inspection holes using boroscope. Inspect for cracks, corrosion and liquid stains.
5. If any discrepancies are found, remove necessary interior furnishing and close-out doors to gain access. Repair any structural discrepancies in accordance with NA-66-1032 Structural Repair Manual and SR-76-023 Corrosion Control Manual. If repairs exceed scope of Structural Repair Manual, contact Sabreliner Corporation Engineering Department.
6. Reinstall interior furnishing and covering removed in steps 1 and 5.

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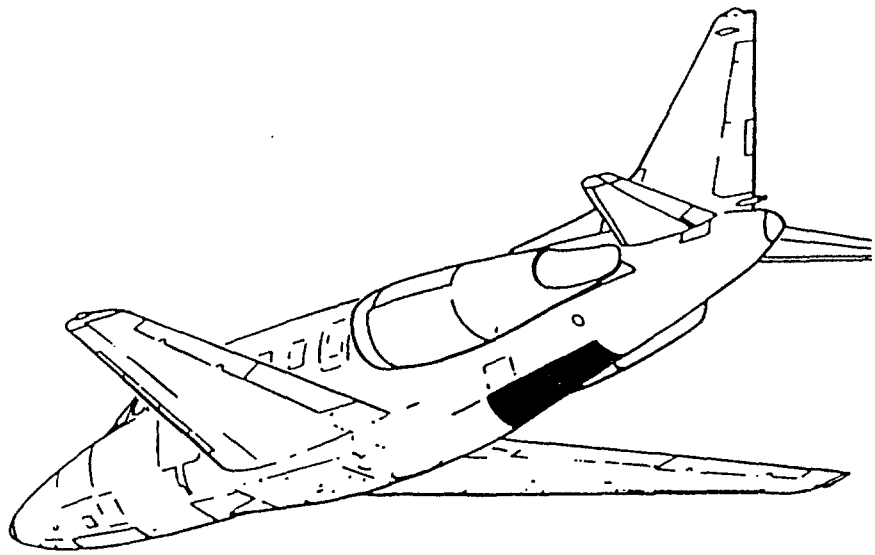

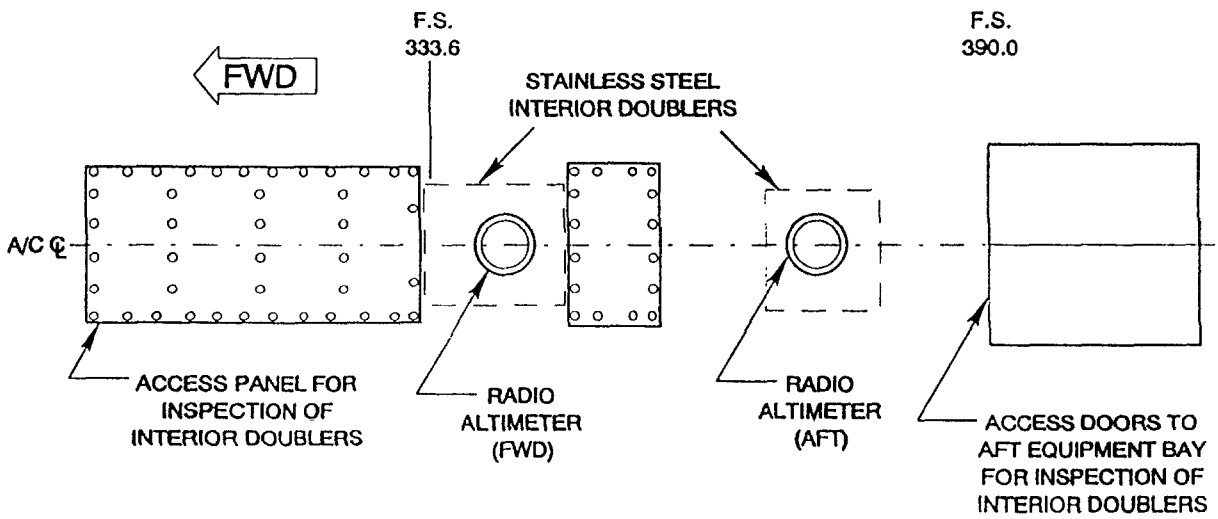

CONFIGURATION MAY VARY BY AIRCRAFT.

Figure 53-00-627 — Inspect Radio Altimeter Antenna, Interior and Exterior Doublers and Fuselage Skin In Area.

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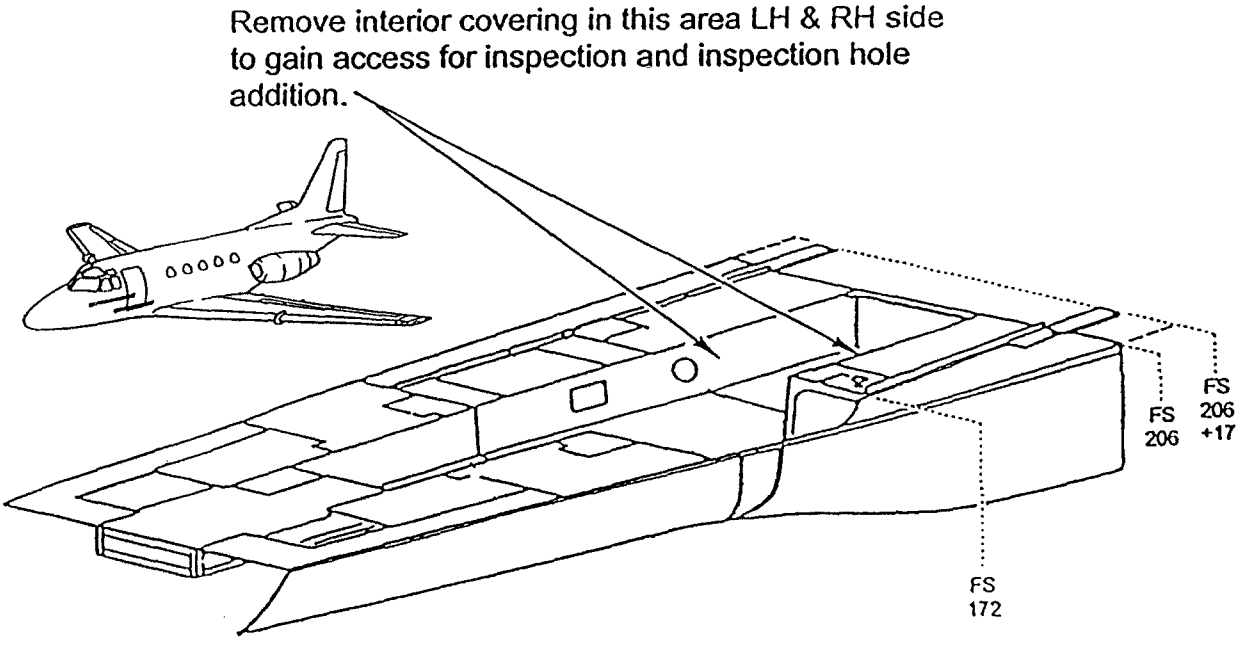
Remove interior covering in this area LH & RH side to gain access for inspection and inspection hole addition.
FS 206
FS 206 +17
FS 172

Figure 53-00-628 — Inspect Fuselage Structure Below Luggage, Galley and Forward Lavatory/Storage Area FS 143 to 206 RH Side and FS 174 to 206 LH Side, WL -21 to WL -29 (Sheet 1 of 2) (Ref: S/B 94-1)

NEW

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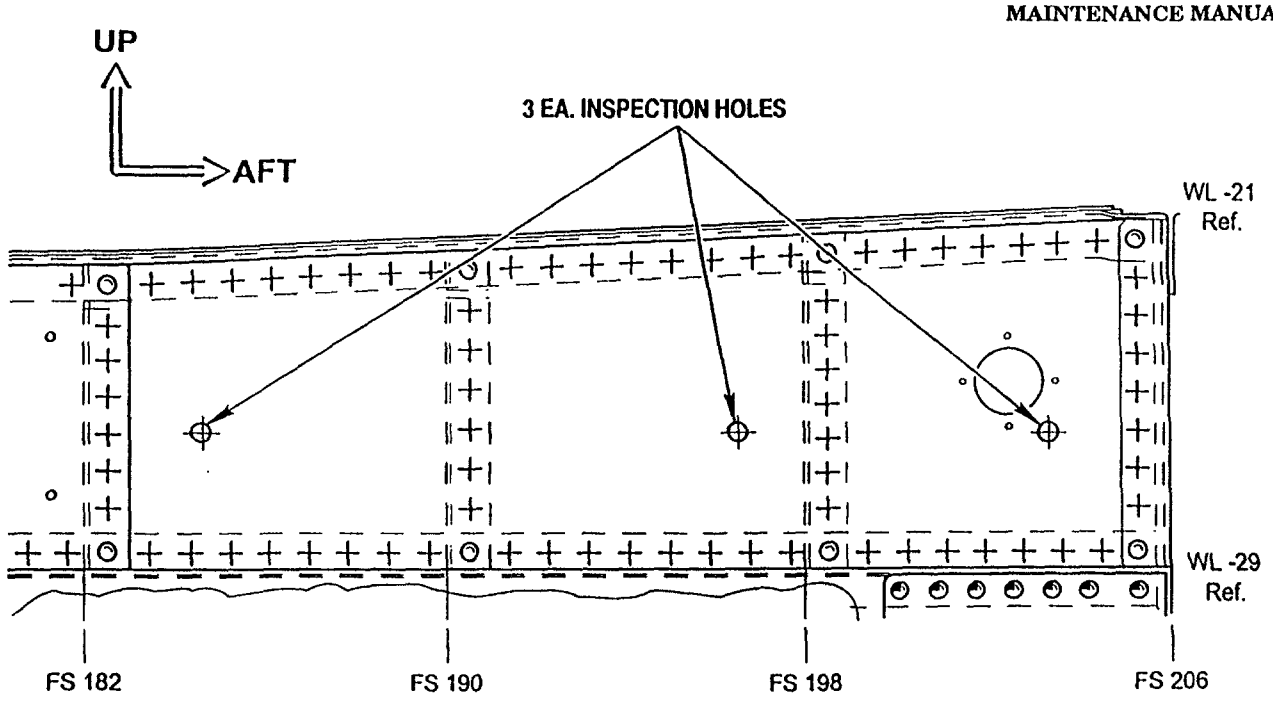
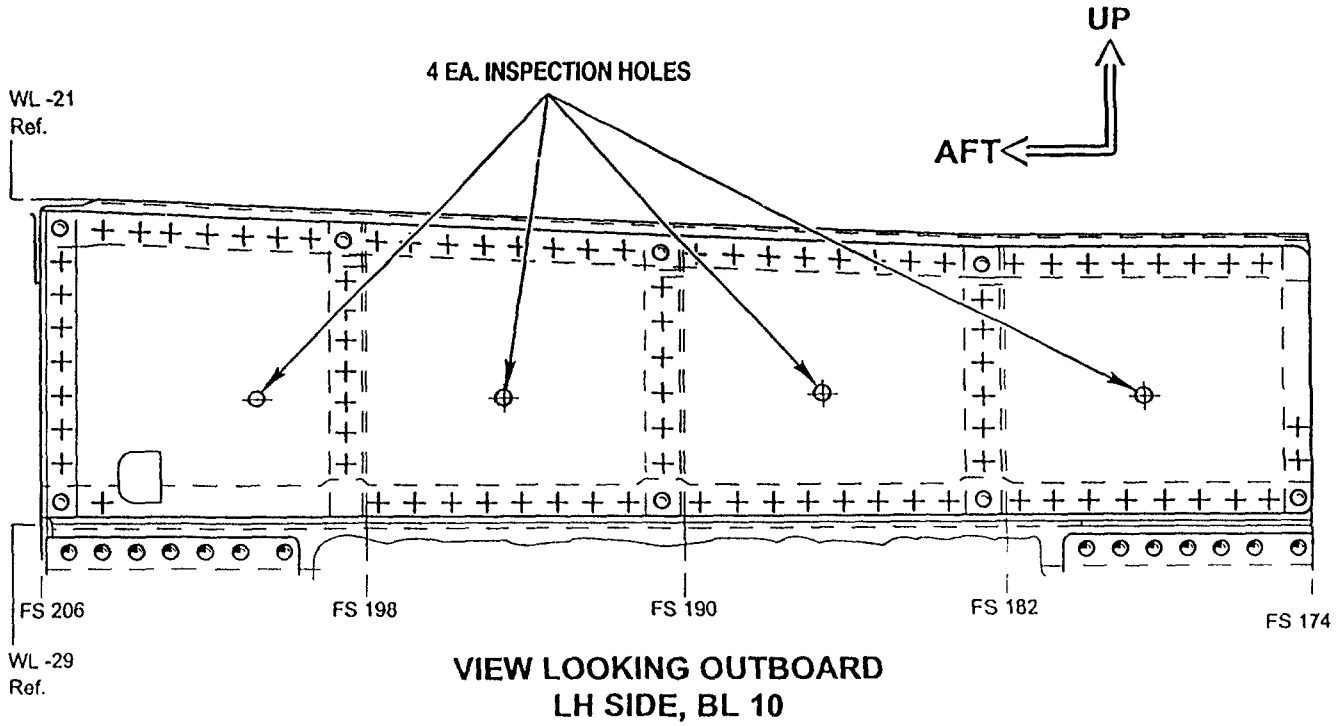


VIEW LOOKING OUTBOARD
RH SIDE, BL 10



VIEW LOOKING OUTBOARD
LH SIDE, BL 10

Figure 53-00-628 — Inspect Fuselage Structure Below Luggage, Galley and Forward Lavatory/Storage Area
FS 143 to 206 RH Side and FS 174 to 206 LH Side, WL -21 to WL -29
(Sheet 2 of 2) (Ref: S/B 94-1)

NEW

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1. Replacing Radome Lightning Diverter Strip

A. To replace radome lightning diverter strip, proceed as follows:

(1) Locate top centerline of radome. Place masking tape 3/4 inch on either side of centerline. Using sandpaper, remove Prestec paint coat from area between masking tape until brown epoxy coat is visible.

(2) Cut through epoxy coat into copper strip. Continue until cut is entire length of strip.

(3) Insert knife blade under cloth covering and gently pry, at one end or the other, until a sufficient area is loosened to provide a pulling surface.

(4) Grasp cloth cover at loosened end and pull from radome. The copper strip should come off with the cloth.

(5) Sandpaper area to be reworked with No. 240 grit and wipe clean with cloth saturated with MEK (methyl ethyl ketone).

(6) Lay 282-300013-31 copper strip assembly into cutout and check for fit. If cutout for strip at station 13-5/16 does not fit, remove excess material.

(7) Clean strip with MEK; then prime with 50-50 solution of general-purpose rubber cement and methyl isobutyl ketone.

(8) Mix and apply a layer of epoxy to cleaned area.

(9) Position copper strip, making sure flat side of strip is on top and built-up area is down. Make sure trailing edge of strip is positioned properly to ensure a good grounding contact.

(10) Cover strip with thin layer of epoxy.

(11) Lay one-inch strip of fiber glass cloth over strip and cover with epoxy.

(12) Cover repaired area with a pressure membrane. Secure membrane with masking tape at both ends.

(13) Cover membrane with grease (any grease will do). Take a wooden tongue depressor (or equivalent) and work topside of membrane until all bubbles and air are out of reworked area. Edges of the reworked area should feather into the radome surface.

(14) Cure reworked area. Curing requires 16 hours at an ambient temperature of 70°F or 2.5 hours in a heated or lighted area at a temperature of 150°F.

(15) Sand repaired area with No. 240 grit until finish is feathered into adjoining areas. Wipe clean with cloth moistened with MEK.

(16) Fill all pinholes, voids, and other slight surface irregularities with Cat-A-Lac white filler putty No. 467-2 (or its equivalent). Allow filler to dry 16 hours or bake at 250°F for 15 minutes.

(17) Wet sand with No. 240 grit, followed with No. 400 grit. Wipe clean with cloth or soft paper toweling.

(18) Apply Prestec No. 2181 (white) surface coating. (Other colors may be used.)

PLATES/SKIN - DESCRIPTION

1. Speed Brake

The speed brake panel is on the lower side of the fuselage, directly aft of the nose wheel well and forward of the main gears. The panel is rectangular and is shaped to fair with the bottom of the fuselage. Hinge arms, attached to the forward side of the panel, attach the panel to the fuselage. Fittings on each side of the panel provide attaching points for the hydraulic actuating cylinders. Structure of the speed brake panel consists of a Chem-Milled outer skin, an inner skin, and angle stiffeners. Cast hinge fittings are attached to the hinge arms on the forward edge of the panel and to the stiffeners on the left and right sides of the panels. These hinge fittings provide attachment points for the hydraulic actuators and fuselage attach fittings.

CHAPTER

54

NACELLES PYLONS

CHAPTER 54

NACELLES/PYLONS

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Inspect — Pylons and Nacelles — S/N 370-1, -7, S/N 380-1 and Subs		601
Inspect — Pylon Attach Fittings F.S. 318 and F.S. 333 — S/N 370-2 thru -6, -8 and -9		602
Inspect — Engine Support Beam Interior (Ref S/B 88-7) (S/N 370-2 thru -6, -8 and -9)		604

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CHAPTER 54

NACELLES/PYLONS

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		603	*	Feb 29/96
	Added	604	*	Feb 29/96
	Added	605	*	Feb 29/96
	Added	606	*	Feb 29/96

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NACELLES/PYLONS

DESCRIPTION

General

On airplanes equipped with the Pratt & Whitney JT12A engine, each engine nacelle group consists of a pylon, a nose section containing the engine air inlet duct, an engine support frame, three upper fairing doors, one lower fairing door, and a fixed stainless-steel fire wall. The nose section assembly is attached directly to the forward face of the engine, forming the engine air inlet duct. The engine support frame, integral with the pylon, carries the main engine mounts as well as the steady-support engine mounts. The complete nacelle structure is cantilevered from the fuselage on the pylon structure. The pylon structure spans fuselage stations 298 and 390. The nacelle is isolated from the pylon by a fumetight, liquid-tight fire wall.

On airplanes equipped with the General Electric CF700-2D-2 engine, each engine nacelle group consists of a pylon, nose section, the upper, lower inboard, and lower outboard access doors, fire seal, aft fairing and engine support structure. The nose section attaches directly to the front face of the engine and guides air to the engine by means of the engine air intake duct mounted internally within the nose section. The remaining intake air is ducted to the aft fan portion of the engine. The three access doors make up the center portion of the nacelle. The upper door and the lower inboard door are hinged at the pylon. The lower outboard door is hinged to the lower edge of the upper door. Each of the access doors have an internal skin attached to longitudinal door frames to provide air passages between the nose section and the engine aft fan housing. The aft fairing is bolted to the forward frame of the aft fan housing and covers the aft fan section of the engine and the engine tailpipe. The thrust reverser is contained in the aft portion of the aft fairing.

The complete nacelle assembly is cantilevered from the pylon by engine support fittings and structural carry-through fittings that extend inboard through the pylons and attach to the fuselage structure.

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NACELLES/PYLONS

INSPECTION/CHECK

NOTE

Additional maintenance practices for the nacelles are covered in Chapter 71.

INSPECT PYLONS AND NACELLES — S/N 370-2 THRU -6, -8 AND -9

1. To inspect pylons, proceed as follows:
   - A. Open access doors and panels on upper and lower surfaces of pylon.
   - B. Exterior skin for cracks, corrosion, loose fasteners, distortion and cleanliness. Loose or missing fasteners.
   - C. Yoke and pylon structure for cracks, distortion, loose or missing fasteners and pylon firewall for proper sealing.
   - D. Fire extinguisher lines for chafing, security and line outlets for obstructions.
   - E. Fuel and PT5 lines for chafing, cuts, security and deterioration.
   - F. Throttle control tubes, bellcranks, push-pull rods and support brackets for security and proper lockwire where applicable.
   - G. Bleed-air duct bellows, clamps, shut off valve, and ducting for cracks, leakage and security.
   - H. Electrical wiring and connector plugs for security, evidence of overheating, and chafing.
   - I. Close and secure all access doors and panels.
2. To inspect nacelles, proceed as follows:
   - A. Remove upper and lower nacelle doors.
   - B. Nacelle skin for cracks, corrosion, loose fasteners, distortion and hot spots.
   - C. Latches and nacelle doors for security.
   - D. Louvers for cracks and distortion.
   - E. Generation inlet and exit ducts for foreign objects and carbon dust which indicates brush wear.
   - F. Check thrust reverser actuating cylinder air scoop for foreign objects, distortion and cracks.
   - G. Inspect nose cone and splitter vanes for cracks, corrosion, loose fasteners, distortion and hot spots.
   - H. Install upper and lower nacelle doors.

INSPECT PYLONS AND NACELLES — S/N 370-1 AND -7 and S/N 380-1 AND SUBS

1. To inspect pylons, proceed as follows:
   - A. Open all access doors and panels on upper and lower surfaces of pylon.
   - B. Exterior skins and removed access panels for cracks, corrosion, loose or damaged fasteners, distortion and cleanliness.
   - C. Pylon structure and outboard web for cracks, distortion, loose or missing fasteners, deteriorated sealant, and damaged rubber seals.
   - D. Fuel heater for cracks and security.
   - E. Fuel heater overboard exhaust for obstructions. Heater compartment drain hole for obstructions.
   - F. Fuel and EPR lines for chafing, cuts, security and deterioration.
   - G. Bleed-air duct bellows, clamps and shutoff valve and ducting for cracks, leakage and security.
   - H. Electrical wiring and connector plugs for security, evidence of overheating and chafing.
   - I. Throttle control tubes and support brackets for security.
   - J. Close and secure access panels.
2. To inspect nacelle doors, air intake duct and engine aft fairing, proceed as follows:
   - A. Remove engine access doors. (Refer to 71-10-01.)
   - B. Remove engine aft fairing access doors. (Refer to 71-10-01.)
   - C. Access doors inner and outer skins for cracks, loose or missing fasteners, distortion and cleanliness.
   - D. Access door hinges, latches and pins for security, cracks, worn or oversize hinge pin holes, and bonding braids for proper attachment.

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INSPECTION/CHECK

INSPECT PYLONS AND NACELLES — S/N 370-1 AND -7 AND S/N 380-1 AND SUBS (Cont'd)

E. Engine aft fairing for cracks, corrosion, loose or missing fasteners, distortion and indications of hot spots.

F. Thrust reverser components and actuating mechanisms for security, corrosion, cracks, heat damage or distortion, foreign objects and cleanliness.

G. Engine air inlet duct for cracks, loose or missing fasteners, corrosion and security of attachment.

INSPECT — PYLON ATTACH FITTINGS F.S. 318 AND F.S. 333 — S/N 370-2 THRU -6, -8 AND -9

Equipment/Tools

- None

Consumables

- None

Inspection

1. Remove inspection access panels. (See Figure 54-00-601.)
2. Clean pylon attach fittings per Sabreliner Corrosion Control Manual (SR-76-23), chapter 20.
3. Inspect pylon attach fittings for cracks and/or corrosion.

NOTE

Any repair accomplished should be made in accordance with Sabreliner Structural Repair Manual (NA-66-1032) and/or Sabreliner Corrosion Control Manual (SR-76-23).

4. Install and secure access panels.

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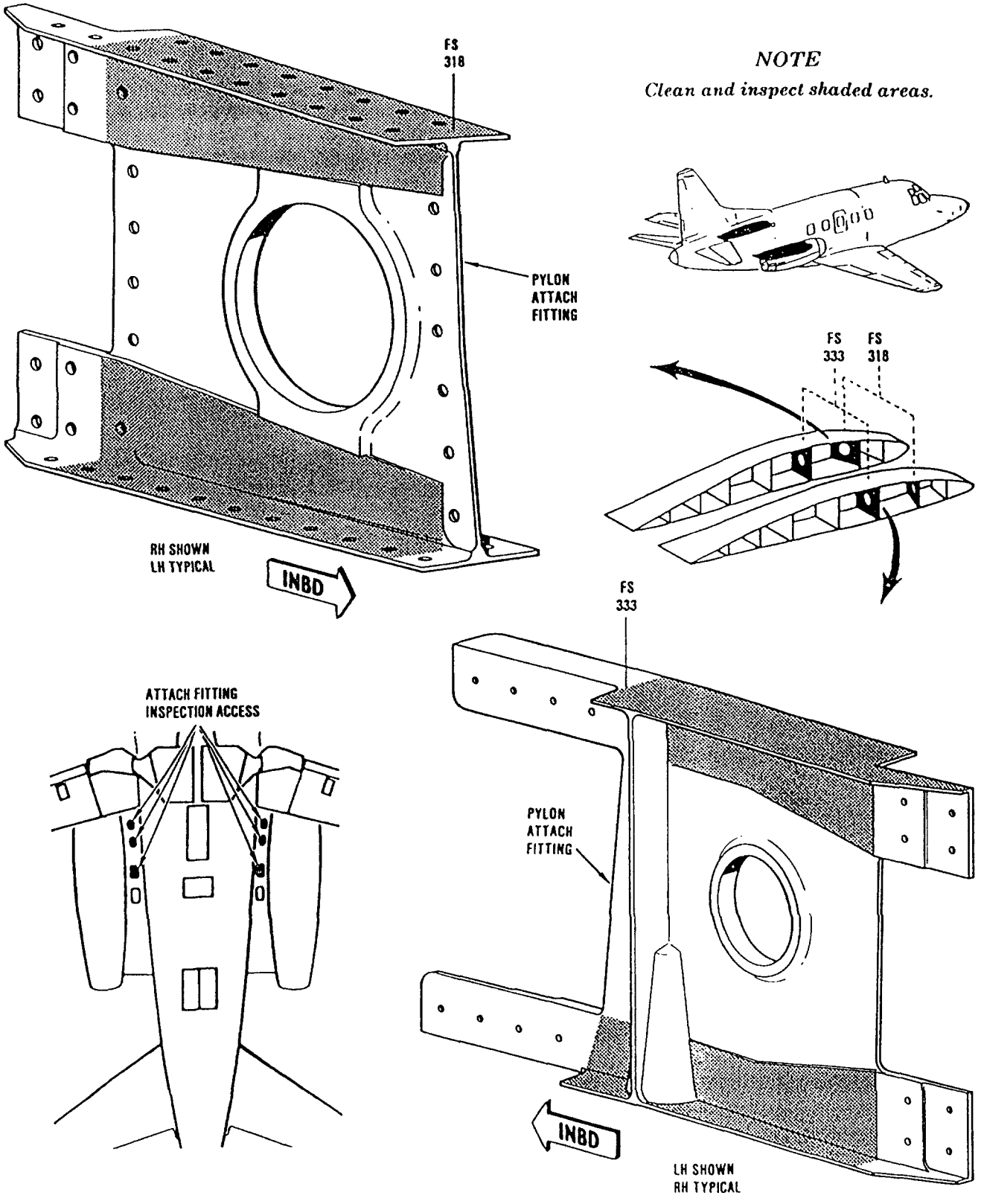

Figure 54-00-601 — Inspect — Pylon Attach Fittings F.S. 318 and F.S. 333

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INSPECTION/CHECK (Continued)

INSPECT — ENGINE SUPPORT BEAM INTERIOR (Ref: S/B 88-7) (S/N 370-2 thru -6, -8 and -9)

Equipment/Tools

- .312 inch drill
- Rod Optic Borescope (Right Angle) and Power Supply

Consumables

- None

Procedure

1. Remove lower LH and RH nacelle covers per steps A thru E.
   A. Open latches on LH and RH engine nacelle covers. See Figure 54-00-602 (Sheet 1 of 2).
   B. Disconnect the two (per engine) fire detector wires at the inboard side of the nacelle.
   C. Remove lockpin from firewall end of sway brace. Stow lockpin in free end of sway brace.
   D. Slide locking sleeve away from joint; fold sway brace and stow in bottom of lower engine nacelle cover.
   E. With nacelle cover in two-thirds full open position, raise nacelle cover slightly and move it inboard to lift it off hinges.

NOTE
If S/B 88-7 has been previously complied with, step 2 will not be necessary. Proceed to step 3.

2. Using a .312 inch drill, make holes in the inboard part of the engine support beam. See Figure 54-00-602 (Sheet 2 of 2).
3. Make sure existing holes are clear of sealant. See Figure 54-00-602 (Sheet 2 of 2).
4. Using a borescope, inspect the engine support beam interior for corrosion or cracks. Especially look for cracks in the outboard flanges of the forward and aft pylon castings.
5. If no corrosion or cracks are found, install lower LH and RH engine nacelle covers removed in step 1.

NOTE
Do not install sealant over inspection holes.

6. If corrosion or cracks are found, contact Sabreliner Corporation Scheduling, Pierre Laclede Center, 7733 Forsyth Boulevard, Suite 1500, St. Louis, MO 63105-1821 or phone 314-863-6880.

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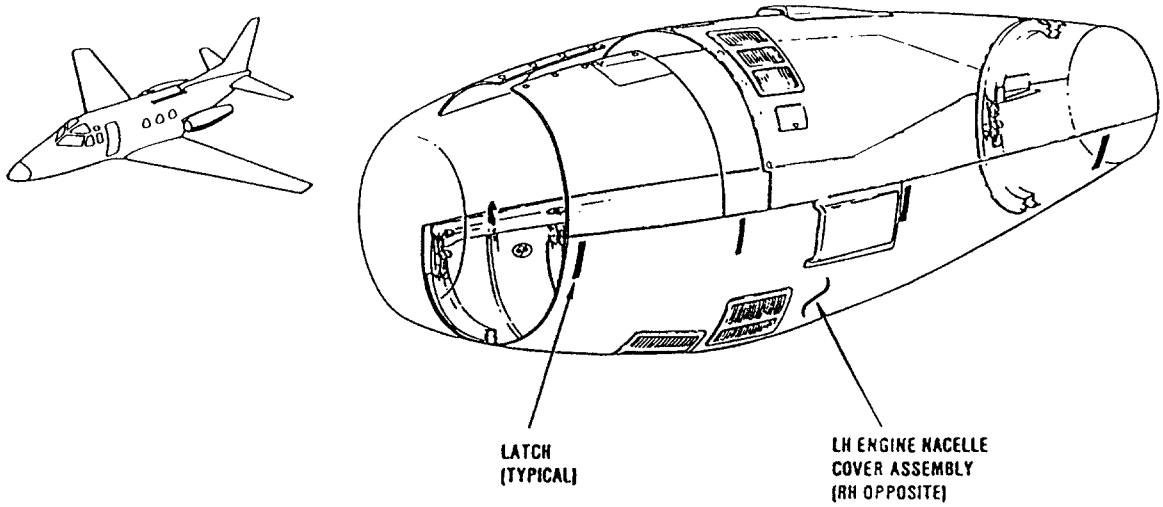

NACELLE COVER ASSEMBLY

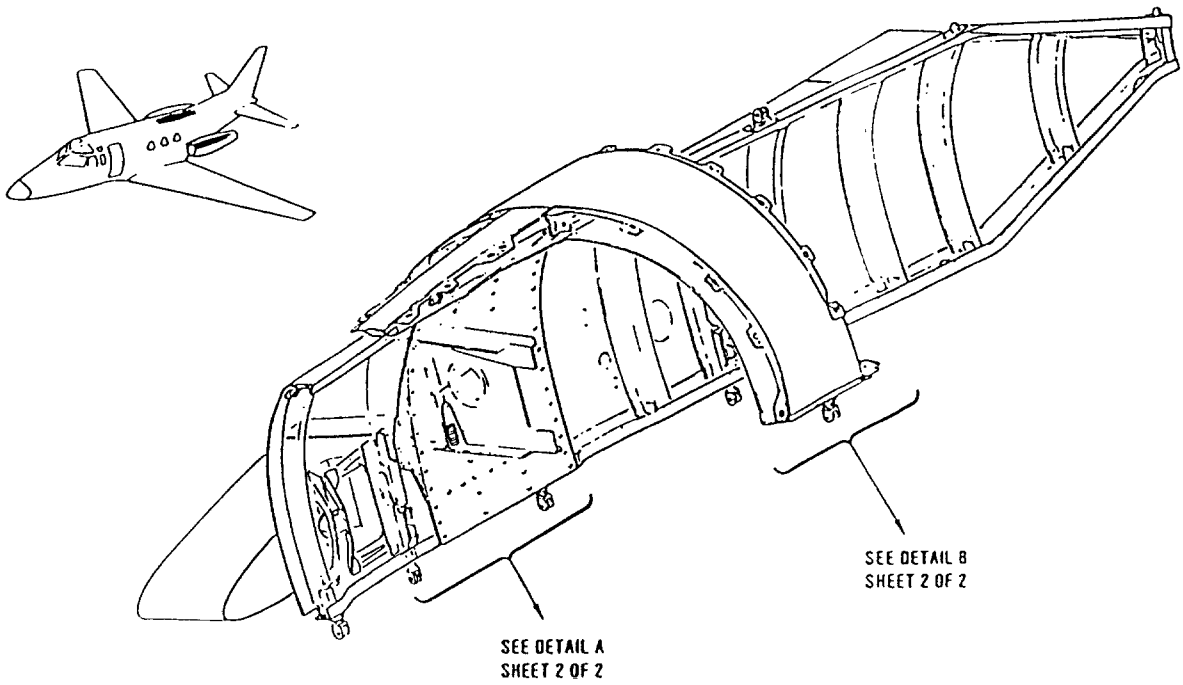

Figure 54-00-602 — Engine Support Beam (Sheet 1 of 2)

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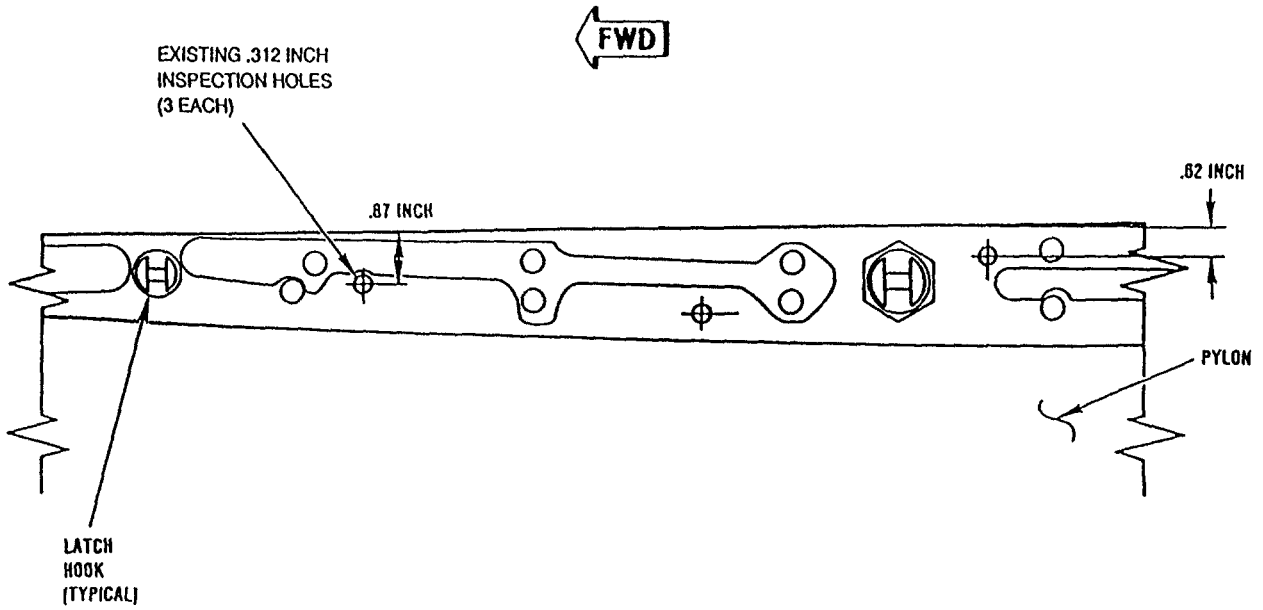
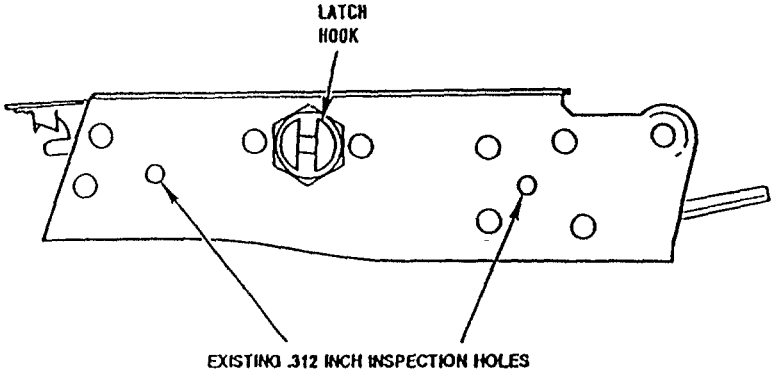


DETAIL A
VIEW LOOKING UP
LH SUPPORT BEAM
INBOARD



DETAIL B
VIEW LOOKING UP
LH SUPPORT BEAM
OUTBOARD

NEW

Figure 54-00-602 — Engine Support Beam (Sheet 2 of 2)

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CHAPTER

55

STABILIZERS

CHAPTER 55

STABILIZERS

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MAJOR CHANGE

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CHAPTER 55

STABILIZERS

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STABILIZERS

DESCRIPTION

GENERAL

The empennage consists of a fixed vertical stabilizer, a rudder, a rudder trim tab, a trimmable horizontal stabilizer, an elevator, and fillets. The vertical and horizontal stabilizers are swept back 30 degrees at the 25 percent plane.

REPAIRS

Repairs must be accomplished in accordance with Sabreliner Structural Repair Manual (Report No. NA-66-1032) instructions or by Federal Aviation Agency approved repair procedures for safe life structure.

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HORIZONTAL STABILIZERS

DESCRIPTION

GENERAL

The horizontal stabilizer is of full cantilever construction. It consists of two aerodynamically neutral panels, connected by a pivoted beam through the fuselage. The complete assembly rotates about a pivot axis at FS 466.53 and water plane 20 for longitudinal trim. The structure is a single-cell box with a rear beam, two forward longitudinal stringers, and transverse ribs. The structure has upper and lower skins, a leading edge skin, and a tip. The upper and lower skins are Chem-Milled.

Because the horizontal stabilizer is movable to provide for longitudinal trim of the airplane, maintenance practices are covered in chapter 27.

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ELEVATOR - DESCRIPTION

1. General

The elevators are neutral interchangeable panels, interconnected through a torque tube. The structure is a single-cell torque box consisting of a rear beam, two longitudinal stringers, and transverse ribs. The rear beam incorporates a sheet-metal channel and a honeycomb sandwich panel trailing edge. The structure is covered with upper and lower skins, a leading edge skin, and a tip. The elevator is suspended from the stabilizer rear beam by three hinges. The elevator is both aerodynamically and statically balanced. Static balance is achieved by a series of steel weights attached to the inner side of the leading edge skin.

The maintenance practices for the elevator are covered in Chapter 27.

VERTICAL STABILIZER

DESCRIPTION

GENERAL

The vertical stabilizer is an all-metal, fully cantilevered structure. The main box structure consists of a full-length rear beam, a short lower front beam, transverse ribs, two longitudinal stringers, and side and leading edge skins. The box section above the rudder is formed by the rear beam, a honeycomb panel trailing edge, and side skins. The side skins are Chem-Milled. The vertical stabilizer is attached to the fuselage by 53 bolts at the rear beam, rivets on the left and right sides, and one bolt at the front beam. Provisions are made for the installation of inflatable deicer boots on the stabilizer leading edge. Two pitot tubes, for the engine pressure ratio system, are installed between stabilizer stations 33 and 40 on the stabilizer leading edge.

The stabilizer is faired into the fuselage by means of a dorsal fin and fillet. The leading edge of the dorsal fin provides a ram-air inlet for the airplane air-conditioning system. Anti-icing of this inlet is provided by an electrically heated Spraymat coating applied to the forward edge of the inlet duct. Airplanes modified by SOK 10 have a spring loaded bird-stopper flapper door located inside the dorsal fin ram-air inlet.

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VERTICAL STABILIZER - MAINTENANCE PRACTICES

1. Removing Vertical Stabilizer (See figure 55-30-201.)

A. Parts referred to in this procedure are identified in figure 55-30-201. To remove vertical stabilizer, proceed as follows:

(1) Remove access panels from sides of fuselage rear section as shown in figure 55-30-201, and remove 370-250001-11 fillet on dorsal of vertical stabilizer.

(2) Disconnect and cap deicer boot air lines and pitot lines at points between stabilizer stations 8 and 17.

(3) Remove fairing around engine pressure ratio system pitot tubes, on stabilizer leading edge, and disconnect pitot heater electrical wiring.

(4) Disconnect pitot heater ground wire at connection just below lower pitot head, and remove clamps as necessary. Pull wiring down through stabilizer leading edge.

(5) Remove screws securing 370-250012-1, 370-2550012-2, 370-250013-11, and 370-250014-11 formers to stabilizer front spar.

(6) Remove rudder. (Refer to Chapter 27.)

(7) Disconnect position light at terminal inside access on left side of fuselage, and remove clamps as necessary.

(8) Disconnect VOR/localizer antenna cable forward of station 443 and remove clamps as necessary.

(9) Remove screws securing vertical stabilizer to 265-313115-1 and -2 fillets, and remove fillets.

(10) Remove 265-250003-1 and -2 fillets from both sides of airplane.

NOTE: The 265-250003-1 and -2 fillets are secured to the stabilizer and fuselage with MS20601-4 rivets.

(11) Remove two bolts securing stabilizer rear spar to 370-240417 support.

NOTE: For the remaining steps of this procedure, two assistants will be required to support, and assist in the removal of, the vertical stabilizer.

(12) Remove the 53 bolts securing stabilizer rear spar to fuselage.

(13) Remove the one bolt securing stabilizer front spar to fuselage.

(14) Remove stabilizer. The stabilizer weighs about 75 pounds.

2. Installing Vertical Stabilizer (See figure 55-30-201.)

A. Parts referred to in this procedure are shown in figure 55-30-201. To install vertical stabilizer, proceed as follows:

(1) Have assistants position stabilizer on fuselage and hold in place.

(2) Install bolt, securing stabilizer front spar to fuselage.

(3) Install the 53 bolts securing stabilizer rear spar to fuselage. Install the two bolts securing stabilizer to 370-240417 support.

(4) Install 265-313115-1 fillet on left side of airplane; then, using screws, secure vertical stabilizer to right and left fillets.

(5) Install 265-250003-1 and -2 fillets on vertical stabilizer and fuselage, using MS20601-4 flush-type, mechanical-expanding blind rivets.

NOTE: If rivet holes have become elongated or are oversize, use next size larger rivets.

(6) Route and connect VOR/localizer antenna cable and install clamps.

(7) Route and connect position light wiring and install clamps.

(8) Install rudder. (Refer to Chapter 27.)

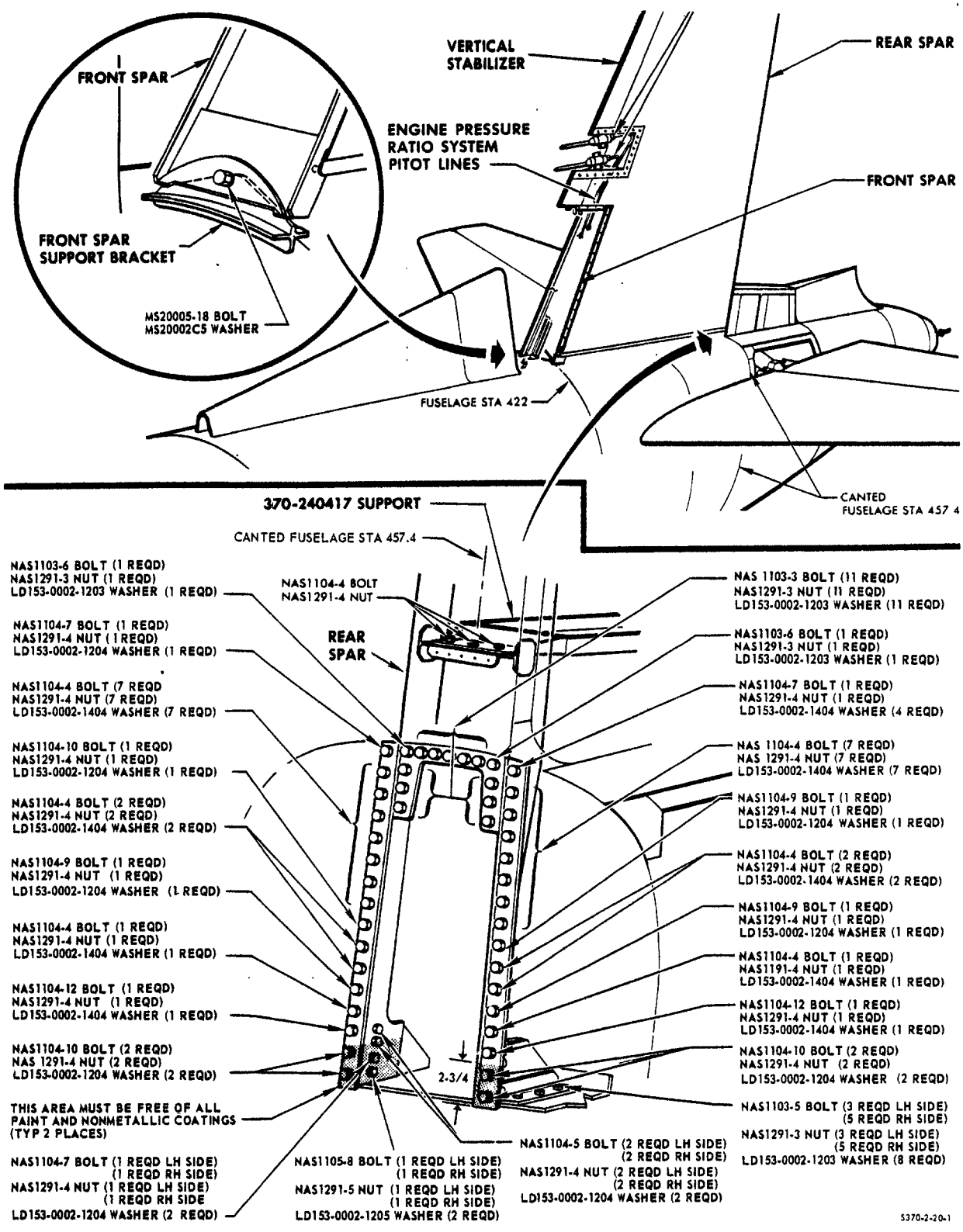
(9) Install screws attaching 370-250012-1, 370-250012-2, 370-250013-11, and 370-250014-11 formers to vertical stabilizer front spar.

(10) Route and connect pitot heater electrical wiring, and install clamps on wiring as shown in figure 55-30-201.

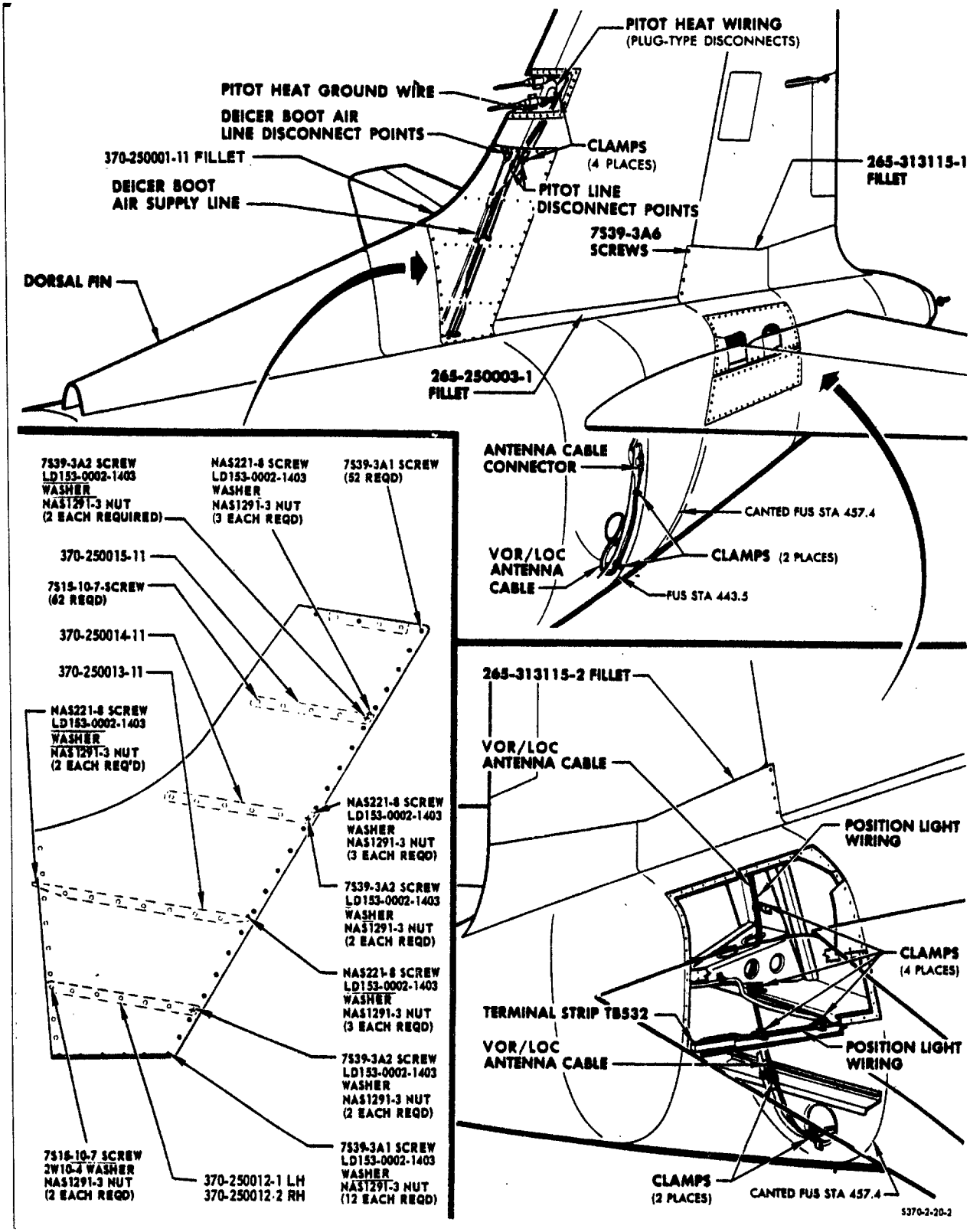
(11) Install pitot tube access fairing on stabilizer leading edge.

(12) Connect deicer boot air supply lines and pitot lines.

(13) Install 370-250001-11 fillet on dorsal fin of stabilizer, and install access panels on fuselage.


FRONT SPAR
FRONT SPAR SUPPORT BRACKET
MS20005-18 BOLT
MS20002C5 WASHER
VERTICAL STABILIZER
ENGINE PRESSURE RATIO SYSTEM PITOT LINES
REAR SPAR
FRONT SPAR
FUSELAGE STA 422
CANTED FUSELAGE STA 457 4
370-240417 SUPPORT
CANTED FUSELAGE STA 457.4
NAS1103-6 BOLT (1 REQD)
NAS1291-3 NUT (1 REQD)
LD153-0002-1203 WASHER (1 REQD)
NAS1104-7 BOLT (1 REQD)
NAS1291-4 NUT (1 REQD)
LD153-0002-1204 WASHER (1 REQD)
NAS1104-4 BOLT (7 REQD
NAS1291-4 NUT (7 REQD)
LD153-0002-1404 WASHER (7 REQD)
NAS1104-10 BOLT (1 REQD)
NAS1291-4 NUT (1 REQD)
LD153-0002-1204 WASHER (1 REQD)
NAS1104-4 BOLT (2 REQD)
NAS1291-4 NUT (2 REQD)
LD153-0002-1404 WASHER (2 REQD)
NAS1104-9 BOLT (1 REQD)
NAS1291-4 NUT (1 REQD)
LD153-0002-1204 WASHER (1 REQD)
NAS1104-4 BOLT (1 REQD)
NAS1291-4 NUT (1 REQD)
LD153-0002-1404 WASHER (1 REQD)
NAS1104-12 BOLT (1 REQD)
NAS1291-4 NUT (1 REQD)
LD153-0002-1404 WASHER (1 REQD)
NAS1104-10 BOLT (2 REQD)
NAS 1291-4 NUT (2 REQD)
LD153-0002-1204 WASHER (2 REQD)
THIS AREA MUST BE FREE OF ALL PAINT AND NONMETALLIC COATINGS (TYP 2 PLACES)
NAS1104-7 BOLT (1 REQD LH SIDE) (1 REQD RH SIDE)
NAS1291-4 NUT (1 REQD LH SIDE) (1 REQD RH SIDE
LD153-0002-1204 WASHER (2 REQD)
NAS1104-4 BOLT
NAS1291-4 NUT
REAR SPAR
NAS1105-8 BOLT (1 REQD LH SIDE) (1 REQD RH SIDE)
NAS1291-5 NUT (1 REQD LH SIDE) (1 REQD RH SIDE)
LD153-0002-1205 WASHER (2 REQD)
2-3/4
NAS1104-5 BOLT (2 REQD LH SIDE) (2 REQD RH SIDE)
NAS1291-4 NUT (2 REQD LH SIDE) (2 REQD RH SIDE)
LD153-0002-1204 WASHER (2 REQD)
NAS 1103-3 BOLT (11 REQD)
NAS1291-3 NUT (11 REQD)
LD153-0002-1203 WASHER (11 REQD)
NAS1103-6 BOLT (1 REQD)
NAS1291-3 NUT (1 REQD)
LD153-0002-1203 WASHER (1 REQD)
NAS1104-7 BOLT (1 REQD)
NAS1291-4 NUT (1 REQD)
LD153-0002-1404 WASHER (4 REQD)
NAS 1104-4 BOLT (7 REQD)
NAS 1291-4 NUT (7 REQD)
LD153-0002-1404 WASHER (7 REQD)
NAS1104-9 BOLT (1 REQD)
NAS1291-4 NUT (1 REQD)
LD153-0002-1204 WASHER (1 REQD)
NAS1104-4 BOLT (2 REQD)
NAS1291-4 NUT (2 REQD)
LD153-0002-1404 WASHER (2 REQD)
NAS1104-9 BOLT (1 REQD)
NAS1291-4 NUT (1 REQD)
LD153-0002-1204 WASHER (1 REQD)
NAS1104-4 BOLT (1 REQD)
NAS1191-4 NUT (1 REQD)
LD153-0002-1404 WASHER (1 REQD)
NAS1104-12 BOLT (1 REQD)
NAS1291-4 NUT (1 REQD)
LD153-0002-1404 WASHER (1 REQD)
NAS1104-10 BOLT (2 REQD)
NAS1291-4 NUT (2 REQD)
LD153-0002-1204 WASHER (2 REQD)
NAS1103-5 BOLT (3 REQD LH SIDE) (5 REQD RH SIDE)
NAS1291-3 NUT (3 REQD LH SIDE) (5 REQD RH SIDE)
LD153-0002-1203 WASHER (8 REQD)
S370-2-20-1

Vertical Stabilizer Installation
Figure 55-30-201 (Sheet 1 of 2)



Vertical Stabilizer Installation
Figure 55-30-201 (Sheet 2 of 2)

VERTICAL STABILIZER

INSPECTION/CHECK

INSPECT — VERTICAL STABILIZER REAR SPAR CAP

Equipment/Tools

- None

Consumables

- None

Inspection

1. Make sure there is no hydraulic and electrical power applied.
2. Inspect vertical stabilizer rear spar cap and surrounding skins for cracks and/or corrosion. (See Figure 55-30-601.)

NOTE

Any repairs accomplished should be made in accordance with Sabreliner Structural Repair Manual (NA-66-1032) and/or Sabreliner Corrosion Control Manual (SR-76-23).

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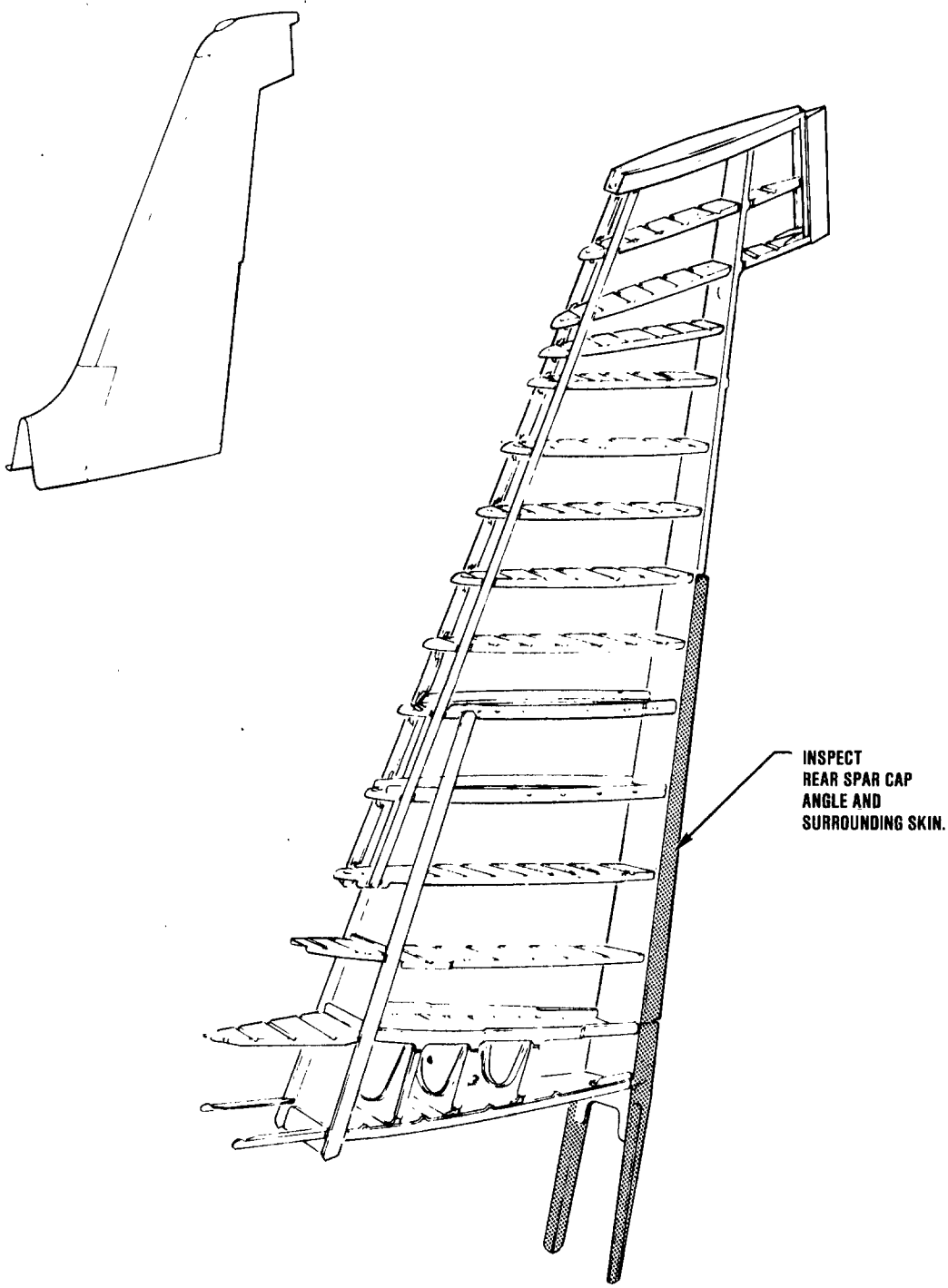

Figure 55-30-601 — Inspect — Vertical Stabilizer Rear Spar Cap

NEW

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RUDDER - DESCRIPTION

1. General

The rudder is a single-cell torque box consisting of a rear beam, two forward longitudinal stringers, transverse ribs, and side and leading edge skins. The rear beam consists of a sheet-metal channel and a honeycomb sandwich panel trailing edge, except in the area of the trim tab. The trim tab is a honeycomb sandwich panel structure with a sheet-metal channel leading edge.

The rudder is suspended from the stabilizer rear beam by three hinges and is rotated by a torque tube attached to the root rib. The rudder is both aerodynamically and statically balanced. Static balance is achieved by a steel weight attached to the inside of the leading edge skin. The trim tab is statically balanced and rotates ±15 degrees.

The maintenance practices for the rudder are covered in Chapter 27.

CHAPTER 56

WINDOWS

CHAPTER 56

WINDOWS

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WINDOWS

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CHAPTER 56

WINDOWS

LIST OF EFFECTIVE PAGES

THIS ENTIRE CHAPTER HAS BEEN REVISED

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*The asterisk indicates pages revised, added, or deleted by the current revision.

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WINDOWS

DESCRIPTION

GENERAL

Cockpit

The cockpit window group consists of two windshield panels, two forward overhead windows, one sliding direct-vision window, two fixed side windows, and two small side overhead windows. The pilot's sliding window is a tool-controlled assembly. (Special tooling is required for acrylic panel replacement.) The windshield panels, forward overhead windows, and direct-vision windows are electrically heated to provide anti-icing, antifogging, and bird impact proofing. The side windows and side overhead windows are all of double-paneled construction, and are connected to a dehydrator system.

Cabin

The cabin windows are of double-paneled acrylic construction and are connected to the dehydrator system, with the exception of those mounted in escape hatches. Escape hatch windows have independent dehydrating systems.

DEHYDRATOR SYSTEM

(See Figure 56-00-1)

The dehydrator system uses two capsules containing a moisture-absorbing chemical to dehydrate the air gap between cockpit and cabin inner and outer window panels. The capsules are connected by tubing and hoses to all of the cabin and cockpit windows, with the exception of the windshield panels, the forward overhead windows, and direct-vision windows in the cockpit and the two escape hatch windows. Dehydrating systems for the escape hatch windows consist of a pair of dehydrator capsules in each hatch.

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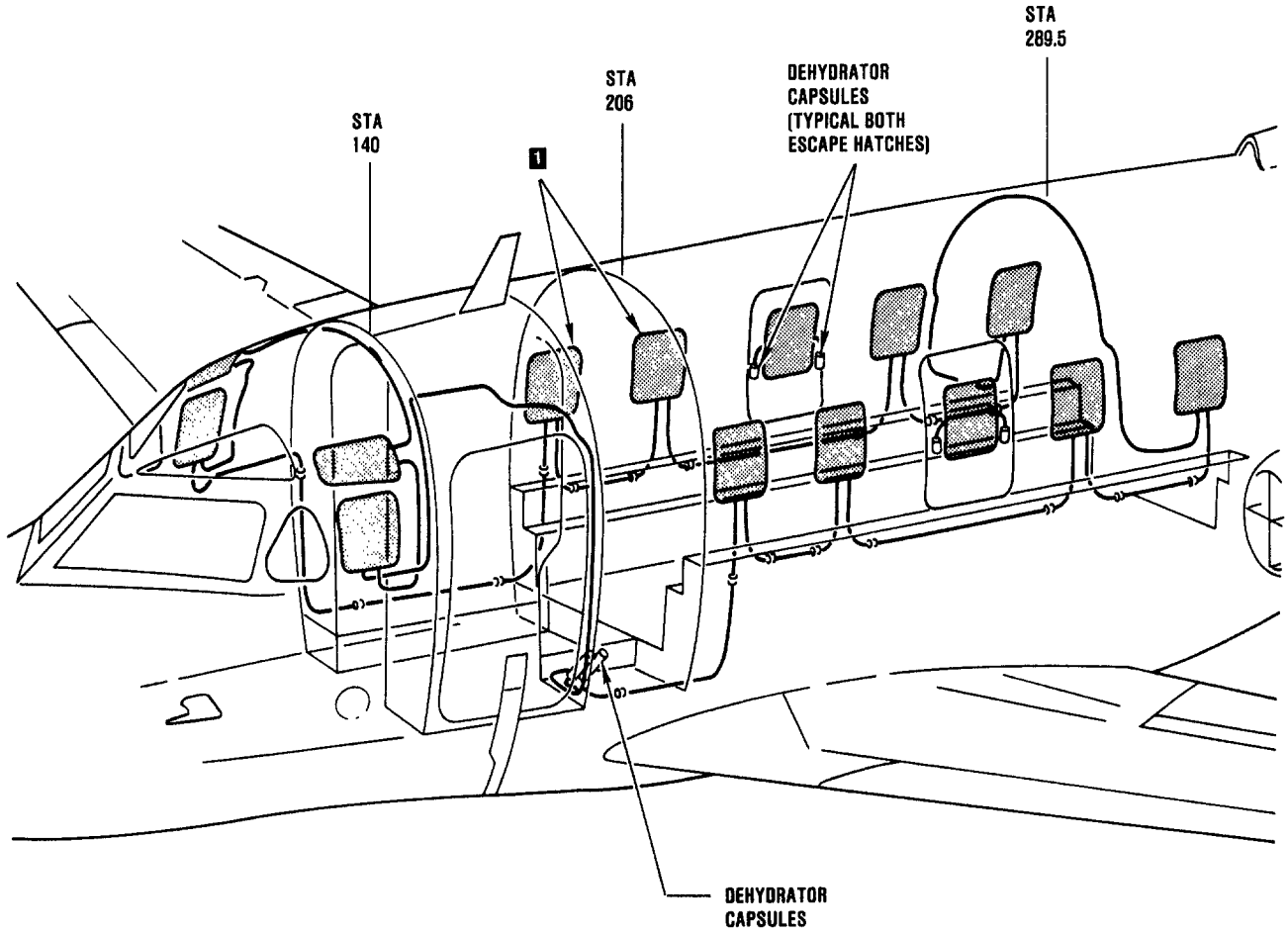

1 NOT INSTALLED ON AIRWAY CHECKER AIRPLANES

S370-2-31-6A

Figure 56-00-1 — Cockpit and Cabin Window Dehydrator System

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WINDOWS

MAINTENANCE PRACTICES

PREPARATION OF WINDSHIELD, COCKPIT, AND CABIN WINDOW SEALING COMPOUNDS

Sealing compounds consist of a base compound and a catalyst. They should be mixed in proportions recommended by the manufacturer. If the sealant is kept at room temperature (approximately 77°F), it should be used within 3 hours after mixing; if the sealant is stored at 0°F, it should be used within 72 hours after mixing. The curing time (before aircraft use) for the sealants is 8 or 24 hours depending on the type of sealant used.

CAUTION

Structural damage to stretched-acrylic window panels may occur if heat is applied for curing. Do not heat over 200°F.

CAUTION

The pilot's sliding window may be damaged if subjected to high temperature during heat curing procedures of adjacent window panels.

Sealing Compounds: PR-1223B½ thru -4 depending on application condition.
PR-1221B½ thru -4 depending on application condition.
PR-1725B½ thru -4 depending on application condition.

Compounds used as a moisture-proofing and aerodynamic filler and for exterior use only are as follows:

PR-340
PR-341 Products Research Co.
PS-247 Los Angeles, CA
PS-895

REPLACING DEHYDRATORS

The main cabin window dehydrator system capsules are near the floor on the left side just aft of the entrance door. Escape hatch window capsules are inside each escape hatch. The dehydrator capsules contain a silica gel desiccant to absorb any moisture in the air trapped between panels of the cockpit and cabin windows. As the silica gel desiccant changes from deep blue (dry condition) to opaloid or bright pink (moisture-saturated), the capsules should be replaced as an assembly or, in the case of main cabin window capsules, the desiccant should be replaced.

Replacement of Main Cabin Window Dehydrators
(See Figure 56-00-1)

1. Gain access to dehydrator capsules.
2. Loosen B nut on elbow fitting at bottom of capsule, and unscrew capsule from fitting.
3. Remove air dryer lines and unions from contaminated capsules and install them on replacement capsules.

NOTE

Capsules may be installed with inlet or outlet fittings in either end.

4. Install O-ring on elbow fitting, screw capsule onto fitting, and tighten B nut.

NOTE

If dehydrators are supported only by the inlet line fitting on the bottom of the dehydrator, some additional means of support must be provided.

5. Install applicable panel or door.

NOTE

Make sure that bushing is installed on elbow fitting beneath B nut.

Replacement of Escape Hatch Capsules

1. Remove reveal from around window.
2. Cut lockwire securing air-dryer line to capsule and remove hose.
3. Loosen clamp securing capsule to hatch structure and remove capsule.
4. Position new capsule in clamp and connect air-dryer line. Secure line to capsule with double wrap of 0.032-inch-diameter Inconel lockwire.
5. Tighten capsule-retaining clamp.

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MAINTENANCE PRACTICES (Continued)

SERVICING DEHYDRATORS

NOTE

The following parts are required for servicing the dehydrators: Seal P/N 5212-51, 568-212, or AN6227-17; and Silica Gel, Specification MIL-D-3716, Grade H, Type IV. The 5212-51 seal may be purchased from Robbins Aviation, 2350 East 38th Street, Los Angeles, CA 90038. The 568-212 seal may be purchased from Plastics, 2100 Hyde Park Blvd., Los Angeles, CA 90016.

1. Remove dehydrator capsules from airplane.
2. Disassemble capsules by removing removable end.
3. Remove fiberglass filter pad and retain for reinstallation.
4. Remove desiccant.
5. Remove seal and replace.
6. Service with desiccant.
7. Reassemble capsules and reinstall.

NOTE

If dehydrators are supported only by the inlet line fitting on the bottom of the dehydrator, some additional means of support must be provided.

REMOVING MOISTURE BETWEEN INNER AND OUTER PANELS OF CABIN WINDOWS

NOTES

- ***This procedure will ensure that the dehydrator system is open, preventing pressure entrapment between the panes, and the possibility of cracked windows. The procedure should be conducted periodically and after installation of an upholstered panel that could contact the dehydrator tubing.***
- ***While windows are pressurized, it may be possible to locate external leaks by using a mild soap solution applied to the exterior surface of the windows.***

1. To gain access to dehydrators, remove cover panel near floor on left side of cabin, aft of entrance door.
2. Disconnect and remove dehydrators.
3. See Figure 56-00-1 to determine direction of flow necessary to place moisture-laden window downstream of most windows.
4. Connect regulated source of dry nitrogen to upstream dehydrator fitting. Connect overboard drain hose to opposite fitting.
5. Carefully apply a maximum of 2 psi nitrogen pressure, and check flow from opposite fitting. If flow is obstructed, investigate and correct.

CAUTION

Do not exceed 2 psi, or cracked windows will result.

6. Continue flow until all windows are dry and free of moisture. Shut off nitrogen supply. Disconnect and remove equipment.
7. Reinstall serviceable dehydrators and safety with lockwire. Install cover panel removed for access.

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FLIGHT COMPARTMENT

MAINTENANCE PRACTICES

REMOVING FIXED WINDOWS

NOTE

The following procedures are typical for all fixed windows or windshield assemblies.

1. Remove cockpit reveal from affected area.
2. Cut sealant away from structure and window.
3. Remove windshield wiper if necessary.
4. Disconnect windshield anti-ice wiring if necessary.
5. Remove attaching fasteners and retainers from windshield and/or window panels.

NOTE

If numerous panel retainers or splices are to be removed, make sure they are identified to correspond with their proper location.

6. Remove panel and place on padded surface to prevent scratching.
7. Push out spacers from around panel edges.

NOTE

Four of the cockpit windows are two-part assemblies (see Figure 56-00-1). On these windows, use a thin putty knife, or equivalent, to carefully separate inner and outer panels.

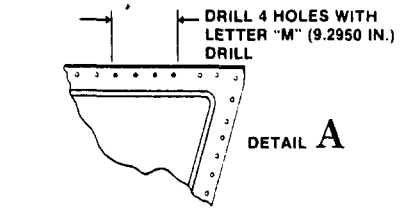
INSTALLING FIXED WINDOWS

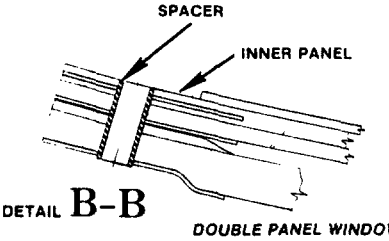
CAUTION

The spacers, when installed on panel assembly, must extend slightly beyond faying surface. This is to prevent fixed structures from exerting clamping pressure on the glass panels. If necessary remove material from inbd side of inner panel to assure spacers are slightly extended, and fixed structure does not bear on glass panels at or between fasteners.

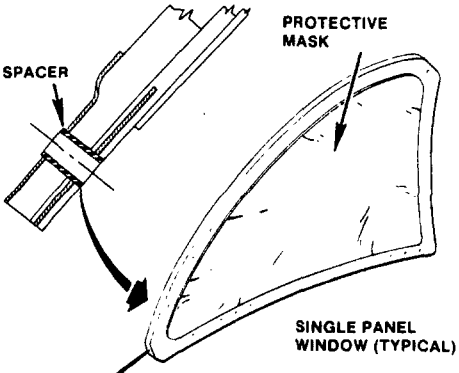
CAUTION

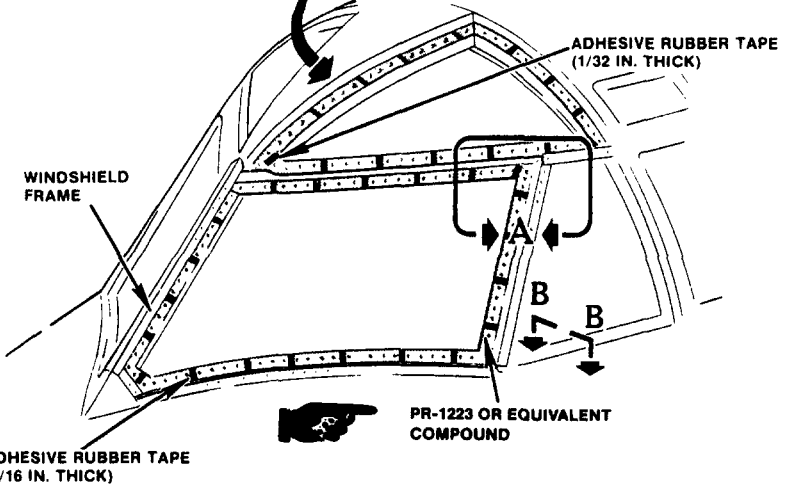
Make a trial fit of new panel in windshield frame. File windshield where necessary for clearance.





DOUBLE PANEL WINDOW (TYPICAL)





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MAINTENANCE PRACTICES (Continued)

INSTALLING FIXED WINDOWS (Cont'd)

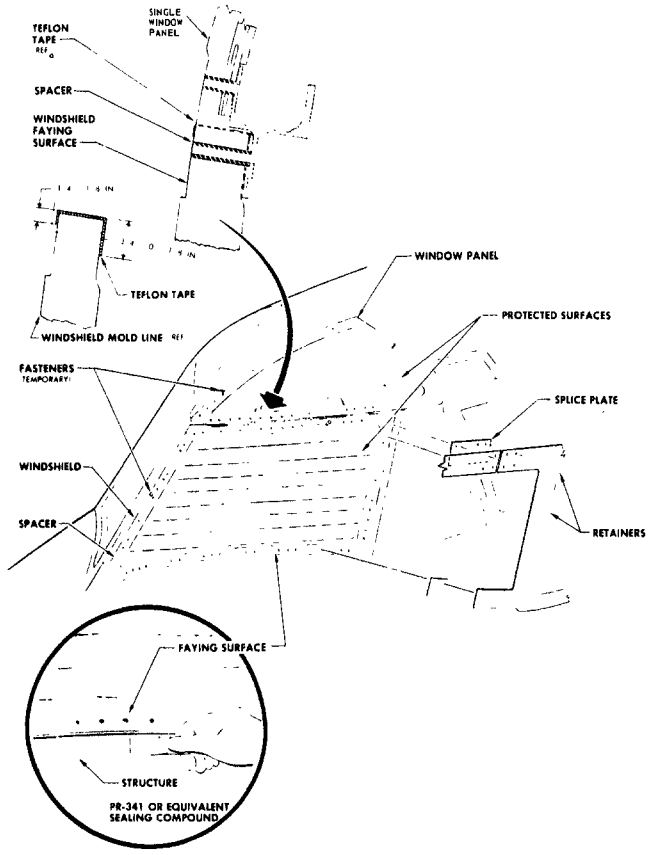
CAUTION

A resistance check of the replacement window panel heater element shall be performed to determine which transformer tap is to be utilized. If the proper transformer tap is not used, the replacement window panel could be damaged. (See Post Installation and Test Adjustments, chapter 30.)

NOTES

- ***The following installation procedure is typical for cockpit windows or windshield assemblies, with the exception of the pilot's sliding window. The cockpit windows that are two-part assemblies should be mated before installation as follows:***
- ***Mask the exposed surface of inner and outer panel to prevent marring or scratching.***
- ***Ensure that facing surfaces of panels are clean and that dehydrator tube is clear. Apply a smooth, even coat of PR1321B or equivalent sealant, about 1/32 in. thick, on faying surface of outer panel, and mate it with inner panel. Install spacers in panel assembly.***
- ***Spacers may be lightly tapped into windshield installation holes. If more force is required, enlarge holes with letter "H" (0.2656 in.) drill except as shown in detail A, page 201.***

1. Prior to installing the windshield and/or the eyebrow windows, the window retainers must be checked for proper fit prior to installation. (See Checking Clearance Bewteen the Windshield and Eyebrow Window Center Beam.)
2. Ensure that window frame and surrounding structure in affected area is clean and that no cracks exist.
3. On window frame place tabs of adhesive rubber tape (Specification MIL-T-6841 [1] or equivalent) 1/32-in. thick, cut in sections about 1/2 × 1 in. and spaced about 4 in. apart.
4. Apply sealant beads of PR-1223 or equivalent interior compound on window or windshield frame.
5. Place window panel or windshield in frame.



[1] May be procured from Nolan Paper Co., 6600 Valley St., Buena Park, CA 90620

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MAINTENANCE PRACTICES (Continued)

INSTALLING FIXED WINDOWS (Cont'd)

6. Apply adhesive Teflon tape (Mystik tape, No. 7503 or equivalent) around periphery of windshield. The tape may be cut and lapped at corners to fit (maximum two thicknesses at lap joints).

NOTE

Make sure that panel or windshield surfaces are protected, to prevent possible marring or scratching.

7. Install spacers in windshield.

NOTE

The spacers, when installed in windshield, will extend slightly beyond faying surface. This is to prevent uneven clamping pressures on the glass panels.

8. With a few fasteners, temporarily secure panel or windshield in place.
9. Spread a thin film of petrolatum (Federal Specification VV-P-236) on contacting surface of retainers and splice plates.
10. On windshield frame, place tabs of adhesive rubber tape (Specification MIL-T-6841) 1/16 inch thick, cut in sections about 1/2 x 1 inch, spaced about 4 inches apart.
11. Using a pressure applicator or clean wooden spatula, apply PR-341 or equivalent sealing compound on structure and faying surfaces. With spatula, spread sealant evenly on both surfaces.
12. Using fasteners indicated, install retainers and necessary splice plates. Torque 7S39-4A21 and 7S39-4A13 screws to 20-30 inch-pounds.
13. Apply masking tape and seal the following areas with PR-341 or equivalent compound:
    A. Gap between retainers and window or windshield.
    B. Skin gap between retainers and structure.

NOTE

Remove tape before sealing compound begins to set.

14. Within 4 hours after application, using Toluol (Federal Specification TT-T-548), clean excessive compound from equipment and areas to which it was applied.

CAUTION

Do not allow Toluol to come in contact with windshield or cockpit windows. The effects of Toluol, when introduced to the acrylic panels, will cause them to crack and craze.

WARNING

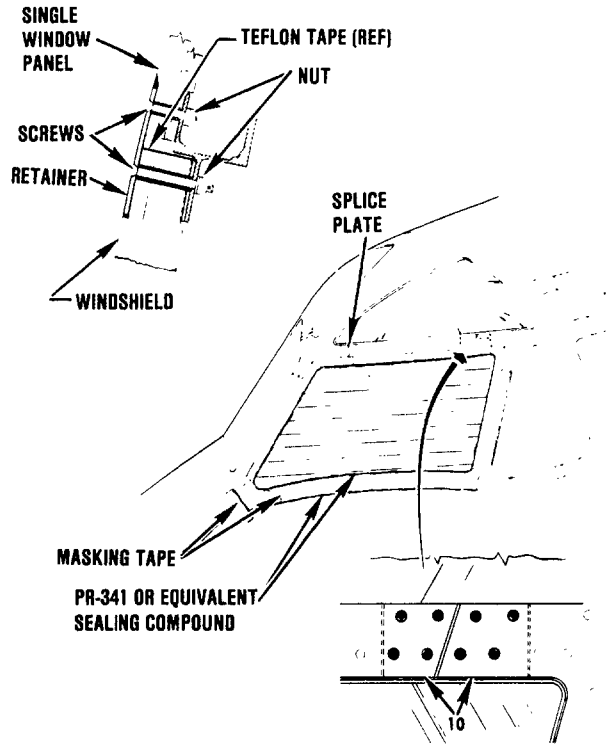
Toluol is highly flammable and toxic. Safety precautions to prevent fire must be observed while using this material. If used in a closed area, forced ventilation must be provided for personnel.

15. Apply pressure sealant in areas specified and perform cabin pressure test. (Refer to chapter 21.)
16. Refer to Postinstallation and Test Adjustments in chapter 30.

CAUTION

Resistance of the heater element must be made to determine which transformer tap is to be utilized. If the proper transformer tap is not connected, the new windshield may be damaged.

17. Perform operational checkout of windshield anti-ice system. (Refer to chapter 30.)
18. Reinstall dehydrator tube connections on double windows. (See 56-00-01.)
19. Install cockpit reveal.



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MAINTENANCE PRACTICES (Continued)

REMOVING PILOT'S SLIDING WINDOW

(See Figure 56-10-201 for identification of parts)

1. Remove reveal to gain access to upper track attach bolts.
2. Fully open pilot's sliding window.
3. Remove truck retaining screw from crank.
4. Remove upper track attaching bolts and lift track to separate truck from crank.
5. Remove screw from end cap on track, pull off cap, and slide out truck assembly.
6. Remove all lower cylinder retaining screws. Remove adapter cylinder retaining screws. Remove adapter cylinder retaining bolt from bellcrank.
7. Grasp handle, torque tube firmly, and lift window assembly from cylinders.

NOTE

Bellcrank should be lifted from cylinders simultaneously with window assembly to prevent the sudden release of torque tube spring tension.

8. Remove lower track trim and bulkhead panel aft of pilot's seat.
9. Remove track attaching bolts; lift out track.
10. With lower track removed, extract pin and slide out stop. Move cylinder assemblies out of track.

INSTALLING AND ADJUSTING PILOT'S SLIDING WINDOW

1. Assemble cylinders on lower track. Install stop on track and secure in place with NAS561P4-18 pin.
2. Install lower track, using same parts as were removed from each track attach point.
3. Locate cylinders in track relative to forward support, adapter, and bellcrank positions when window assembly is fully open.
4. Mate window assembly with cylinders, and install cylinder retaining screws. Install adapter cylinder retaining bolt in bellcrank.

NOTE

A slight preloading of the torque tube will be necessary when mating the adapter and bellcrank to the cylinders.

5. Slide truck on upper track. Install end cap on track, using NAS221-19 screw and NAS1291-3 nut.
6. Slide truck into crank, and install truck retaining screw.
7. Install upper track, using same parts as were removed from each track attach point.
8. Slide window to closed position.
9. Rotate eccentric adjustment on lower cylinders and truck, to position rollers in track. Extend roller at adapter to its full forward limit. Extend roller at forward support to maximum aft position. Adjust truck to position where truck rollers are centered within upper track.
10. Tighten cylinder and truck retaining screw, and bolt at bellcrank adapter.
11. Adjust and secure upper and lower track by use of serrated plates and shims to match position of rollers on window assembly throughout its full range of travel.

NOTE

Make sure truck rollers remain centered within track during upper track adjustment.

12. Loosen and rotate cylinder assemblies to make final adjustment of window assembly, relative to structure.

NOTE

Fore-and-aft adjustment can be made by rotating adapter cylinder. For inboard and outboard adjustments, rotate eccentric in bellcrank, forward support, and crank.

13. Torque cylinder and truck retaining screws, and bolt at adapter to tolerances indicated in Figure 56-10-201.
14. Install bulkhead panel, lower track trim, and cockpit reveal.

INSTALLATION OF PILOT'S SLIDING WINDOW SEAL AND APPLYING SEALING COMPOUND TO PILOT'S SLIDING WINDOW

To replace the sliding window seal only, perform steps 1, 2 and 4 thru 9. If both the seal and sealing compound around sliding window are to be replaced, perform all steps of the following procedure.

1. Remove old window seal.
2. Using an oil-free clean cloth, clean inner surface of windshield retainer with naptha (MIL-N-15178). Do not remove paint.
3. If the sealing compound around the sliding window is to be replaced, accomplish the following steps prior to installing seal:
   A. Remove existing sealing compound from sliding window.

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MAINTENANCE PRACTICES (Continued)

INSTALLATION OF PILOT'S SLIDING WINDOW SEAL AND APPLYING SEALING COMPOUND TO PILOT'S SLIDING WINDOW (Cont'd)

B. After removing the sealing compound, check the adhesion of the metal primer. If found to be loose, remove metal primer.

C. Using an oil-free clean cloth, thoroughly clean area where sealing compound is to be applied (see Figure 56-10-201).

NOTE

Petrolatum (VV-P-236 or equivalent) may be applied over any surface where adhesion of the sealing compound is not desired.

D. Apply a single layer of 3/4" wide Polyhem #105 (or equivalent) double coated tape on the inside perimeter of the windshield retainer.

NOTES

- ***To prevent an irregular surface, ensure that the tape splices do not overlap.***
- ***The Polyhem #105 double coated tape is .013 inches thick and can be substituted only by a double-coated tape having this exact thickness.***

E. Apply a .063 inch (3/4 inch wide) synthetic rubber shim over the double coated tape.

F. Apply a thin layer of petrolatum to synthetic rubber shim.

G. Apply sealing compound to sliding window. Ensure that a sufficient amount of sealant is applied and that it is spread evenly.

NOTE

Sealing compound PR-341, PS-247 or Pro-seal 895 may be used.

H. Close pilot's sliding window and allow to set for a minimum of 24 hours.

I. After the setting period, open sliding window and trim excess sealing compound as required.

J. Remove the synthetic rubber shim and tape from windshield retainers.

K. Repeat step 2.

3. Rough the bonded surface of seal with a wire brush or sandpaper, then remove all loose particles or contaminants, by using an oil-free clean cloth moistened with naptha.
4. Apply a coat of EC 1357 adhesive (Minnesota Mining Co.) to the seal and inner surface of windshield retainer.
5. Allow first coat of adhesive to set for 60 minutes, then apply a second coat to each surface.
6. When adhesive becomes tacky, install seal. See Figure 56-10-201 for proper seal to windshield retainer edge distance.

NOTE

The butt-splice should be located at the top of the windshield retainer. Only one butt-splice is allowed.

7. Clean adhesive from area as required.
8. Perform cabin pressure test. (Refer to chapter 21.)

INSPECTION OF THE COCKPIT ENCLOSURE CENTER SUPPORT BEAM

1. Remove all electrical power from airplane.
2. To gain access to the beams, proceed as follows:
   A. Loosen fasteners securing electrical control panel to structure, then lower panel assembly (See Figure 56-10-202).
   B. Loosen fasteners securing fire extinguishing control panel to structure, then remove or lower fire extinguishing control panel.
3. Remove insulation material from area to be inspected (see Figure 56-10-202).
4. Visually inspect entire length of each center support beam mold line flange for cracks. See Figure 56-10-202 for inspection area.

NOTE

If a crack is detected in either flange, rework will be required in accordance with Sabreliner Structural Repair Manual (NA-66-1032). Contact Sabreliner Corporation for further repair information.

5. Install the electrical and fire extinguishing control panels.
6. If any control panel electrical plug is disconnected, perform an operational checkout of the applicable system.

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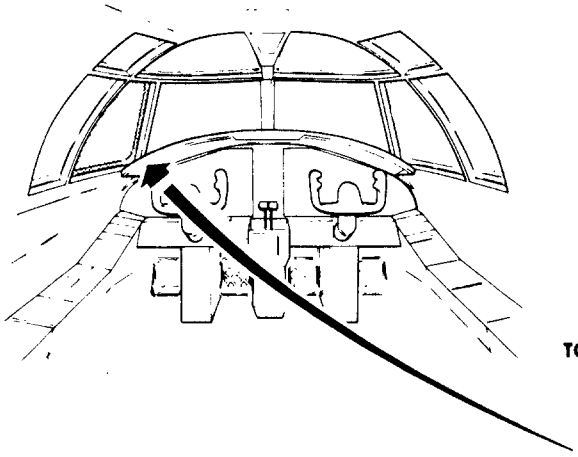
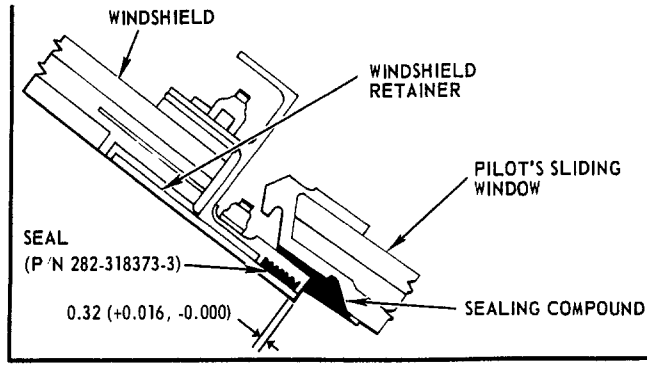
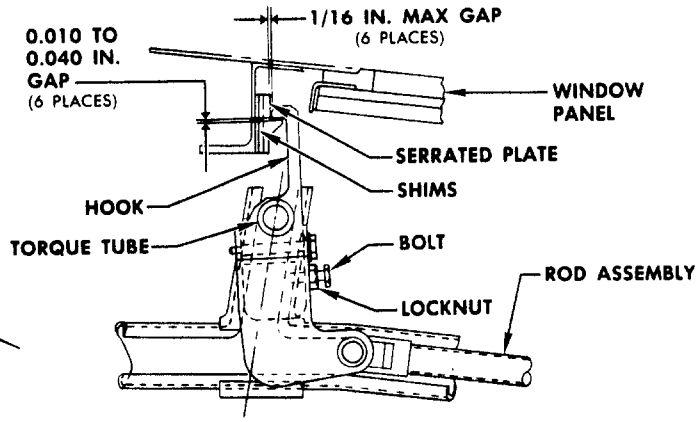
NOTE

Window panels cannot be replaced at repair stations. Special tooling is required to hold window assembly attach fittings in proper relationship with the track assembly.







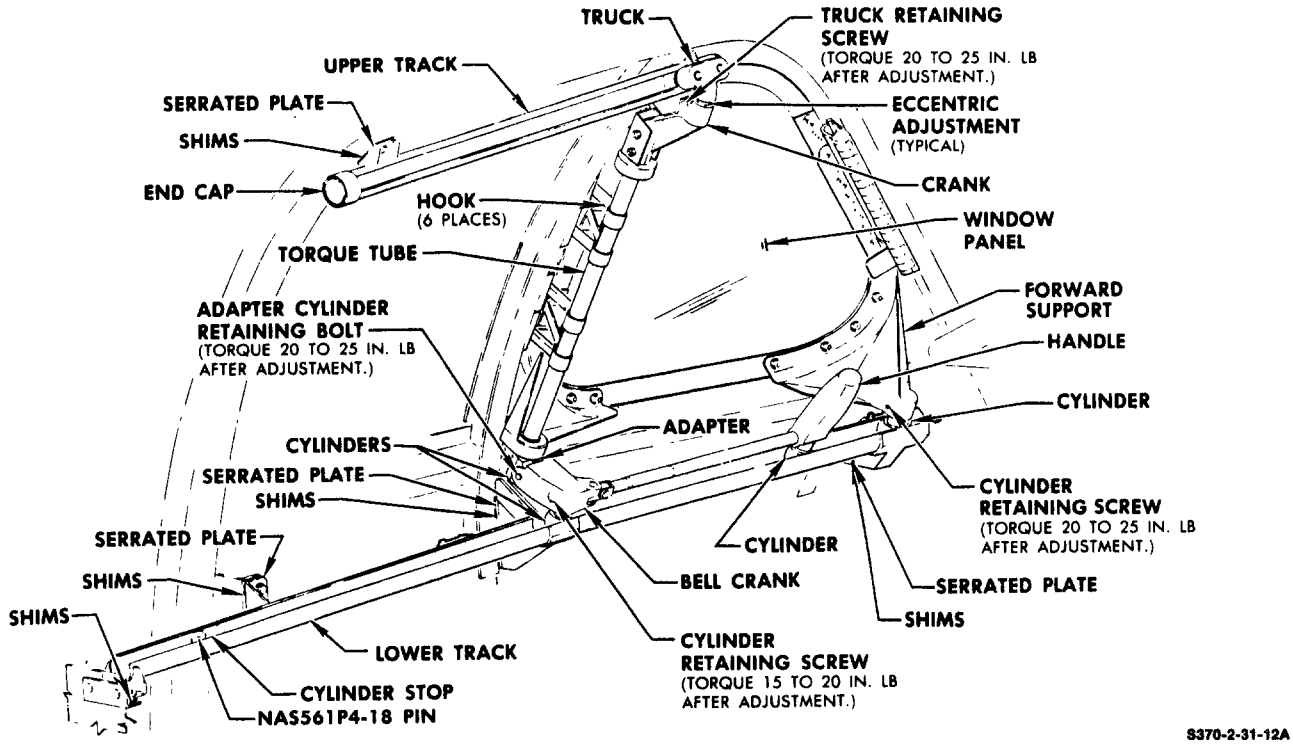


Figure 56-10-201 — Pilot's Sliding Window Installation

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CHECKING CLEARANCE BETWEEN THE WINDSHIELD AND EYEBROW WINDOW CENTER BEAM FLANGES AND THE WINDOW RETAINERS — EYEBROW WINDOWS AND WINDSHIELDS REMOVED

This procedure consists of checking the cockpit enclosure window retainers for proper installation. Subsequent to performing this procedure, the inspection requirement as depicted in chapter 5 for inspecting the center beam flanges for cracks is no longer required. Subsequent to any applicable repairs on the windshield center beam assembly and prior to installation of the cockpit enclosure windows, perform a window retainer fit check as follows:

1. Position the eyebrow and center windshield retainer on the center beam assembly.
2. Remove a sufficient number of spacers from the eyebrow and windshield panels, then install spacers and screws at every sixth hole along the outer edge of both retainers.
3. Ensure that all required screws are installed through all spacers prior to tightening any screw.

CAUTION

Do not install the inboard row of screws (short screws) in either retainer. This will prevent the center beam flange from being drawn up to the under side of the retainer preventing a false clearance measurement.

4. Tighten the screws in the lower retainer. Install washers and nuts on the screws of the upper retainer. Tighten nuts.
5. Using a .003 in. feeler gauge from the opening that the windows occupy, measure the clearance between the center windshield beam flange and the retainers (see Figure 56-10-203).
6. If the clearance exceeds .003 inch at any point of the flange(s), refer to Shimming Cockpit Enclosure Window Retainers — Windows Removed in this chapter. (See Table of Contents.)

CHECKING CLEARANCE BETWEEN THE EYEBROW WINDOW AND WINDSHIELD CENTER BEAM FLANGES AND RETAINERS — EYEBROW WINDOWS AND WINDSHIELDS INSTALLED

This procedure consists of checking the eyebrow window and windshield retainers for proper installation and may be performed subsequent to the visual inspection of the center beam flanges (reference S/B No. 1).

1. Remove all electrical power from airplane.
2. Cover the windshield and eyebrow windows with a suitable covering as to protect from scratches and/or window breakage.
3. Remove electrical control panel assembly from cockpit overhead to gain access to retainer nuts (see Figure 56-10-203).
4. Remove fire extinguisher control panel assembly.
5. Peel back insulation from over center beam flanges.
6. To check clearance between the eyebrow window and retainer, proceed as follows:
   A. Remove the inboard row (both inboard rows) of screws from the retainer (see Figure 56-10-203).

NOTE

Two men are required to measure the clearance between the retainer and flange. One man will be on the outside and a man in the cockpit to hold a metal plate against the bottom part of the flange of the hole being measured. This prevents the depth gauge from protruding below the flange.

   B. With a man in the cockpit holding a flat metal plate against the flange of both inboard rows of screw holes and using a depth gauge calibrated in thousands (.000), measure the depth of each screw hole in both inboard rows of screws.

NOTE

As each hole is measured, the clearance may be determined by subtracting the thickness of the retainer and flange. The retainer is .080 inch thick and the flange is .060 inch; therefore, if a total depth is measured at .140 inch, there is no clearance between the retainer and flange. However, if a total depth of .200 inch is measured, then a clearance of .060 inch exists between the flange and retainer.

   C. Mark retainer with a suitable marking pen on any areas that are over .143 inch. Record all clearances on a separate sheet of paper.
7. If the clearance exceeds .143 inch (includes the .003 inch allowable gap), refer to Shimming Cockpit Enclosure Window Retainers — Eyebrow Windows and Windshields Installed.
8. To check clearance between center beam flanges and windshield retainer, proceed as follows:
   A. Remove windshield center retainer assembly.
   B. Remove sealant from both inboard flanges of the center beam.
   C. Lay a single layer of Saran Wrap [1] (or equivalent) over the entire length of both inboard flanges (across 20 screw holes on each side).

[1] Locally Procured

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CHECKING CLEARANCE BETWEEN THE EYEBROW WINDOW AND WINDSHIELD CENTER BEAM FLANGES AND RETAINERS — EYEBROW WINDOWS AND WINDSHIELDS INSTALLED (Cont'd)

D. Using pliable clay or a fast setting sealant, apply two 3/16" beads on each side of the screw holes of both flanges.

E. Lay a single layer of Saran Wrap over the beads. This will prevent the sealant or clay from sticking to the retainer and/or the center beam flanges.

F. Install the center beam retainer using a screw in every fourth hole in the outer row of screws *only*. Install all screws prior to securing any screw. Tighten screws alternating from side to side starting at the bottom of the retainer.

NOTE

If a sealant is used for the clearance check, allow a sufficient curing time for sealant to properly set.

G. Using a marking pen, mark the Saran Wrap at each hole location (20 holes on each side).

H. Remove windshield retainer. Carefully remove the two sheets of Saran Wrap with clay (or sealant) filler and lay on a flat surface. Do not remove Saran Wrap from filler

I. Mount a dial indicator to the flat surface and zero the dial (with the plunger against the flat surface).

NOTE

Use a round tip on the dial indicator so the tip will ride over the exterior surface of the Saran Wrap.

J. Raise the dial indicator plunger and slide the Saran Wrapped filler under the plunger.

K. As the Saran Wrapped filler is slowly pulled under the dial indicator, record thickness between each screw hole.

L. Repeat steps J and K using the opposite flange Saran Wrapped filler.

M. Using the recorded measurements as in step K, determine the gap that exists between the center windshield beam retainer and flanges. If clearance is over .003 inch, refer to Shimming Cockpit Enclosure Window Retainers — Eyebrow Windows and Windshields Installed.

[1] May be procured from Nolan Paper Co., 6600 Valley St., Buena Park, CA 90620.

SHIMMING COCKPIT ENCLOSURE WINDOW RETAINERS — EYEBROW WINDOWS AND WINDSHIELDS REMOVED

Any clearance in excess of .003 inch between the window retainers and the flanges of the center beams requires the use of a metal shim fabricated from 7075-T6 bare Al Sheet or a liquid shimming material [1]. The liquid shim material *cannot* be used with clearance over .060 inch; therefore, any gaps over .060 inch must be shimmed to a gap of less than .060 inch using 7075-T6 bare Al Sheet stock of the proper gauge and then the shim riveted to the beam flanges where applicable. To shim window retainers, proceed as follows:

1. With the retainers installed and secured as per step 7 of the previous procedure, obtain a .060 inch feeler gauge and measure the clearance between the center windshield beam flange and the retainers (see Figure 56-10-203 for areas to measure).
2. As each flange is being measured, if any part of the flange has a gap of more than .060 inch, mark each area with a suitable marking pen.
3. With all flanges marked as to the areas that exceed .060 inch gap, determine (measure with feeler gauge) and record the maximum gap at each marked location.
4. Determine the thickness of the 7075-T6 material to be used to shim the gap in the marked areas to under .060 inch clearance. Use the figures recorded in step 3. For example: if an area for a distance of six screw holes measures .097 inch, then a shim of 7075-T6 material of .070 inch gauge should be used.

NOTE

If the measured flange has a uniform gap across its total length, a strip of 7075-T6 material of the correct gauge may be used to shim the gap to no greater than the .003 inch allowable clearance.

5. Remove retainers and clean the beam flanges (both sides) and apply a coat of polyamide epoxy primer.
6. Fabricate the required length and thickness of 7075-T6 shim material. After the material is cut to length, feather both ends of the shim. Clean shim and apply a coat of polyamide epoxy primer to all surfaces.
7. Using a minimum thickness (5 to 10 mils) of liquid shim material [1], apply it to the flange(s) in the area that the shim is to be installed, then position eyebrow window shim(s) over marked area and secure with rivets, [flush pull rivets (3/32") are acceptable]. Use only the required quantity of rivets to retain the shim in position. Rivets are installed between existing screw holes. Countersink rivet holes to maintain a flat outer shim surface.

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MAINTENANCE PRACTICES (Continued)

SHIMMING COCKPIT ENCLOSURE WINDOW RETAINERS — EYEBROW WINDOWS AND WINDSHIELDS REMOVED (Cont'd)

8. Locate and drill (No. 11 drill) holes through shim(s) to pick up existing eyebrow window flange holes.
9. Mark lower retainer shims to pick up existing flange holes prior to riveting shim(s) to flanges. Remove shim(s) on lower retainer and drill (No. 11 drill) holes in shim(s). Install shim(s) on lower retainer flanges as per step 7.
10. To apply liquid shim material to eyebrow window retainer center beam flanges, proceed as follows:

NOTE

Pot life of an 8 oz. tube of liquid shim material [1] (cartridge type) is twenty minutes from the start of the mixing process. After the plunger of the cartridge is depressed to break the diaphragm separator within the tube, a minimum of 60 complete strokes shall ***be completed to properly mix the shim material. After mixing, the shim material may be applied with a Semco 250 (or equivalent) sealant gun.***

A. Clean flanges of center beam thoroughly and ensure that flanges are properly primed (polyamide epoxy primer).

B. Clean and apply a release agent [2] (vaseline may also be used) to the under side of the retainer on an area of one inch on each side to the inboard bolt holes (where the flanges come in contact with the retainer) and to the screw countersunk areas of the inboard rows of screws.

C. Get all hardware and tools to the area prior to mixing the shim material.

D. Apply a strip of tape to the underside of the flanges along its entire length. This prevents the shim material from squeezing out the flange holes.

E. Mix the shimming material and apply 2 beads, (3 beads in the area of .020 inch gap and over), (3/16" by 3/16") of shim material to both flanges.

F. Install retainer using the outer row of screws and applicable spacers in every fourth hole. Install screws and spacers prior to securing any screw. Tighten screws alternating from side to side starting at the forward end of the retainer.

G. After the retainer is secured, the liquid shim material will squeeze out the retainer holes on the top side. Wipe the excess material from the holes and countersinks. A countersink tool (100°) may be used by *hand* to clean the shim material from the countersunk screw holes. Also wipe the excess material from the under side of the retainer on both sides of the flanges.

CAUTION

To prevent the shimming material from lifting when the retainer is removed, ensure that the inboard row screw holes (top of retainer only) are clear of the excess shimming material. The flange holes will be drilled clear after the material has hardened.

H. Allow a minimum of 4 hours for the shimming material to cure before removing the retainer.

I. After the cure period, remove all screws from retainer, then starting at the aft edge of the retainer, slowly and carefully lift the retainer.

J. Locate existing flange holes and drill (No. 11 drill) thru the shimming material. Holes may be drilled from the bottom if desired. Use light pressure on drill.

K. Clean all excess shimming material from area adjacent to the flanges. A small hand-held grinding tool may be used.

CAUTION

If a grinding tool is used, ensure that no surrounding structure is touched with the grinding wheel.

L. Remove release agent from retainer and countersunk screw holes.

11. To apply liquid shim material to the lower retainer, proceed as follows:

A. Install lengths of 3/16" diameter silicone rubber (or equivalent) as plugs in both inboard rows of nut plates. Ensure that plugs are inserted to a depth as not to allow the shimming material to enter the nut plates.

NOTE

The plugs should be installed far enough in the nut plate as to retain them in position and to ensure that the liquid shimming material does not enter the nut plate. The plugs should be pointed on the end that protrudes through the retainer and only long enough to protrude through the retainer. This facilitates the installation of the retainer and ensures that the plugs do not flatten out under the retainer as it is secured in place.

[1] Hysol Liquid Shim #734, Hysol Division of Dexter Corp., 15051 Don Julian Road, Industry, CA 91744
[2] Frekote 33, Frekote Inc., Santa Ana, CA

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MAINTENANCE PRACTICES (Continued)

SHIMMING COCKPIT ENCLOSURE WINDOW RETAINERS — EYEBROW WINDOWS AND WINDSHIELDS REMOVED (Cont'd)

B. Apply release agent to under side of retainer where flange contacts retainer.

C. Apply two beads (3/16" by 3/16") of liquid shim material to each flange.

D. Position retainer over free ends of rubber plugs, then lower retainer against flanges and secure with a minimum of four screws (with spacers installed) on each outboard side of the retainer. Tighten screws.

E. Allow a minimum of 4 hours cure time, then remove retainer and rubber plugs. Clean excess shim material from surrounding areas as per step 10 K.

F. Remove release agent from under side of retainer.

G. Install window retainers as per existing instructions. Refer to Installing Cockpit Fixed Windows and Windshields in this chapter.

SHIMMING COCKPIT ENCLOSURE WINDOW RETAINERS — EYEBROW WINDOWS AND WINDSHIELDS INSTALLED

Any clearance in excess of .003 inch between the eyebrow window and windshield retainer and the flanges of the center beams requires the use of a metal shim fabricated from 7075-T6 bare Al Sheet or a liquid shimming material. The liquid shim material *cannot* be used with clearances over .060 inch; therefore, any gaps over .060 inch must be shimmed to a gap of less than .060 inch using 7075-T6 bare Al Sheet stock of the proper gauge, and then the shim riveted to the beam flanges where applicable. If this procedure is performed, the inspection requirements as depicted in chapter 5 for inspecting the center beam flanges for cracks is no longer required. To shim the eyebrow window retainer, proceed as follows:

1. Remove eyebrow window retainer.
2. With both flanges (upper and lower flanges) marked as to the areas that exceed .060 inch gap, determine the thickness of the 7075-T6 material to be used to shim the gap in the marked areas to under .060 inch clearance. Using the eyebrow window retainer as an example: if an area for a distance of six screw holes measures .213 inch, then a shim of 7075-T6 material of .070 inch (.213 minus .143) gauge should be used.

NOTE

If the measured flange has a uniform gap across its total length, a strip of 7075-T6 material of the correct gauge may be used to shim the gap to under the .003 inch gap.

3. Remove both retainers and clean the beam flanges (both sides) and apply a coat of polyamide epoxy primer.
4. Fabricate the required length and thickness of 7075-T6 shim material. After the material is cut to length, feather both ends of the shim. Clean shim and apply a coat of polyamide epoxy primer to both sides.

CAUTION

Use extreme care when riveting shim into position, to prevent possible breakage of eyebrow windows.

5. Using a minimum thickness (5 to 10 mils) of liquid shim material, apply it to the flange(s) in the area that the shim is to be installed, then position eyebrow window retainer flange shim(s) over marked area and secure with rivets [flush pull rivets (3/32") are acceptable]. Use only the required quantity of rivets to retain the shim in position. Rivets are installed between existing screw holes. Countersink shim rivet holes to maintain a flat outer shim surface.
6. Locate and drill (No. 11 drill) holes through shim(s) to pick up existing eyebrow window flange holes.
7. Mark lower retainer shims to pick up existing flange holes prior to riveting shim(s) to flanges. Remove shim(s) and drill (No. 11 drill) holes in shim(s). Install shim(s) on lower retainer flanges as per step 5.
8. To apply liquid shim material to eyebrow window retainer center beam flanges, proceed as follows:

NOTE

Pot life of an 8 oz. tube of liquid shim material [1] (cartridge type) is twenty minutes from the start of the mixing process. After the plunger of the cartridge is depressed to break the diaphragm separator within the tube, a minimum of 60 complete strokes shall be completed to properly mix the shim material. After mixing, the shim material may be applied with a Semco 250 (or equivalent) sealant gun.

A. Clean flanges of center beam thoroughly and ensure that flanges are properly primed (polyamide epoxy primer).

B. Clean and apply a release agent [2] (vaseline may also be used) to the under side of the retainer on an area of one inch on each side to the inboard bolt holes (where the flanges come in contact with the retainer) and to the screw countersunk areas of the inboard rows of screws.

[1] Hysol Liquid Shim #734, Hysol Division of Dexter Corp., 15051 Don Julian Road, Industry, CA 91744
[2] Frekote 33, Frekote Inc., Santa Ana, CA

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SHIMMING COCKPIT ENCLOSURE WINDOW RETAINERS — EYEBROW WINDOWS AND WINDSHIELDS INSTALLED (Cont'd)

C. Get all hardware and tools to the area prior to mixing the shim material.

D. Apply a strip of tape to the under side of the flanges along its entire length. This prevents the shim material from squeezing out the flange holes.

E. Mix the shimming material and apply 2 beads (3 beads in the area of .020 inch gap and over) (3/16" by 3/16") of shim material to both flanges.

F. Install retainer using a screw in every fourth hole in the outer row of screws only. Install all screws prior to securing any screw. Tighten screws alternating from side to side starting at the forward end of the retainer.

G. After the retainer is secured, the liquid shim material will squeeze out the retainer holes on the top side. Wipe the excess material from the holes and screw countersunk holes. A countersink tool (100°) may be used by *hand* to clean the shim material from the countersunk screw holes. Also wipe the excess material from the under side of the retainer.

CAUTION

To prevent the shimming material from lifting when the retainer is removed, ensure that the inboard row screw holes (top of retainer only) are clear of the excess shimming material. The flange holes will be drilled clean after the material has hardened.

H. Allow a minimum of 4 hours for the shimming material to cure before removing the retainer.

I. After the cure period, remove all screws from retainer then, starting at the aft edge of the retainer, slowly and carefully lift the retainer.

J. Locate existing flange holes and drill (No. 11 drill) thru the shimming material. Holes may be drilled from the bottom if desired. Use light pressure on drill.

K. Clean all excess shimming material from area adjacent to the flanges. A small hand-held grinding tool may be used.

CAUTION

If a grinding tool is used, ensure that no surrounding structure is touched with the grinding wheel.

L. Remove release agent from retainer and countersunk screw holes.

9. To apply liquid shim material to the lower retainer, proceed as follows:

A. Install lengths of 3/16" diameter silicone rubber (or equivalent) as plugs in both inboard rows of nut plates. Ensure that plugs are inserted to a depth as not to allow the shimming material to enter the nut plates.

NOTE

The plugs should be installed far enough in the nut plates as to retain them in position and to ensure that the liquid shimming material does not enter the nut plate. The plugs should be pointed on the end that protrudes through the retainer and only long enough to protrude through the retainer. This facilitates the installation of the retainer and ensures that the plugs do not flatten out under the retainer as it is secured in place.

B. Apply release agent to under side of retainer where flange contacts retainer.

C. Apply two beads (3/16" by 3/16") of liquid shim material to each flange.

D. Position retainer over free ends of rubber plugs, then lower retainer against flanges and secure with a minimum of four screws (with spacers installed) on each outboard side of the retainer. Tighten screws.

E. Allow a minimum of 4 hours cure time, then remove retainer and rubber plugs. Clean excess shim material from surrounding areas as per step 8K.

F. Remove release agent from under side of retainer.

G. Install window retainers as per existing instructions. Refer to Installing Cockpit Fixed Windows and Windshields in this chapter (see Table of Contents).

10. If clearance is within limits, replace screws in eyebrow retainer and install windshield center retainer as per existing instructions. Refer to applicable steps in Installing Cockpit Fixed Windows and Windshields in this chapter (see Table of Contents).

11. Replace insulation, then install electrical and fire extinguisher control panels.

12. Remove covering from window.

13. If any electrical plugs or wires are removed from the control panels, perform an operational checkout of the affected system.

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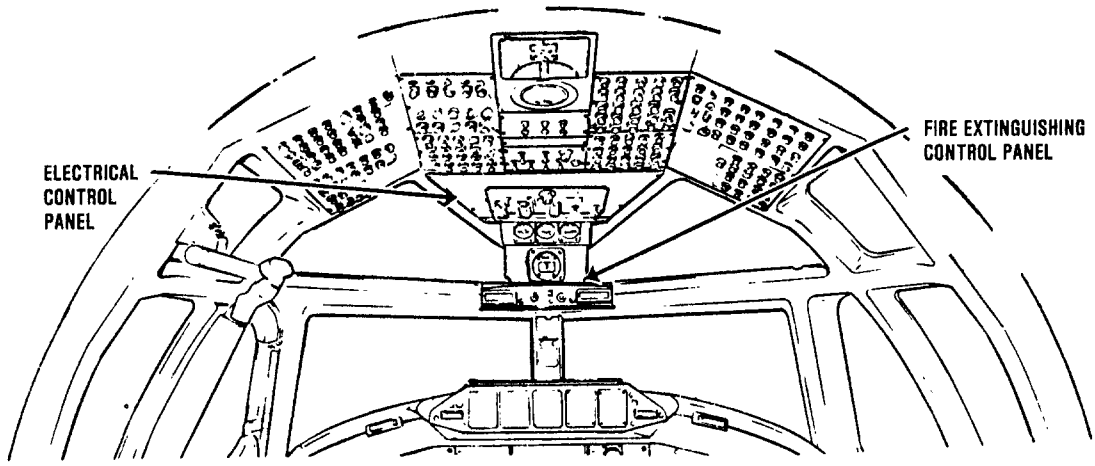

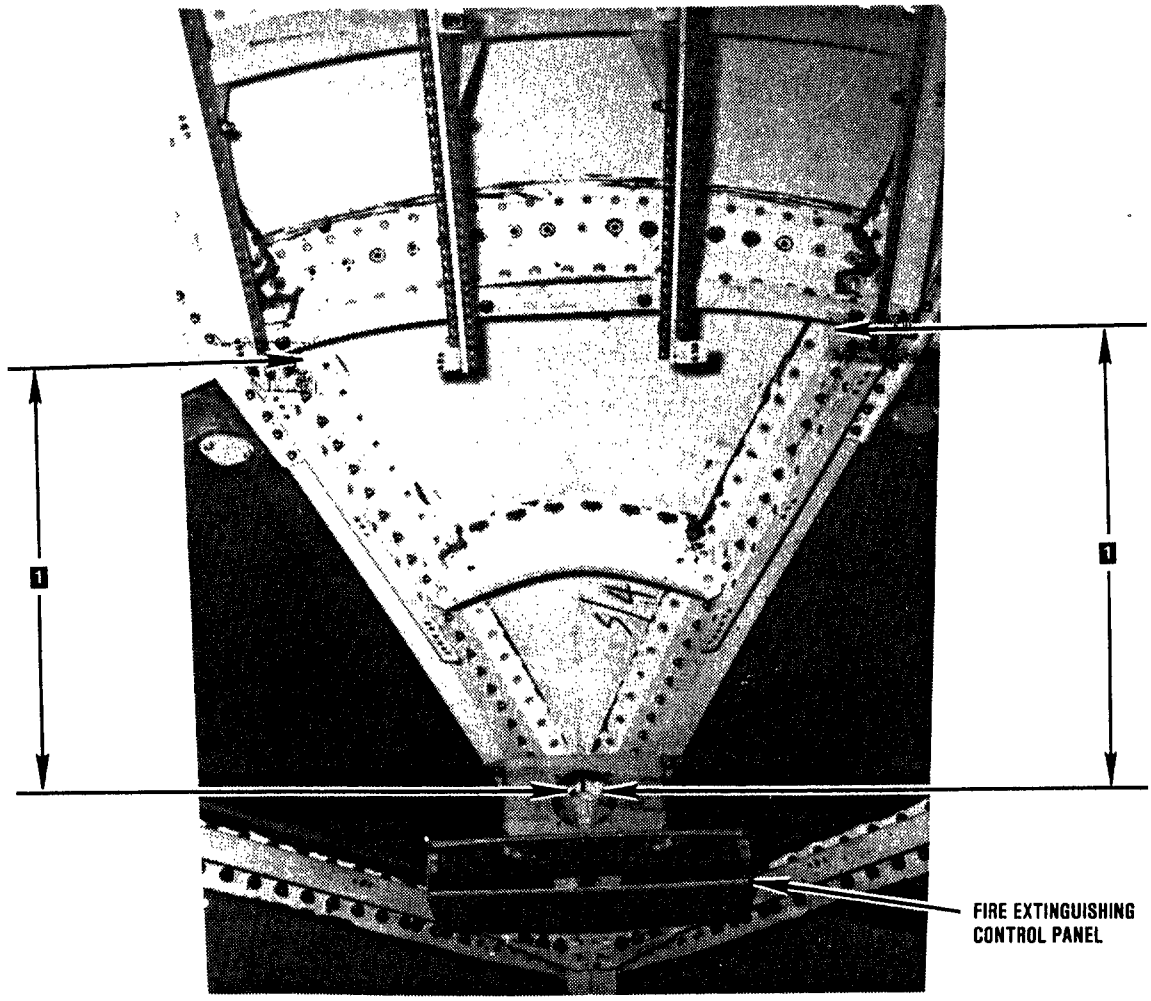

1 INSPECT ENTIRE LENGTH OF FLANGE FOR CRACKS

Figure 56-10-202 — Cockpit Enclosure Center Support Beams

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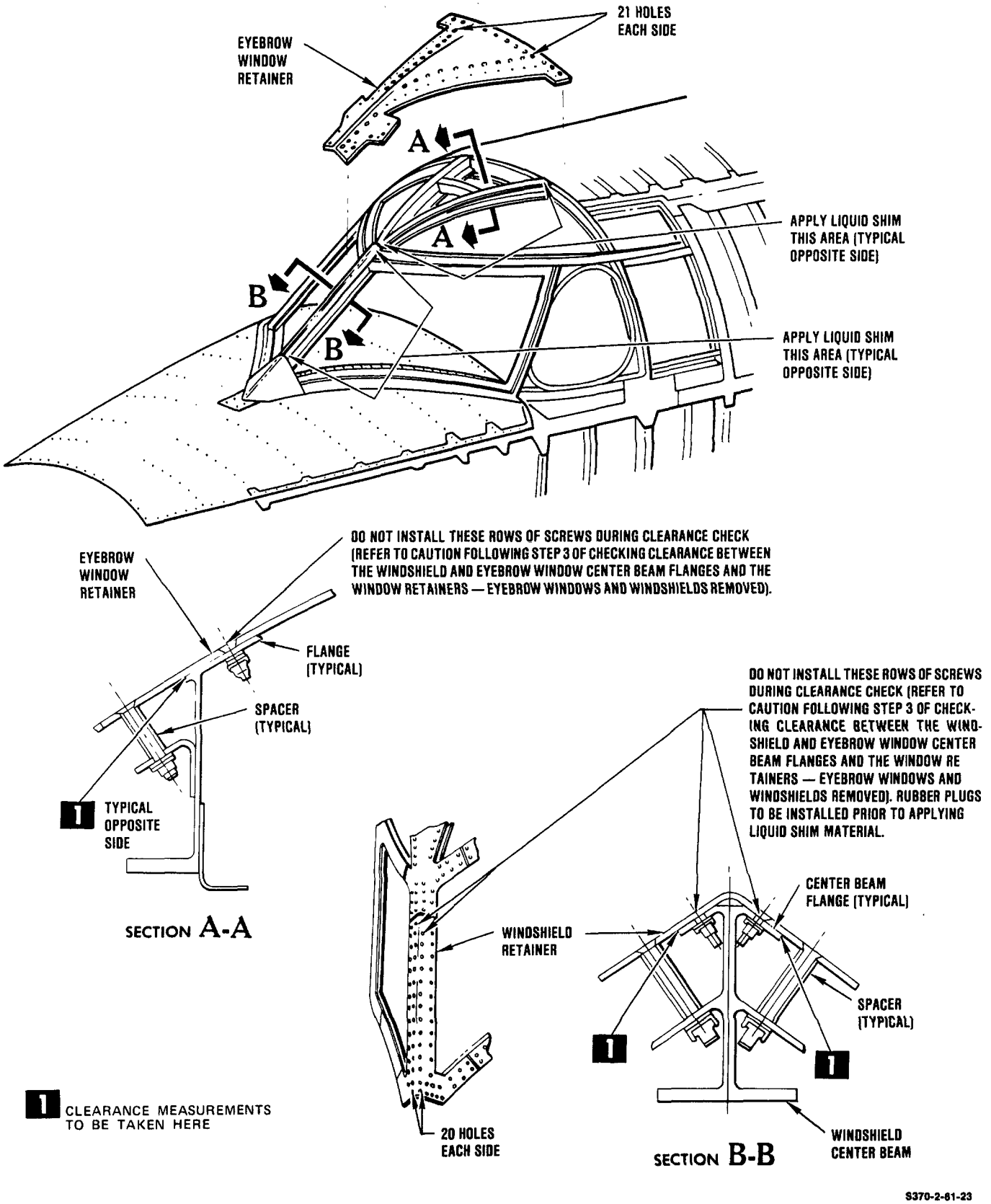
EYEBROW WINDOW RETAINER
21 HOLES EACH SIDE
A
A
B
B
APPLY LIQUID SHIM THIS AREA (TYPICAL OPPOSITE SIDE)
APPLY LIQUID SHIM THIS AREA (TYPICAL OPPOSITE SIDE)
DO NOT INSTALL THESE ROWS OF SCREWS DURING CLEARANCE CHECK (REFER TO CAUTION FOLLOWING STEP 3 OF CHECKING CLEARANCE BETWEEN THE WINDSHIELD AND EYEBROW WINDOW CENTER BEAM FLANGES AND THE WINDOW RETAINERS — EYEBROW WINDOWS AND WINDSHIELDS REMOVED).
EYEBROW WINDOW RETAINER
FLANGE (TYPICAL)
SPACER (TYPICAL)
1 TYPICAL OPPOSITE SIDE
SECTION A-A
DO NOT INSTALL THESE ROWS OF SCREWS DURING CLEARANCE CHECK (REFER TO CAUTION FOLLOWING STEP 3 OF CHECKING CLEARANCE BETWEEN THE WINDSHIELD AND EYEBROW WINDOW CENTER BEAM FLANGES AND THE WINDOW RETAINERS — EYEBROW WINDOWS AND WINDSHIELDS REMOVED). RUBBER PLUGS TO BE INSTALLED PRIOR TO APPLYING LIQUID SHIM MATERIAL.
CENTER BEAM FLANGE (TYPICAL)
WINDSHIELD RETAINER
SPACER (TYPICAL)
1
1
1 CLEARANCE MEASUREMENTS TO BE TAKEN HERE
20 HOLES EACH SIDE
WINDSHIELD CENTER BEAM
SECTION B-B
S370-2-61-23

Figure 50-10-203 — Window Retainer(s) to Center Beam Flanges Clearance Measurements

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CABIN

MAINTENANCE PRACTICES

REMOVING CABIN WINDOWS
(See Figure 56-20-201)

1. Remove interior trim as necessary to gain access to window panel.
2. Cut sealing compound away from support, dehydrator tube support clips, and outer panel.
3. Remove screw from support and panel assembly.
4. Lift out panel assembly and place on padded surface to prevent marking or scratching.
5. Remove retainer from inner panel assembly.

INSTALLING CABIN WINDOWS
(See Figure 56-20-201)

1. To prevent marring or scratching, mask outer surface of outer and inner panels.
2. Apply a thin coat of PR1321B sealant on faying surface of outer panel, and mate the two panels.
3. Install spacers through holes around edges of inner and outer panels. The spacer should be flush with the surrounding area. Use sealant to retain spacers if necessary.

NOTE

Ensure that optical surfaces of panels are clean and that dehydrator holes in panel are clear.

4. Fit retainer to inner panel and, to maintain hole alignment, insert two or three 7S39-3A12 screws through assembly.
5. Spread a thin coat of petrolatum (Federal Specification VV-P-236) on surface to which panel assembly is to be installed.
6. Apply a single bead of PR1223B2 sealant on edge of outer panel.
7. Position bottom of panel between dehydrator tube support clips and frame; then swing it into place.
8. To brace panel assembly in frame, temporarily install NAS1291-3 nuts on alignment screws.
9. Align panel attaching holes with structure, and fasten them in place, using required fasteners.
10. Remove alignment screws, and install panel assembly to structure.

NOTES

- ***Install all fasteners loosely before tightening, to prevent buckling of support and retainers.***
- ***Lock-O-Seal washers (600-015-10) are installed between retainer and LD153-0011-0011 washers.***

11. Mask outer panel, and seal gap between structure and panel, using PR1223B2 sealing compound.

NOTE

Remove masking tape before sealant "sets up."

12. Pressure-seal area around dehydrator tube support clips and inner panel. Perform cabin pressure test (refer to chapter 21).
13. Install cabin reveal and trim panel.

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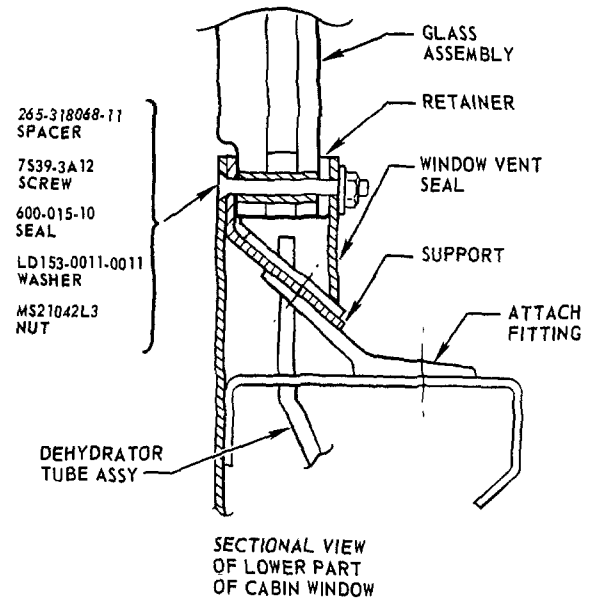

SECTIONAL VIEW OF LOWER PART OF CABIN WINDOW

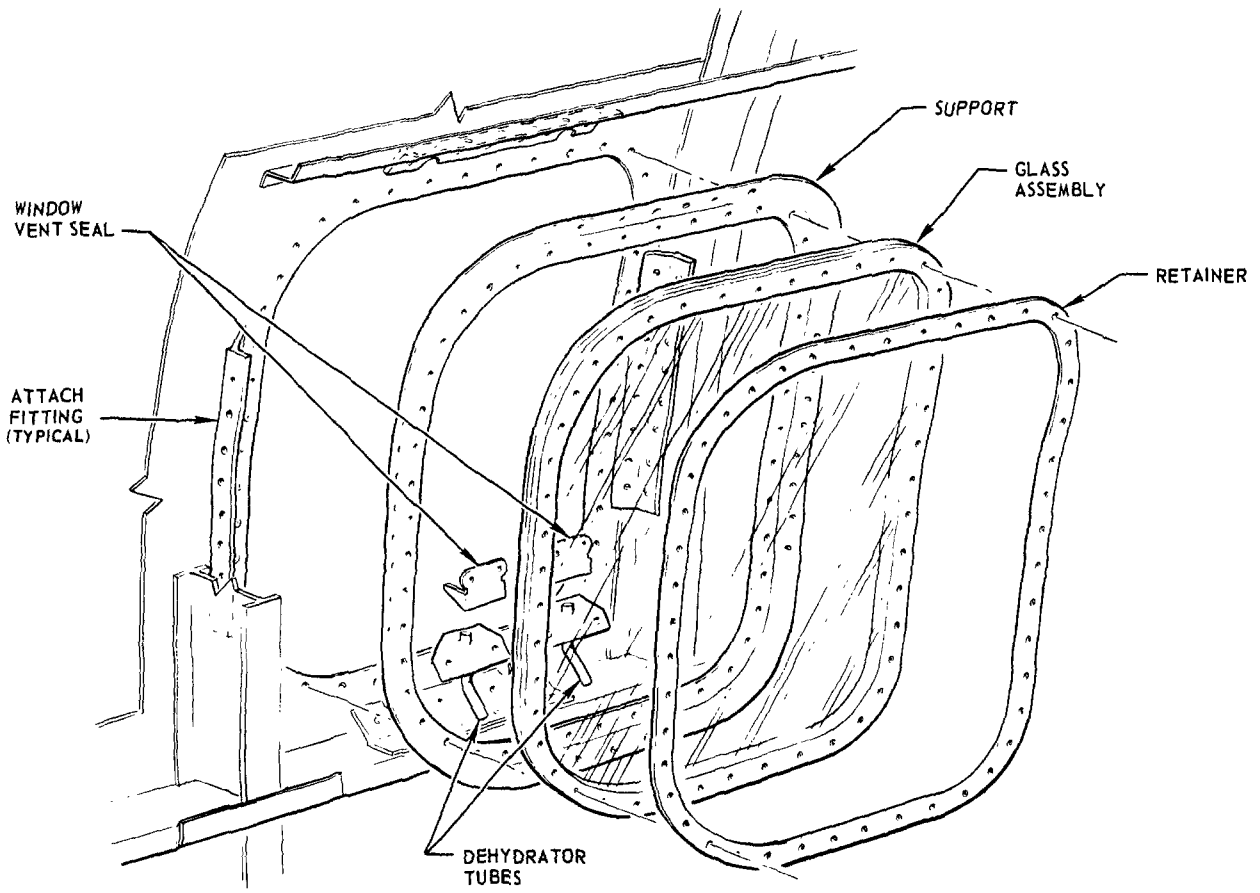

Figure 56-20-201 — Cabin Window Installation

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CHAPTER 57

WINGS

CHAPTER 57

WINGS

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CHAPTER 57

WINGS

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WINGS

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* The asterisk indicates pages revised, added, or deleted by the current revision.

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WINGS - DESCRIPTION

1. General

The wing of the Sabreliner is an all-metal, full-cantilever type consisting of two sections. The two wing sections are joined at a common centerline rib with splice joints on the top and bottom surfaces. A skid is attached to the bottom of the lower splice joint to protect the wing in case of a wheels-up landing.

Provisions are made for installation of inflatable deicer boots on the wing inboard leading edge, the lower surface of the wing leading edge, and the upper surface of the wing slats.

A set of five interconnected slats extend along each wing, and are designed to permit automatic retraction or extension as aerodynamic forces act upon them. Panels are provided in the wing leading edge for access to the slat tracks and deicer boot hoses.

Electrically operated flaps extend along the inboard trailing edge of the wing. The cable-operated ailerons extend along the outboard trailing edge. The trailing edges of the slats, flaps, ailerons, and wing inboard trailing edge are constructed of honeycomb sandwich panels.

Each wing contains a 451.5-gallon integral fuel cell. The front and rear spars, along with the wing skins and ribs, form four fuel bays. In addition, the wing leading edge inboard of the slats contains fuel. Five spring-loaded drains are built into each panel of the wing to permit draining the sump of each fuel bay. All controls and wiring are routed along the trailing edge of the wing.

A sheet-metal angle is installed on the upper inboard trailing edge of the right-hand wing. This angle acts as a scupper drain for upper wing surface water seepage, to prevent moisture from accumulating and freezing on the aileron control sectors.

A fairing on the lower center section of the wing contains the ADF loop and sense antennas.

MAIN FRAME - DESCRIPTION AND OPERATION

1. General

The wing main frame is of a box-type construction, using two spars and tapered machine-milled upper and lower skins. The two wing sections are joined at a common centerline rib with splice joints on the top and bottom surfaces. Each wing frame contains a 451.5-gallon integral fuel cell. The front and rear spars, along with the wing skins and ribs, form four fuel bays for the fuel cell.

CAUTION: REPAIRS OR MODIFICATIONS INVOLVING THE USE OF STEEL OR OTHER MAGNETIC MATERIALS (INCLUDING FASTENERS) ARE PROHIBITED WITHIN A 36-INCH RADIUS OF THE COMPASS SYSTEM FLUX GATES IN EACH WING TIP. NON-MAGNETIC MATERIALS ARE THE ONLY ACCEPTABLE MATERIALS THAT MAY BE ADDED IN THESE AREAS.

MAIN FRAME

MAINTENANCE PRACTICES

SEPARATING FUSELAGE AND WING
(See Figure 57-10-201)

CAUTION

On airplanes with teflon coated bearings, to prevent damage to the teflon coating, do not apply lubrication to the four wing attach bolts.

Equipment/Tools

- Hoisting Sling, P/N 306-810021-11.
- Jacks, 5-ton Tripod Type.
- Jacking and Mooring Kit.

Procedure:

1. Drain fuel and hydraulic systems, and disconnect batteries (refer to chapter 12).
2. Attach hoisting sling to fuselage (see Figure 57-10-201).
3. Remove wing trailing edge inboard sections.
4. Remove two access doors on bottom of fuselage inboard of wing leading edges (refer to chapter 6, Figure 6-00-3).
5. At wing trailing edge, disconnect and cap hydraulic, air, and fuel system lines. Disconnect electrical plugs, fuel quantity system connections, landing gear emergency release cable, aileron control rod, and flap interconnect cable.
6. Disconnect ADF loop and sense antenna coaxial cables (four) at fuselage floor connectors accessible through access hole in bottom of fuselage, just aft of right wheel well. Remove cables from supporting clamps attached to fuselage structure. Coil cables, and temporarily attach them to wing rear spar.

NOTE

The antennas are inside the fairing, attached to the underside of the wing center section.

7. Remove entire wheel well detachable fairing structure extending from wing rear spar to aft end of wheel wells, by removing attaching bolts, rivets, and screws.
8. Disconnect bonding jumpers in left and right wheel wells.
9. Disconnect deicer boot hoses (if installed) inboard of wing leading edges.
10. Install jack adapters and support wing at jack points with 5-ton jacks (refer to chapter 7, Figure 7-20-201).

NOTE

Do not support complete airplane on three jacks.

11. Remove the two aft and two forward attaching bolt keepers; then remove the four attaching bolts.
12. Carefully raise fuselage off wing.
13. Lower fuselage, and support it at engine pylon and nose jack points.

SEPARATING WING PANELS

Equipment/Tools

- Wing Hoisting Sling, P/N 265-810028.
- Jacks, 5-ton Tripod Type.

Procedure:

1. Separate wing from fuselage.
2. Install P/N 265-810028 sling on wing panel (see Figure 57-10-202).
3. Support other wing panel with a jack at jack point and a suitable support at inboard end (see Figure 57-10-202).
4. Disconnect aileron control cable.
5. Remove access panels on front and rear spars and close-out panels on rear spar.
6. Remove fairing and skid from bottom of wing.
7. Remove attaching bolts and separate wing panels.

JOINING WING PANELS

Equipment/Tools

- Wing Hoisting Sling P/N 265-810028
- Jack — 5-ton Tripod Type

Procedure:

NOTES

- ***Refer to Figure 57-10-202 for this procedure.***
- ***All bolts used for joining wing panels require individual magnetic inspection before installation.***

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MAINTENANCE PRACTICES (Continued)

JOINING WING PANELS (Cont'd)

1. Install front stub spars on wing panels.
2. Install rear stub spars on wing panels.
3. Support wing panel at jack point and at inboard section.
4. With 265-810028 sling, lift panel to be installed and place it next to the other panel.
5. Place centerline rib in position, mate fuel lines to centerline rib fittings, and reinstall clamps on station 24 rib.
6. Place upper splice joint in position, align holes, and retain with two bolts at each end.
7. Place lower splice joint in position, align holes, and retain with two bolts at each end.
8. Install remaining bolts in upper splice joint (see Figure 57-10-202, Sheet 2 of 2).
9. Install remaining bolts in lower splice joint.
10. Install remaining bolts in front stub spar.
11. Install remaining bolts in rear stub spar.
12. Torque all wing attaching bolts per the following:

    NAS625 and 7S28A-5 Series, torque 140 to 208 inch-pounds.

    NAS626 and 7S28A-6 Series, torque 190 to 351 inch-pounds.
13. Install access doors on front and rear spars.
14. Install skid and fairing on bottom of wing.
15. If moistureproofing sealing compound at scupper angle on upper wing skin over aileron sectors is disturbed, replace it with new sealant to prevent moisture from entering aileron sector area.

INSPECTING WING FRONT AND REAR SPARS

The following equipment (or equivalent will be required to perform this inspection):

NAME	MODEL	SIZE
Rod Optic Borescope	BF0-2119D (Direct Vision)	.210 inch dia. 19 inch long
Rod Optic Borescope	BF0-2119A (Right Angle Vision)	.210 inch dia. 19 inch long
Fibre-Optics Light Supply	BLS-97	15K Foot Candle
Fibre-Optics Bundle	F0-400-5A	5MM × 6 inch

Prior to accomplishing this inspection, the areas to be inspected (refer to Figure 57-10-203) must be cleaned of all dirt, grease, and foreign materials using methyl ethyl ketone, trichloroethylene, naphtha (or equivalent) and clean cloths, swabs or brushes.

CAUTION

Use extreme care during the cleaning process not to scratch the painted surfaces of the areas to be inspected.

To inspect the upper and lower spar cap flanges, splice plate flanges, and wing-to-fuselage attach fitting attaching lugs on the front spar, proceed as follows:

1. Remove access doors on fuselage forward of the front spar.
2. Make certain areas to be inspected are clean and dry.
3. Insert applicable borescope between wing-to-fuselage seal and inspect outboard portion of upper spar cap flange and upper portion of wing-to-fuselage attach fitting attaching lugs for cracks or corrosion (area 1 on Figure 57-10-203).
4. Insert applicable borescope up through access door openings to inspect the upper and lower spar cap flanges and splice plate flanges for cracks or corrosion (areas 2 and 3 on Figure 57-10-203).
5. Insert applicable borescope between lower wing-to-fuselage seal and inspect outboard portion of lower spar cap flange and lower portion of wing-to-fuselage attach fitting attaching lugs for cracks and corrosion (area 4 on Figure 57-10-203).

To inspect the upper and lower spar cap flanges and splice plate flanges of the rear spar centersection area, the rear spar lugs and main landing gear actuating cylinder attaching point, proceed as follows:

1. Make certain areas to be inspected are clean and dry.
2. Using applicable borescope inspect rear spar upper and lower cap flanges, rear spar splice plate and flanges and flange-to-web radii in areas inboard of wing-to-fuselage attach fittings (areas 5 and 6 of Figure 57-10-203).
3. With the flaps in the full extended position and the ailerons either up or down as required, use fibre optics referenced to inspect the lugs on the aft side of the rear spar between rear spar sections 37.833 and 270.371 for cracks (area 7 of Figure 57-10-203).
4. Using applicable borescope inspect thickened areas of vertical webs attaching the main gear actuating cylinder to the inboard trailing edge rib structure just aft of wing-to-fuselage attach fitting on the rear spar for cracks (area 8 of Figure 57-10-203).

If no cracks are found, reinstall removed access doors.

If cracks are found, notify Sabreliner Corporation Engineering Department.

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MAINTENANCE PRACTICES (Continued)

LUBRICATE WING-TO-FUSELAGE ATTACH ANGLE BOLTS — S/N 370-1 THRU -9, AND 380-1 THRU -4

Remove the two forward wing-to-fuselage attach bolts, inspect, lubricate and re-install as follows:

1. Jack the airplane using normal jacking procedures (refer to chapter 7). Raise the airplane in a level position, do not lift the landing gear from the floor.
2. Place a fourth jack under the center wing skid jack point. Do not lift the wing with jack; support the adapter only.
3. Remove access panels from lower fuselage at wing forward attach points. (See applicable Access Provisions illustration in chapter 6.)
4. Remove LH and RH bolt retainers (see Figure 57-10-204).
5. Remove LH and RH small flanged access panel (each secured with eight screws) from rear bay to gain access to nuts for forward wing-to-fuselage attach (retainer) bolts.

CAUTION

Do not remove both LH and RH wing-to-fuselage forward attach bolts simultaneously to prevent misalignment of wing and fuselage attach fittings.

6. Remove wing-to-fuselage (retaining) attach bolt.
7. Remove wing-to-fuselage attach bolt. Adjust jack under center wing skid jack point slightly up and down to relieve load on bolt.

NOTES

- ***The attach bolt is tapped for a 3/8 × 24 bolt which may be installed to aid in removing the attach bolt.***
- ***After wing-to-fuselage attach bolt has been removed, do not readjust either of the four jacks until the bolt has been reinstalled.***

8. If attach bolt is corroded, polish bolt with crocus cloth or extra fine emery cloth. Apply a coating of MIL-G-21164 grease to shank after cleaning.
9. Apply a coating of MIL-G-21164 grease to inner bore of inner race of bearing installed in wing-to-fuselage attach support.
10. Reinstall wing-to-fuselage attach bolt and wing-to-fuselage (retaining) attach bolt. Torque retaining bolt to 10-20 inch pounds; safety with lockwire.
11. Repeat steps 7 through 10 for opposite wing-to-fuselage forward attach bolts.
12. Reinstall LH and RH bolt retainers, access doors and access panels removed in steps 3, 4 and 5.
13. Remove jacks.

INSPECT WING EXTERIOR

Remove applicable access panels as necessary and inspect the following:

1. Wing slats for security of attachment; interconnects for cracks, skin for cracks, dents, tears, corrosion, and loose rivets; trailing edge for delamination.
2. Slat tracks for chipping of surface plating, corrosion, and cleanliness.
3. Wing slats for freedom of operation.
4. Upper and lower wing skin for cracks, buckles, dents, corrosion, loose rivets or screws, indication of fuel leaks, and required placards for legibility, trailing edge for delamination.
5. Wing tip for condition and security; fuel vent outlet clear.
6. Single point refueling receptacle cap and gravity fuel filler cap for condition and security.
7. Position and strobe light for condition and security.
8. Static dischargers for condition, security and evidence of looseness.
9. Aileron skin for tears, dents, cracks and security. Ground bonding wires for fraying and security.
10. Aileron hinges for cracks, hinge bolts for looseness and proper safety.
11. Aileron push-pull rods, bellcranks, sector wheel, cables and cable pulleys for condition, security and alignment.
12. Wing flap skin for cracks, buckles, dents, corrosion, and loose rivets or screws, trailing edge for delamination; seal for cuts or tears. Ground bonding wires for fraying and security.
13. Flap tracks and rollers for corrosion, cleanliness, chipping of surface plating; supports for security and serrated washers for looseness.
14. Flap actuator shaft and anti-rotation sleeve for cracks or breaks in areas adjacent to sleeve attach screws.
15. Deicer boots for damage and security (if applicable).
16. Flap actuator interconnect flexible shaft and couplings visually for damage and security, disconnects and coupling for safety.
17. Accessible fuel lines in wheel well for chafing, security, leakage and condition.

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MAINTENANCE PRACTICES (Continued)

INSPECT WING EXTERIOR (Cont'd)

18. Accessible hydraulic lines and components in wheel well for chafing, security, leakage and condition.
19. Install any access panels that were removed to accomplish the visual inspection.

INSPECT LOWER WING SKIN FUEL DRAIN VALVE CAVITIES AND FUEL DRAIN VALVE SUPPORTS FOR CORROSION

To inspect drain valve cavities, proceed as follows:

1. Defuel airplane. (Refer to chapter 28.)
2. Remove landing gear wing fairing.
3. Remove the ten fuel drain valves as depicted. See Figure 57-10-205 for local manufactured tool and valve location. If screens are installed on fuel drain valves, remove screens per S/B 81-1.
4. Inspect fuel drain valve cavities for corrosion and presence of cati-coat primer (cati-coat primer is buff in color, same color as inside of wing tank).
5. If corrosion is present and/or absence of cati-coat primer, refer to S/B 71-1 for rework procedures.

To inspect the fuel drain valve supports (P/N 277-481005) proceed as follows:

1. Through fuel drain cavity opening, attach a length of safety wire to each fuel drain valve support.
2. Remove two screws securing each drain valve support in place.
3. Remove each drain valve support from inside of the wing by sliding it out through the fuel drain cavity.
4. Visually inspect all surfaces of the drain valve support for indications of corrosion.
5. If there are no indications of corrosion and the cati-coat primer film is intact, proceed to Step 6. If corrosion is detected the following substeps must be accomplished.
   A. Remove all primer from drain valve support.
   B. Determine extent and depth of corrosion damage.
   C. If corrosion depth is more than 0.010 inch, the drain valve support must be replaced. If corrosion damage is less than 0.010 inch, carefully remove all corrosion residue using power operated cloth buffing wheels and a fine (325 grit) aluminum oxide buffing compound. The corrosion removal area should be blended into the surrounding area to form a saucer shaped depression.
   D. Brush cadmium plate all bare or exposed areas of the base metal.
   E. Apply a coating of cati-coat primer to the drain valve support. Use the cati-coat primer touch-up kit (P/N 306-053015) or refer to the Structural Repair Manual (NA-66-1032) for application instructions.
6. Reinstall drain valve supports in wing. Torque attaching screws to 20-25 inch-pounds.

CAUTION

Do not overtorque drain valves, overtorque will cause distortion of the valve head which may cause fuel leaks.

7. Install new O-rings (MS29513-129) on drain valves. Install drain valves and torque to 50-70 inch-pounds and install wing fairing.
8. Fuel airplane and check fuel drain valves and wing fairing for leaks.

INSPECT WING JOINT RIB ASSEMBLY FOR CORROSION

The following special inspection or equipment will be required:

Cold Light Supply Model ILK-4
Probe, 010869, A080-071-045-45 or Equivalent
Probe, 010870, A080-060-110-60

This borescope equipment is available from Olympus Corp. of America, Industrial Fiberoptics, 4 Nevada Drive, New Hyde Park, NY 10040.

CAUTION

Corrosion in the wing joint rib area is caused by not draining water from the wing tank fuel sumps daily.

1. Defuel airplane (refer to chapter 28).
2. Remove necessary panels to gain access to the four lower inboard wing tank fuel drain valves.
3. Loosen four fuel drain valves using a spanner-type wrench, and unscrew each fuel drain valve from the fuel drain valve support (see Figure 57-10-205 for valve locations and a description of the removal and installation tool).
4. Remove two screws securing each fuel drain valve support to the wing lower skin.
5. Remove each valve support from wing through fuel drain valve opening.
6. Remove access doors, 2 ea, inside of fuel boost pump bay (see Figure 57-10-207).

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MAINTENANCE PRACTICES (Continued)

INSPECT WING JOINT RIB ASSEMBLY FOR CORROSION (Cont'd)

7. Insert fiber optic borescope through fuel drain valve and access door opening and examine entire length of lower flange of wing center rib for evidence of corrosion (see Figure 57-10-207).

NOTE

Corrosion is present if paint has bubbled up or metal surface is exposed through the paint.

8. See Special Inspections, chapter 5. If corrosion is detected on edge of flange (see Figure 57-10-207) take the appropriate corrective action listed in step A or B below. If no corrosion is detected, proceed to step 9.
   A. On Model NA 265-70 airplanes S/N 370-1 thru -9 and Model NA 265-80 airplanes S/N 380-1 thru -43, if corrosion is present the wing joint rib must be inspected ultrasonically within 30 days of date corrosion is detected. Contact Sabreliner Corporation in St. Louis, MO for ultrasonic inspection. If ultrasonic inspection reveals cracks in wing joint rib, the wing must be demated, seperated, and wing joint rib removed for crack repair and corrosion correction. If no cracks are found during ultrasonic inspection, the corrosion must be corrected within one year from date of discovery. Corrosion repair will require wing demate and separation. Refer to Corrosion Control Manual, Report SR-76-023, for rework.
   B. On Model NA 265-80 airplanes S/N 380-44 and Subs, ultrasonic inspection is not required. Corrosion must be corrected within one year from date of discovery. Corrosion repair will require wing demate and separation. Refer to Corrosion Control Manual, Report SR-76-023, for rework
9. Insert a serviceable drain valve support into each of four fuel drain valve opening and align screw holes in support with screw holes in lower wing skin. Install screws and torque to 20-25 inch-pounds.

CAUTION

Do not overtorque drain valves. Overtorquing will cause distortion of the valve head which may cause fuel leaks.

10. Using Petrolatum VV-P-236 (or equivalent), lightly lubricate and install new O-rings (MS29513-129) on serviceable drain valves. Thread valves into valve supports using wrench shown in Figure 57-10-205. Torque to 50-70 inch-pounds.
11. Fuel airplane and check replaced fuel drain valves for operation and leaks.
12. Reinstall access panels removed in step 2.

JOINING FUSELAGE AND WING

Equipment/Tools:

- P/N 306-810021-11 Sling
- P/N 50K25117 Jack — 5-Ton Hydraulic Tripod
- P/N 265-810125 Kit — Jacking and Mooring

Procedure:

NOTE

Refer to Figure 57-10-201 for this procedure.

1. Support wing at jack points.
2. Using hoisting sling, lower fuselage onto wing. Keep a preload on fuselage sling until wing and fuselage are mated and fuselage nose jack is installed (step 7). Do not support fuselage on wing center section jack.

CAUTION

Lower fuselage carefully to prevent damage to structure or loose lines.

3. With fuselage positioned, check clearance at forward and aft attach fittings, and install wing forward attach bolts.
4. Install wing aft attach bolts.

CAUTION

On airplanes with teflon coated bearings, to prevent damage to the teflon coating, do not apply lubrication to the four wing attach bolts.

CAUTION

Tool P/N T-1062755 (Figure 57-10-206) is used to align bearing with surrounding structure prior to installing wing attach bolts. On airplanes with teflon coated bearings, this tool shall be used to prevent damage to the teflon coating on the bearings.

5. Torque wing forward attach bolts to 10-20 inch-pounds and safety with MS20995F47 wire. Install access doors with keepers.
6. Torque wing aft attach bolts to 60-75 inch-pounds and safety with MS20995F32 wire. Install brackets with keepers.

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MAINTENANCE PRACTICES (Continued)

JOINING FUSELAGE AND WING (Cont'd)

7. Install fuselage nose jack; then remove wing center section jack and hoisting sling.
8. In left wheel well, connect hydraulic and fuel lines, electrical wiring, fuel quantity indicating system wiring, and landing gear emergency release cable.
9. In right wheel well, connect hydraulic and fuel lines, electrical wiring, fuel quantity indicating system wiring, landing gear emergency release cable, and aileron control rod.
10. At wing trailing edge, install 265-100034 support (see Figure 57-10-201, Sheet 2 of 2).

NOTE

Maximum clearance between the inboard wing trailing edge retainer plates and the pins is 0.030.

11. Install inboard wing trailing edge sections.
12. Install entire wheel well detachable fairing structure extending from wing rear spar to aft end of wheel wells.
13. Check that all clamps are properly installed on deicer boot air supply lines, hydraulic lines, and electrical wiring that were connected when fuselage and wing were mated.
14. Connect ADF loop and sense antenna coaxial cables (4) at fuselage floor connectors and clamp to fuselage structure.
15. Connect batteries and service fuel and hydraulic power systems (refer to chapter 12). Make full operational check of landing gear system, flight control system, pneumatic deicer system, and fuel system.

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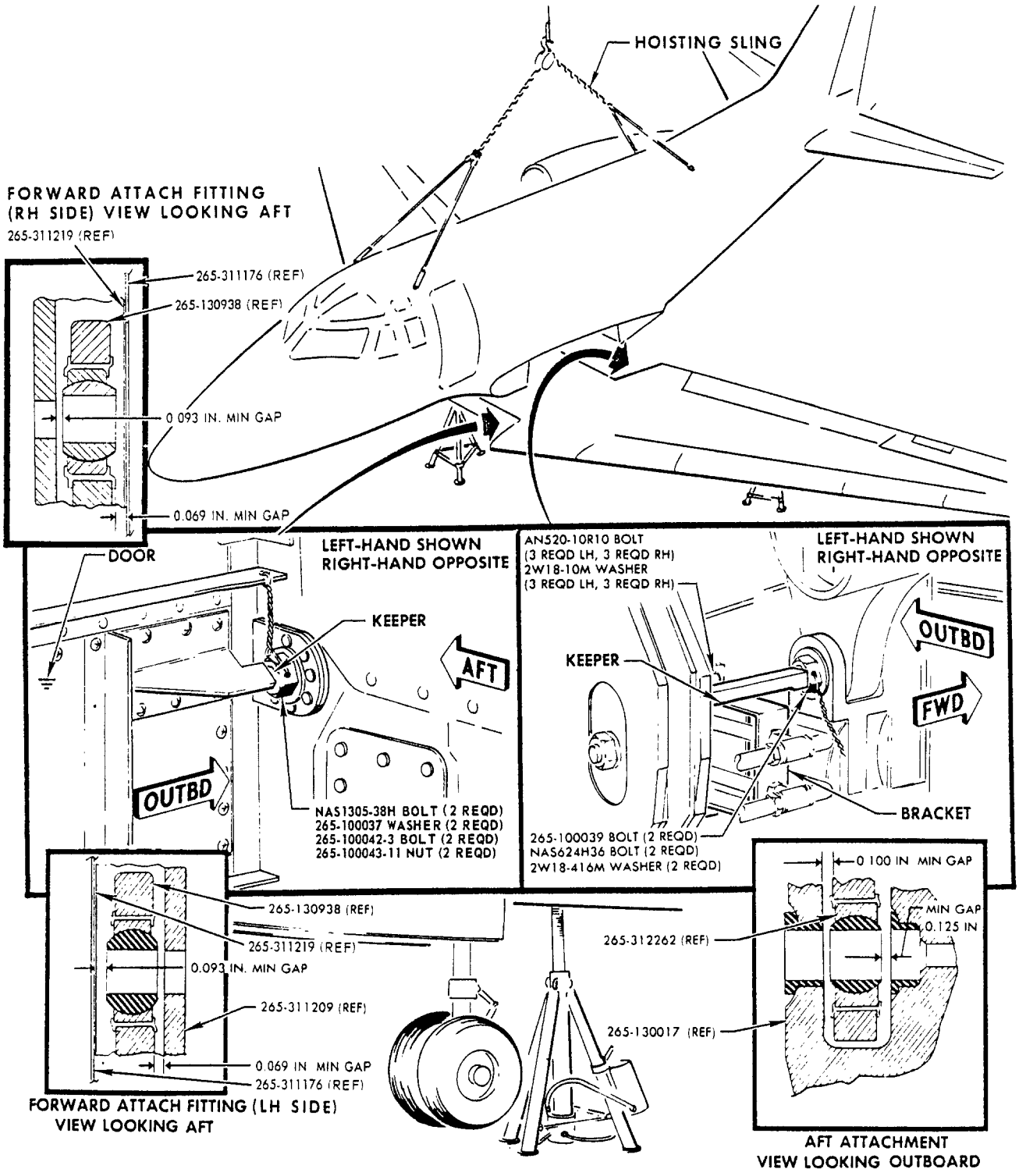

Figure 57-10-201 — Joining Fuselage and Wing (Sheet 1 of 2)

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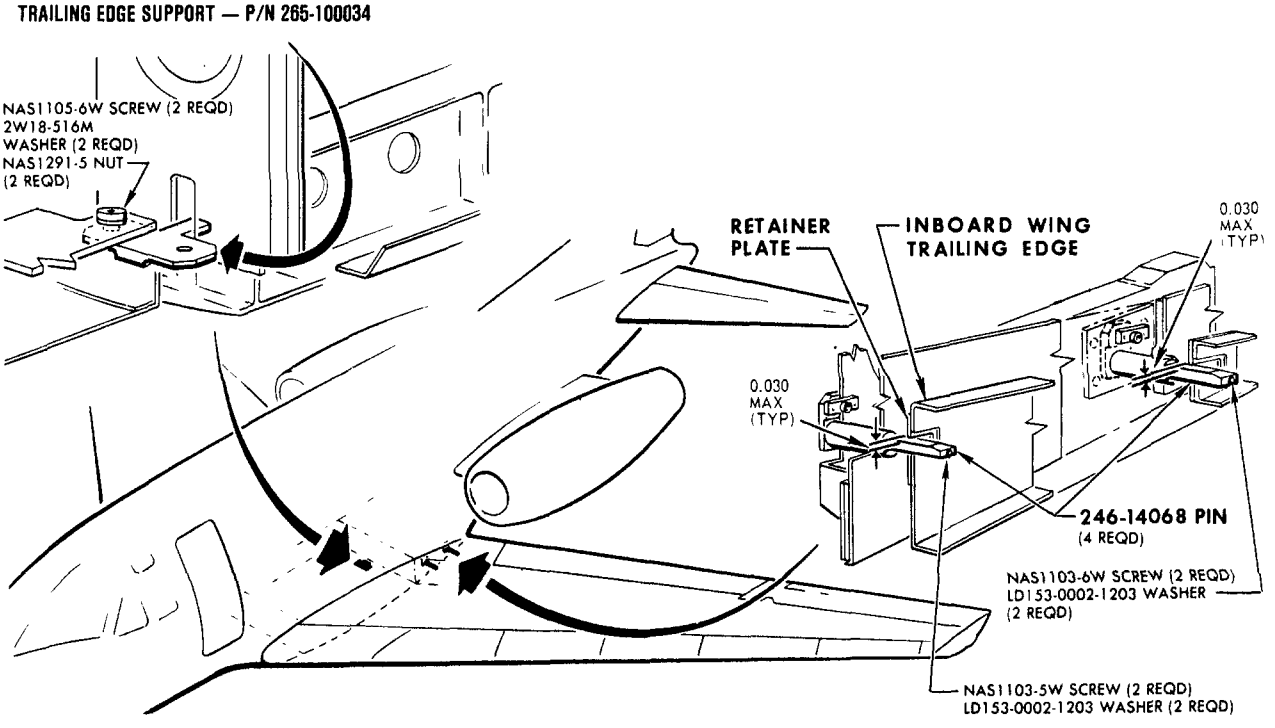

Figure 57-10-201 — Joining Fuselage and Wing (Sheet 2 of 2)

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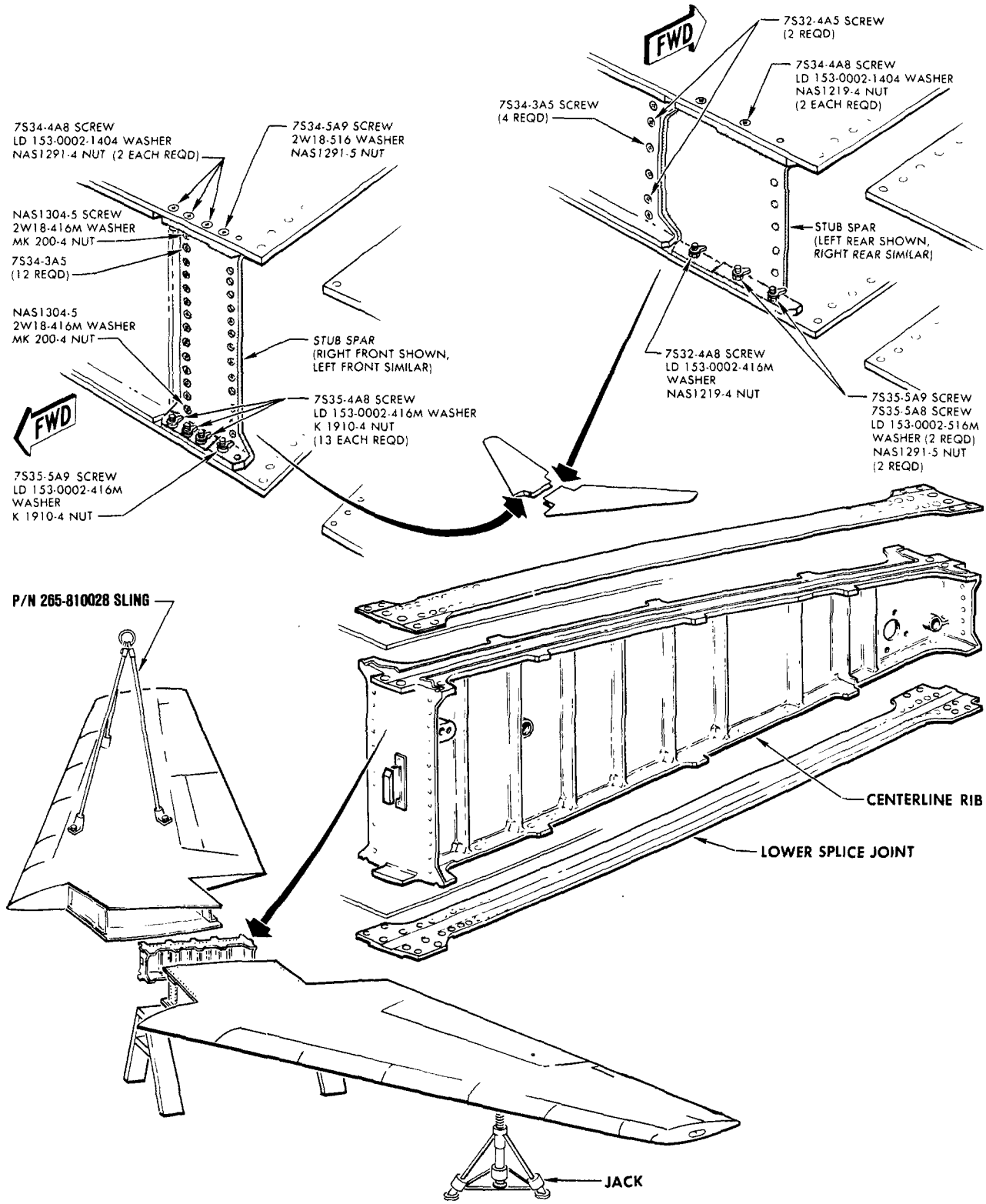

Figure 57-10-202 — Joining Wing Panels (Sheet 1 of 2)

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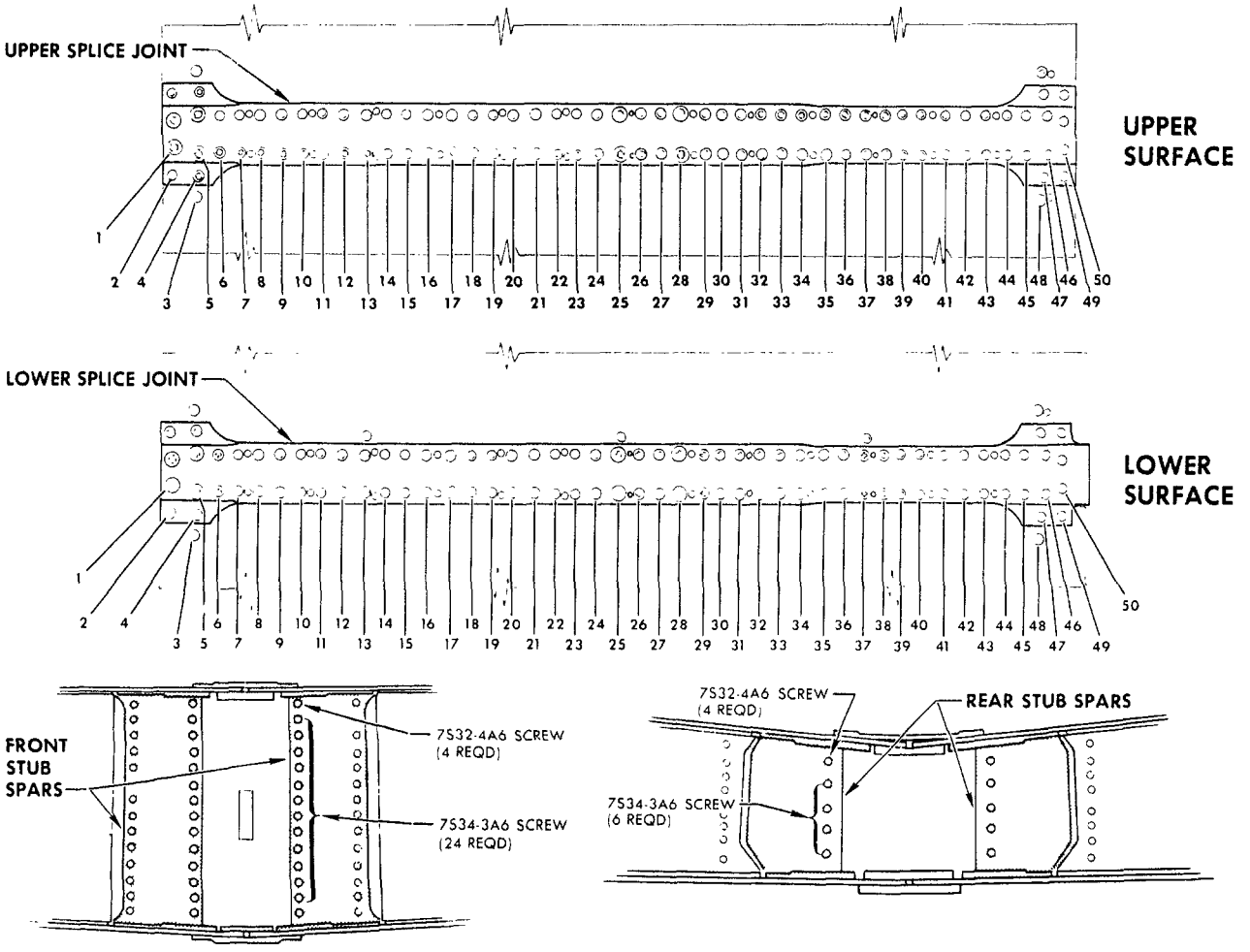

UPPER SURFACE

NO.	SCREW (2 REQD)	NO	SCREW (2 REQD)	NO.	SCREW (2 REQD)
1	NAS626-17	18	NAS625-15	35	NAS625-13
2	NAS625-12	19	NAS625-15	36	NAS625-13
3	7S28A-5-11	20	NAS626-15	37	NAS625-13
4	NAS625-13	21	NAS626-14	38	NAS625-13
5	NAS626-17	22	NAS625-14	39	NAS625-13
6	NAS625-17	23	NAS625-14	40	NAS625-13
7	NAS625-17	24	NAS626-14	41	NAS625-13
8	NAS626-16	25	NAS626-14	42	NAS625-13
9	NAS626-16	26	NAS625-14	43	NAS625-13
10	NAS625-16	27	NAS625-14	44	NAS625-13
11	NAS625-16	28	NAS626-14	45	NAS625-13
12	NAS626-16	29	NAS625-14	46	NAS625-13
13	NAS626-16	30	NAS625-14	47	NAS625-11
14	NAS625-15	31	NAS625-14	48	7S28A-5-10
15	NAS625-15	32	NAS625-14	49	NAS625-11
16	NAS626-15	33	NAS625-14	50	NAS625-11
17	NAS626-15	34	NAS625-13		

LOWER SURFACE

NO.	SCREW (2 REQD)	NO.	SCREW (2 REQD)	NO	SCREW (2 REQD)
1	7S28A-6-17	18	NAS625-15	35	7S28A-5-14
2	7S28A-5-16	19	NAS625-15	36	7S28A-5-14
3	7S28A-5-10	20	NAS626-15	37	7S28A-5-14
4	7S28A-5-15	21	NAS626-15	38	7S28A-5-14
5	7S28A-6-17	22	NAS625-15	39	NAS625-14
6	7S28A-5-17	23	NAS625-15	40	NAS625-14
7	NAS625-17	24	NAS626-15	41	NAS625-14
8	NAS626-16	25	7S28A-6-15	42	NAS625-14
9	NAS626-16	26	7S28A-5-15	43	NAS625-14
10	NAS625-16	27	7S28A-5-15	44	NAS625-14
11	NAS625-16	28	7S28A-6-15	45	NAS625-14
12	NAS626-16	29	7S28A-5-15	46	NAS625-14
13	NAS626-16	30	7S28A-5-15	47	NAS625-14
14	NAS625-16	31	7S28A-5-14	48	7S28A-5-8
15	NAS625-16	32	7S28A-5-14	49	NAS625-12
16	NAS626-16	33	7S28A-5-14	50	NAS625-14
17	NAS626-16	34	7S28A-5-14		

NOTE

- ***All bolts used for joining wing panels require individual magnetic inspection before installation.***
- ***NAS625 and 7S28A-5 Series, torque 140 to 208 inch-pounds.***

  NAS626 and 7S28A-6 Series, torque 190 to 351 inch-pounds.

Figure 57-10-202 — Joining Wing Panels (Sheet 2 of 2)

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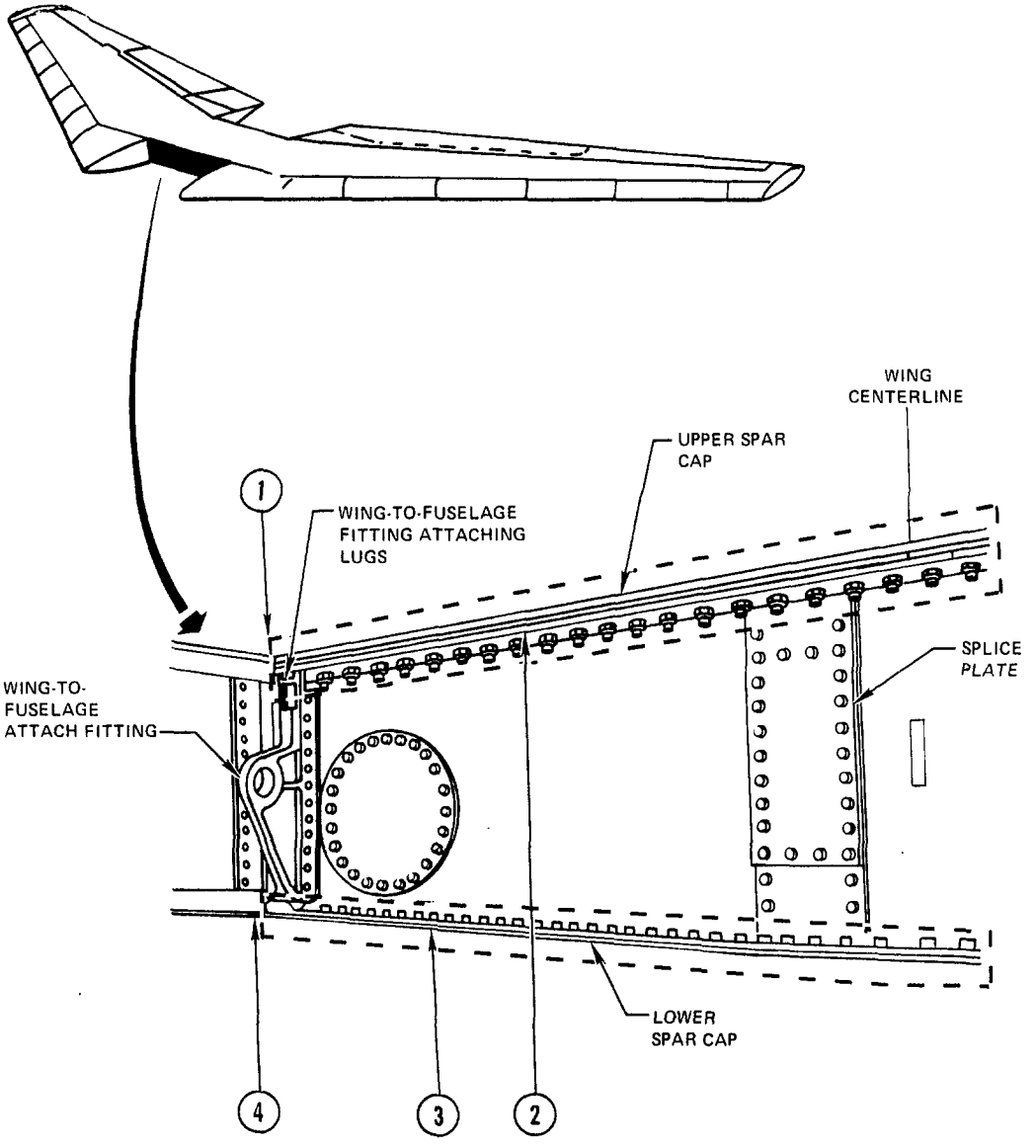

VIEW OF WING FRONT SPAR - CENTER SECTION (RH)

AREA (1) Outboard end of upper spar cap flange and upper portion of wing-to-fuselage fitting attaching lugs

AREA (2) Upper spar cap flange and splice plate flange

AREA (3) Lower spar cap flange and splice plate flange

AREA (4) Outboard end of lower spar cap flange and lower portion of wing-to-fuselage fitting attaching lugs

S370-2-11-6

Figure 57-10-203 — Inspecting Wing Front and Rear Spars (Sheet 1 of 2)

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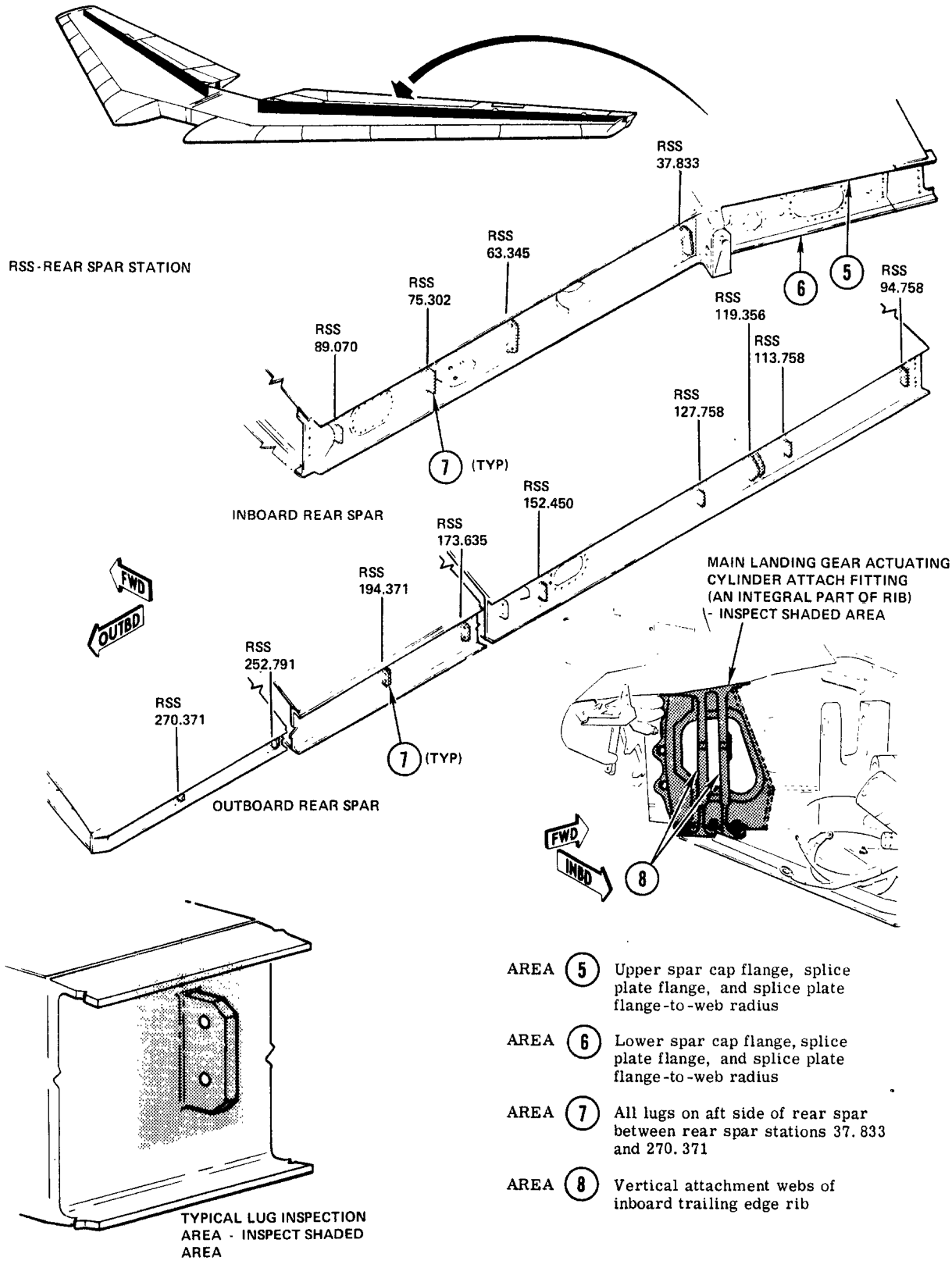

AREA (5) Upper spar cap flange, splice plate flange, and splice plate flange-to-web radius

AREA (6) Lower spar cap flange, splice plate flange, and splice plate flange-to-web radius

AREA (7) All lugs on aft side of rear spar between rear spar stations 37.833 and 270.371

AREA (8) Vertical attachment webs of inboard trailing edge rib

S370-2-11-7

Figure 57-10-203 — Inspecting Wing Front and Rear Spars (Sheet 2 of 2)

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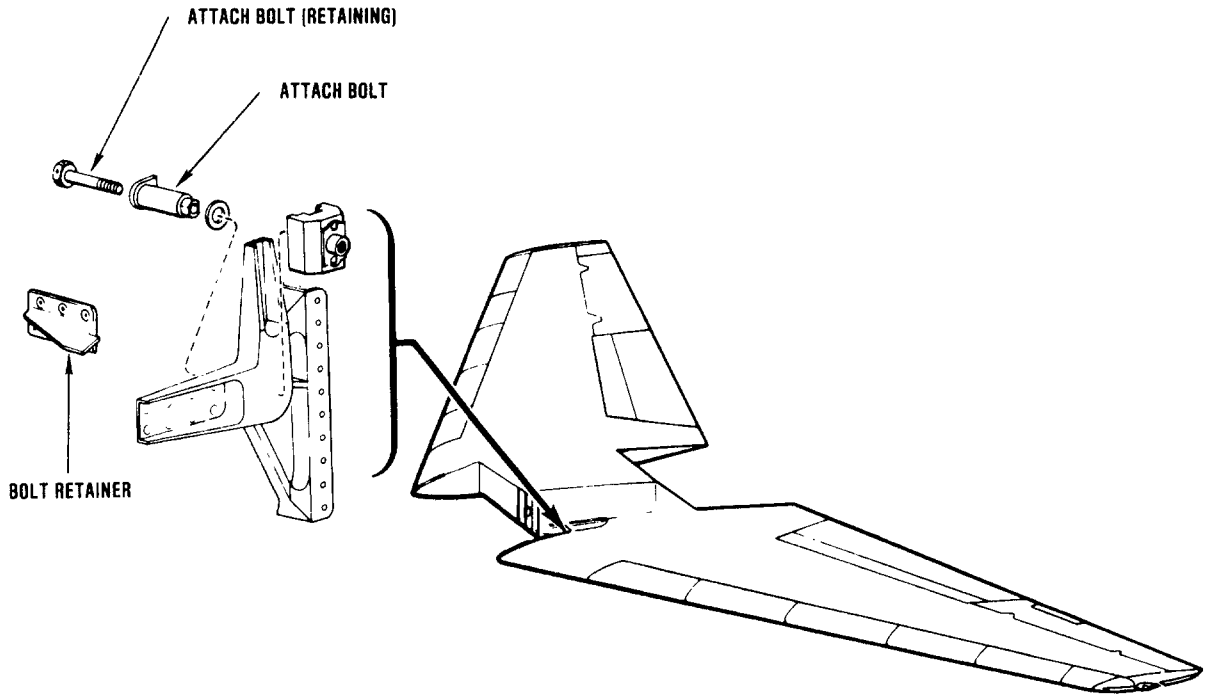

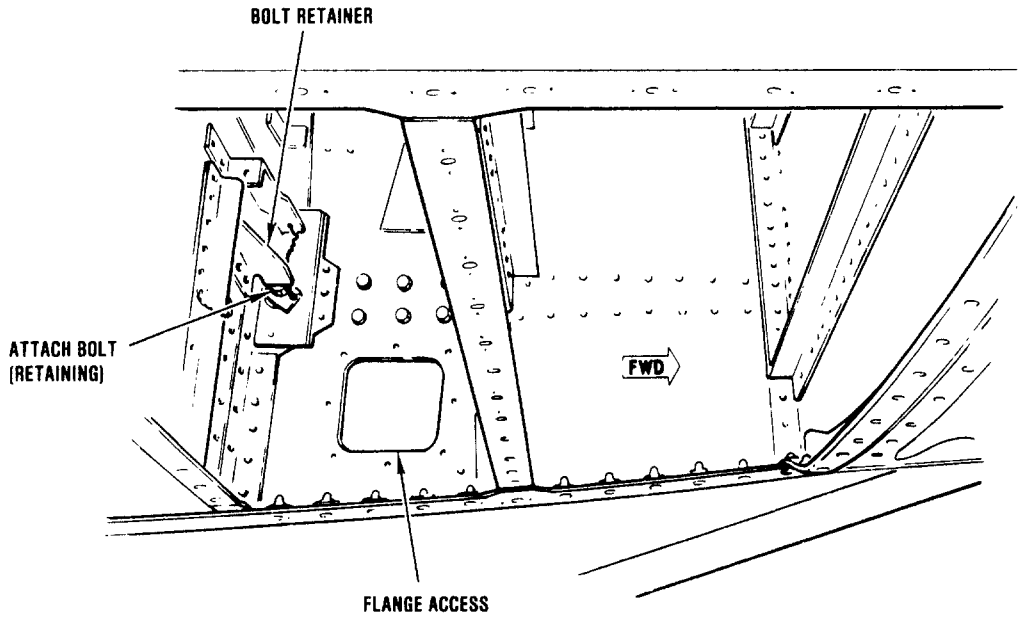

S304-2-00-11

Figure 57-10-204 — Lubrication of Forward Wing-to-Fuselage Attach Bolts and Bearings
S/N 370-1 Thru -9, 380-1 thru -4

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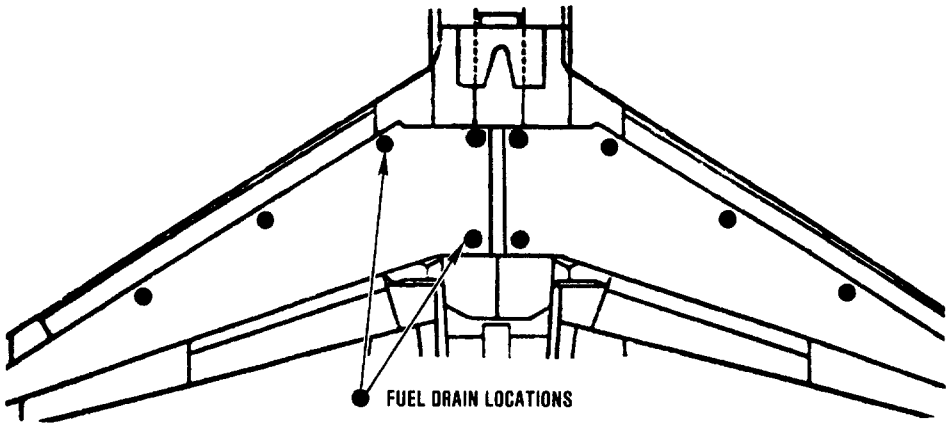
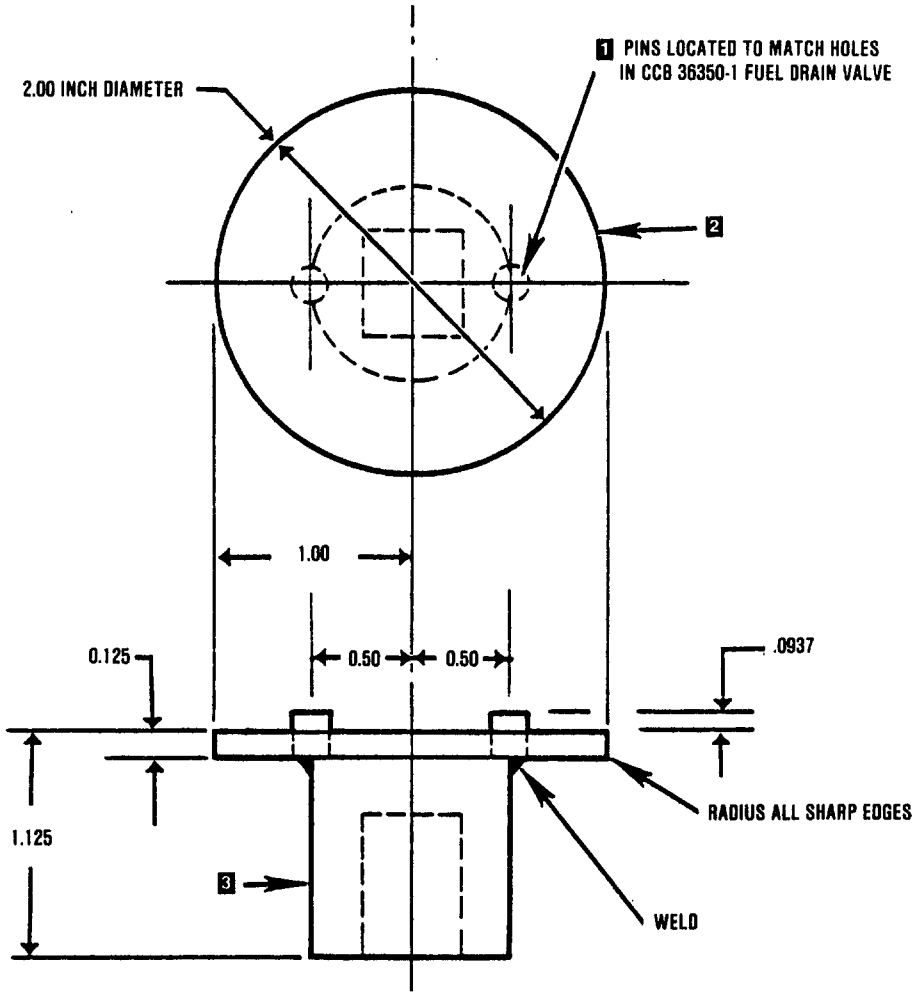


DRAIN VALVE REMOVAL AND INSTALLATION TOOL (LOCAL MANUFACTURE)



1 STEEL 3/16 INCH DOWEL PIN
2 COLD ROLL STEEL 1/8 INCH PLATE
3 STANDARD 1/2 INCH DRIVE ADAPTER

Figure 57-10-205 — Lower Wing Skin Drain Valves

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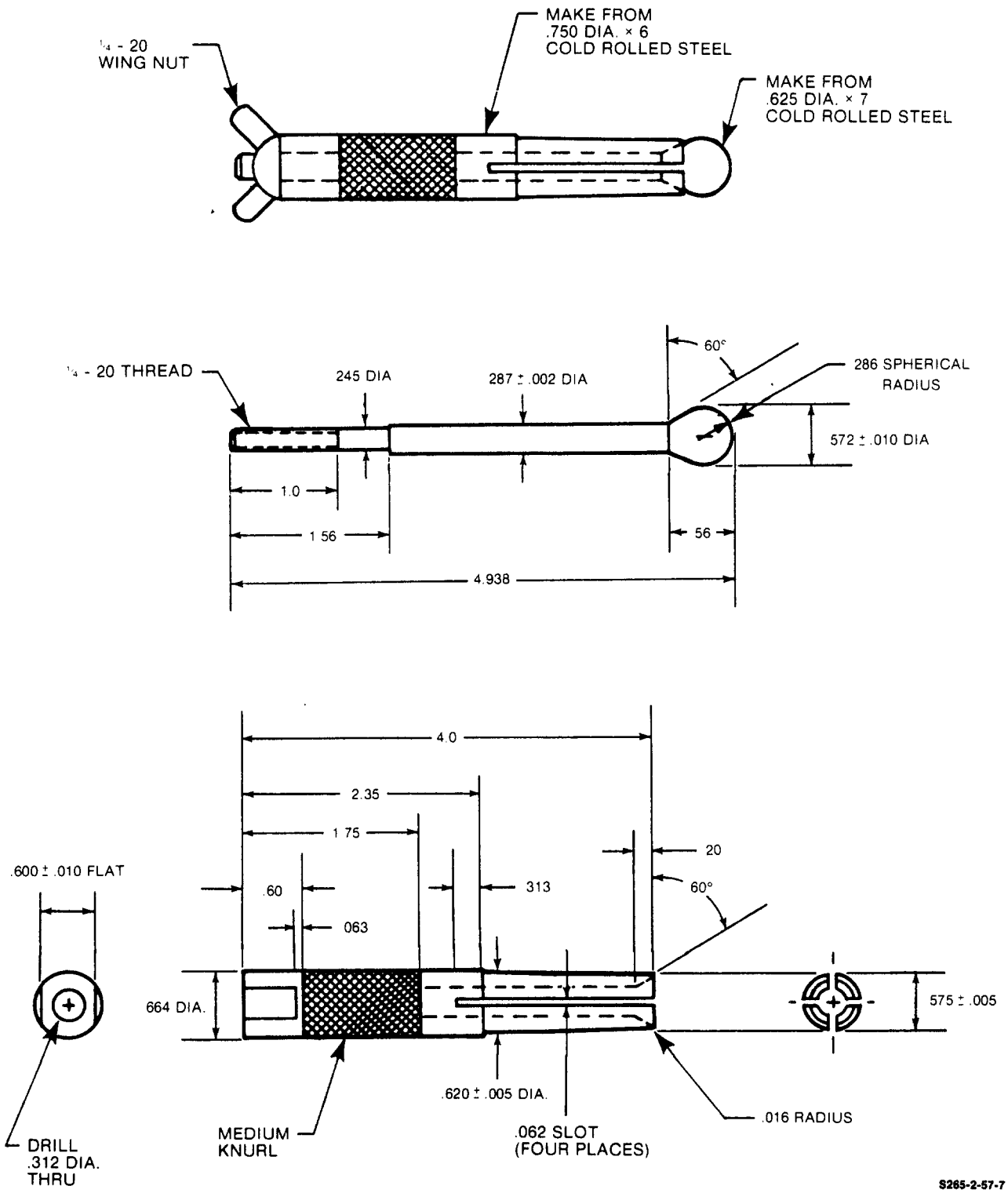

NOTES

- *All dimensions given in inches unless otherwise noted.*
- *Tool P/N T-1062755.*

Figure 57-10-206 — Fuselage to Wing Bolt Alignment Tool (Local Manufacture)

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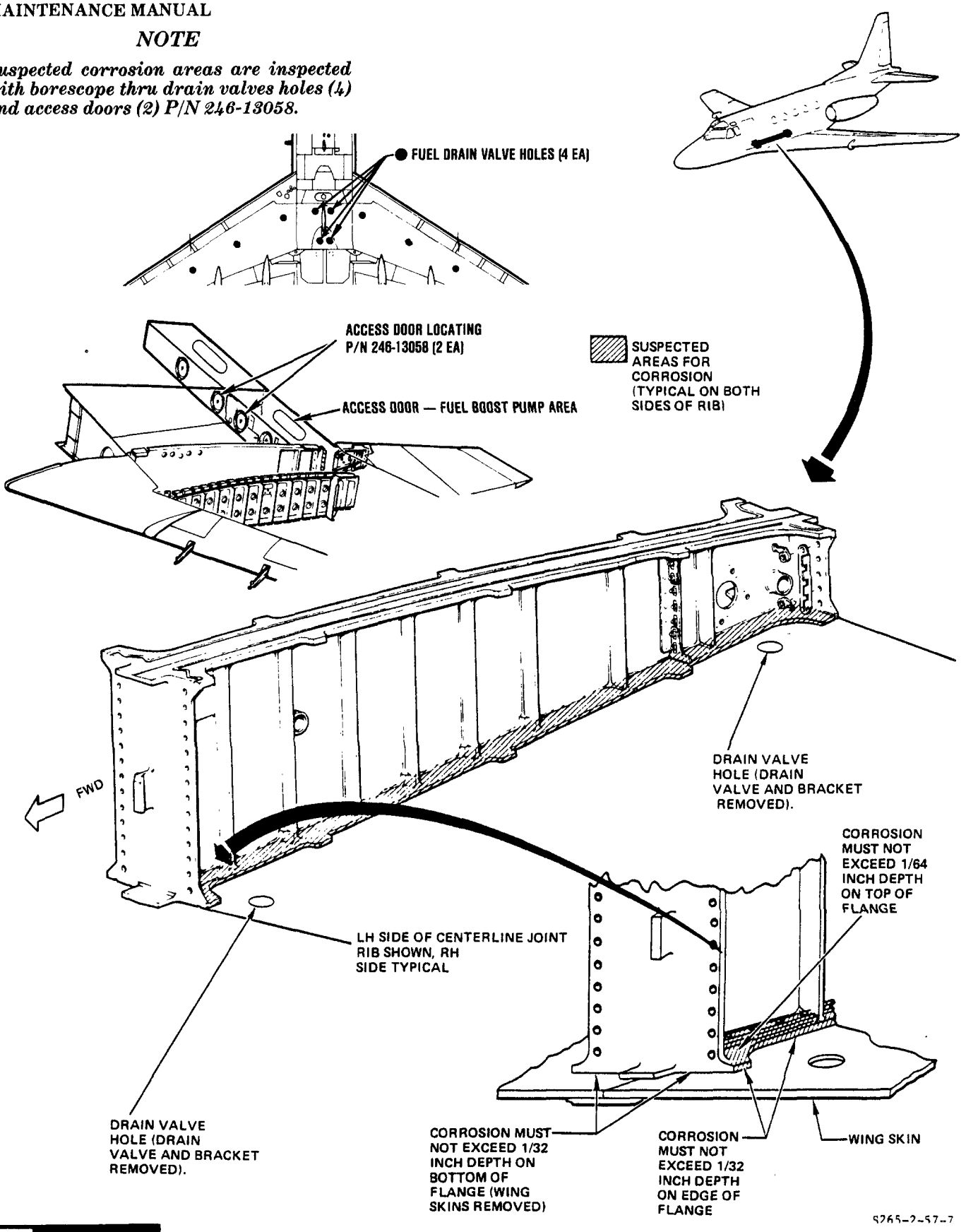
NOTE

Suspected corrosion areas are inspected with borescope thru drain valves holes (4) and access doors (2) P/N 246-13058.


FUEL DRAIN VALVE HOLES (4 EA)
ACCESS DOOR LOCATING P/N 246-13058 (2 EA)
ACCESS DOOR — FUEL BOOST PUMP AREA
SUSPECTED AREAS FOR CORROSION (TYPICAL ON BOTH SIDES OF RIB)
FWD
DRAIN VALVE HOLE (DRAIN VALVE AND BRACKET REMOVED).
CORROSION MUST NOT EXCEED 1/64 INCH DEPTH ON TOP OF FLANGE
LH SIDE OF CENTERLINE JOINT RIB SHOWN, RH SIDE TYPICAL
DRAIN VALVE HOLE (DRAIN VALVE AND BRACKET REMOVED).
CORROSION MUST NOT EXCEED 1/32 INCH DEPTH ON BOTTOM OF FLANGE (WING SKINS REMOVED)
CORROSION MUST NOT EXCEED 1/32 INCH DEPTH ON EDGE OF FLANGE
WING SKIN
9265-2-57-7

MAJOR CHANGE

Figure 57-10-207 — Corrosion Repair — Wing Joint Rib Assembly

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MAINTENANCE PRACTICES (Continued)

INSPECTION OF WING FLAP TRACK OUTBOARD SPAR LUG

Equipment/Tools

- Screwdriver, Torq-Set, EX-170-5/16

Consumables

- Alodine 1201
- Epoxy-Polyamide Primer MIL-P-23377
- Penetrant 2 Dy-Chek
- Remover 3 Dy-Chek
- Developer Dy-Chek

Inspection

NOTE

This procedure is applicable to both left and right hand outboard flap track wing spar lugs.

1. Remove flap. Refer to Sabreliner Maintenance Manual Chapter 27.
2. Remove outboard wing flap track. Refer to Sabreliner Maintenance Manual, Chapter 27.
3. Remove paint from outboard flap track wing spar lug. Make a dye penetrant check of the entire lug for cracks. Make sure there are no cracks around the spar lug radius or radiating out from the bolt hole. See figure 57-10-208.

NOTE

If the spar lug for outboard flap track is not cracked, perform steps 4 and 5.

4. Radius any sharp edges (upper and lower) on spar lug, using fine emery cloth. See figure 57-10-208. Clean, brush alodine treat and paint all areas where paint was removed using Epoxy-Polyamide primer.
5. Install outboard flap track as follows;
   A. Install two screws attaching flap track to the lower wing skin; torque nuts 140 to 203 inch-pounds using Torq-Set screwdriver.
   B. Check for alignment of upper holes in flap track with hole in spar lug. See figure 57-10-208. If holes do not align, remove two screws installed in step A; then shim under flap track as necessary to obtain slip fit for NAS1105-8 attaching bolt with flap track attach screws installed and nuts torqued.

NOTE

Use 2034-T3 aluminum for shim stock. Paint shim(s) with Epoxy-Polyamide primer.

   C. Install NAS1105-8 bolt. Torque nuts 140 to 203 inch-pounds.
   D. Perform step 14.

NOTE

If spar lug for outboard flap track is cracked, perform steps 6 thru 14.

6. Defuel the airplane. Refer to Sabreliner Maintenance Manual Chapter 28.
7. Remove flapper valve access door located approximately eight inches inboard of spar lug for access to rework. Refer to Sabreliner Illustrated Parts Catalog, figure 2-10.
8. Accomplish rework per Sabreliner Modification Drawing 282-130004.
9. After rework is completed, clean wing fuel tank free of all metal chips, contamination, etc.
10. Install flapper vale access door.
11. Refuel airplane and check for leaks.
12. Clean, brush alodine treat, and repaint all areas where paint was removed, using epoxy-polyamide primer.
13. Install outboard flap track per step 5, this procedure.
14. Install flap and check rigging and operation in accordance with existing instructions in Sabreliner Maintenance Manual. Refer to Chapter 27.

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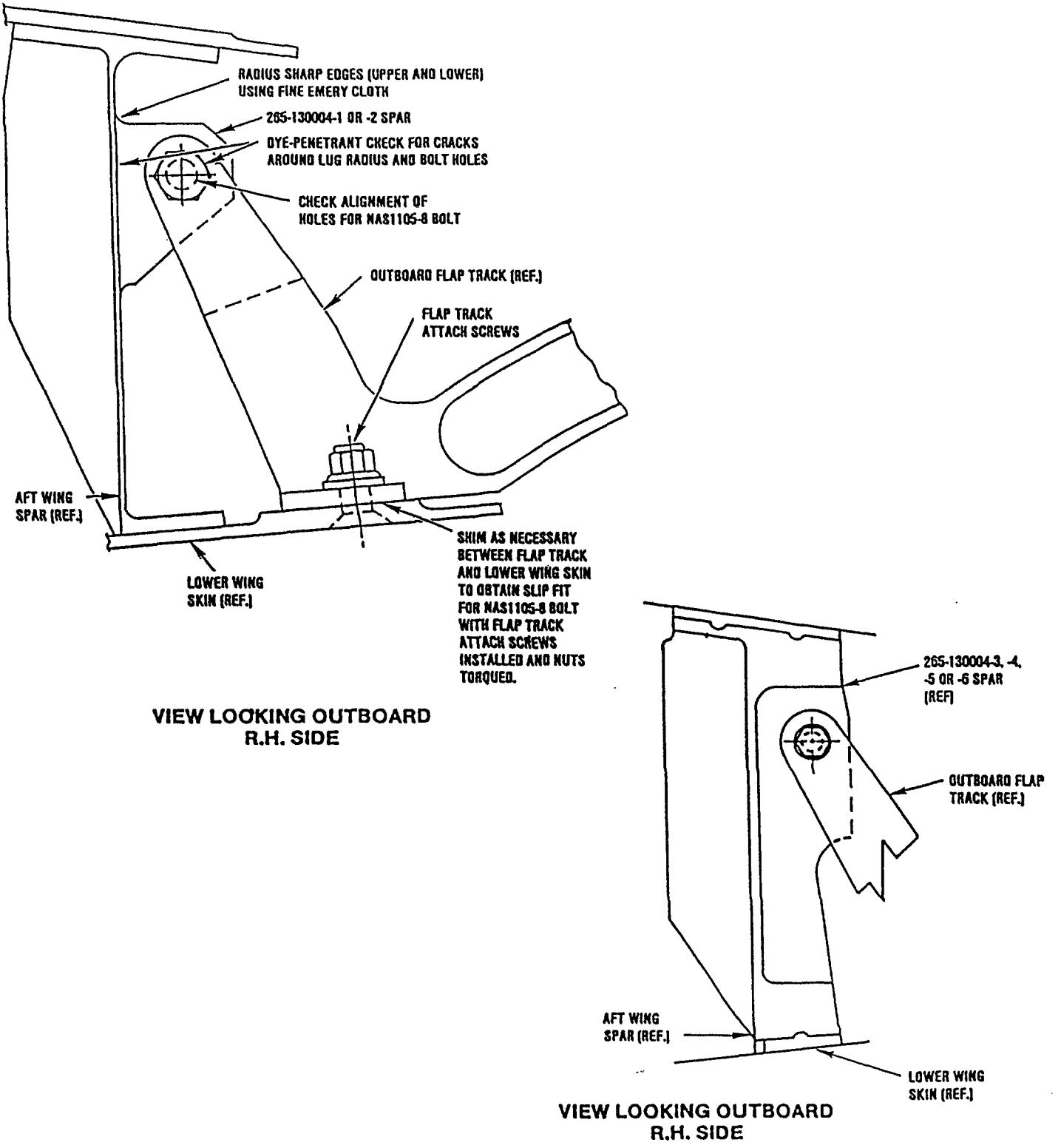
RADIUS SHARP EDGES (UPPER AND LOWER) USING FINE EMERY CLOTH
265-130004-1 OR -2 SPAR
DYE-PENETRANT CHECK FOR CRACKS AROUND LUG RADIUS AND BOLT HOLES
CHECK ALIGNMENT OF HOLES FOR NAS1105-8 BOLT
OUTBOARD FLAP TRACK (REF.)
FLAP TRACK ATTACH SCREWS
AFT WING SPAR (REF.)
LOWER WING SKIN (REF.)
SHIM AS NECESSARY BETWEEN FLAP TRACK AND LOWER WING SKIN TO OBTAIN SLIP FIT FOR NAS1105-8 BOLT WITH FLAP TRACK ATTACH SCREWS INSTALLED AND NUTS TORQUED.
VIEW LOOKING OUTBOARD R.H. SIDE
265-130004-3, -4, -5 OR -6 SPAR (REF)
OUTBOARD FLAP TRACK (REF.)
AFT WING SPAR (REF.)
LOWER WING SKIN (REF.)
VIEW LOOKING OUTBOARD R.H. SIDE

Figure 57-10-208 — Inspection of Wing Flap Track Outboard Spar Lug

NEW

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AUXILIARY STRUCTURE - DESCRIPTION

1. General

The auxiliary structure of the wings consists of the wing tips. The wing tips are laid-up glass cloth assemblies with aluminum alloy T-shaped caps fastened to the extreme outboard edges. The aluminum alloy cap is electrically bonded to the aluminum alloy wing structure. This tip is designed to protect the integral wing fuel tank from the effects of lightning strikes.

AUXILIARY STRUCTURE

MAINTENANCE PRACTICES

REMOVING WING TIP

WARNING

The clearance between the compass compensator and wingtip structure are very close. Extreme care should be taken when removing or installing the wingtips. Flexing the wingtip up and down during installation or removal may result in damage to the compensator.

1. Remove attaching screws and pull tip outward.
2. Disconnect navigation light wiring.
3. Disconnect bonding jumper.

INSTALLING WING TIP

WARNING

The clearance between the compass compensator and wingtip structure are very close. Extreme care should be taken when removing or installing the wingtips. Flexing the wingtip up and down during installation or removal may result in damage to the compensator.

1. Connect navigation light wiring
2. Connect bonding jumper.
3. Place wing tip in position and install attaching screws.

NOTE

When installing a new wing tip, some trimming may be necessary to obtain the proper skin gap. The skin gap should be 0 to 1/16 inch.

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AUXILIARY STRUCTURE

INSPECTION/CHECK

INSPECT — WING TIPS, WING FRONT SPAR, LEADING EDGE SKIN, RIB AND STIFFENERS

Equipment/Tools

- None

Consumables

- Sealant PR1422GB

Inspection — Wing Tips

WARNING

The clearance between the compass compensator and wingtip structure are very close. Extreme care should be taken when removing or installing the wingtips. Flexing the wintip up and down during installation or removal may result in damage to the compensator.

1. Remove attaching screws and pull tip outward.
2. Disconnect navigation light wiring.
3. Disconnect bonding jumper.
4. Remove outboard trailing edge. (See Figure 57-20-601.)
5. Remove outboard leading edge.
6. Clean nut plate area of old sealant.
7. Inspect nut plates and surrounding area for cracks and/or corrosion. (See Figure 57-20-601.)

NOTE

Any repairs accomplished should be made in accordance with Sabreliner Structural Repair Manual (NA-66-1032) and/or Sabreliner Corrosion Control Manual (SR-76-23).

8. Install outboard leading edge.
9. Install outboard trailing edge.
10. Connect navigation light wiring.
11. Connect bonding jumper.
12. Place wingtip in position and install attaching screws.
13. Seal wingtip, outboard leading edge, and outboard trailing edge using (PR 1422 GB) sealant, per Sabreliner Corrosion Control Manual (SR-76-23), chapter 20.

Inspection — Wing Front Spar, Leading Edge Skin, Rib and Stiffeners

1. Remove screws from each wing leading edge door to gain access. (LH and RH) (See Figure 57-20-601.)
2. Inspect wing leading edge spar, leading edge skin, rib, and stiffeners for cracks and corrosion.
3. Inspect nut plates and surrounding area on wing leading edge skin for cracks and corrosion.

NOTE

Any repairs accomplished should be made in accordance with Sabreliner Structural Repair Manual (NA-66-1032) and/or Sabreliner Corrosion Control Manual (SR-76-23).

4. Reinstall each wing leading edge door.

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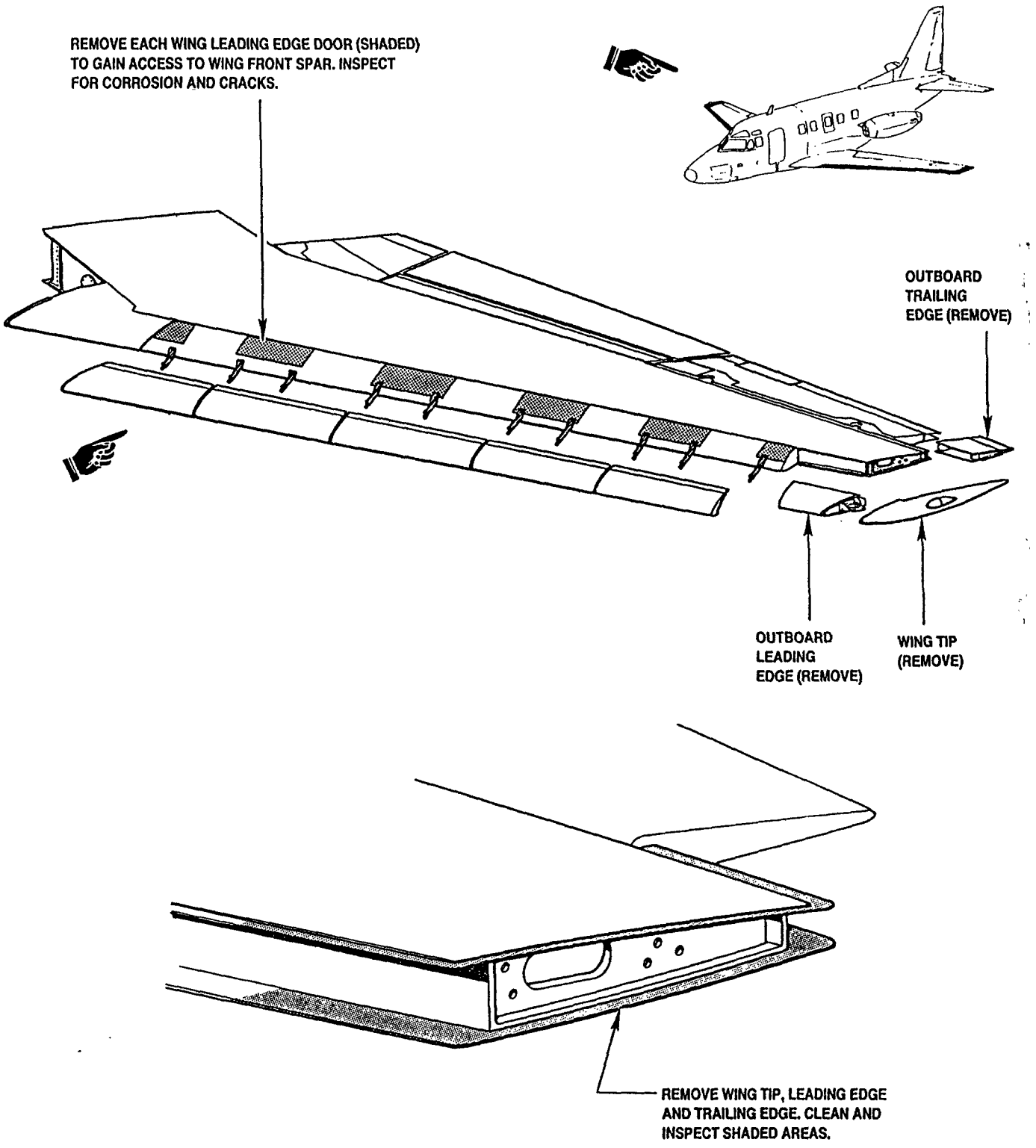
REMOVE EACH WING LEADING EDGE DOOR (SHADED) TO GAIN ACCESS TO WING FRONT SPAR. INSPECT FOR CORROSION AND CRACKS.
OUTBOARD TRAILING EDGE (REMOVE)
OUTBOARD LEADING EDGE (REMOVE)
WING TIP (REMOVE)
REMOVE WING TIP, LEADING EDGE AND TRAILING EDGE. CLEAN AND INSPECT SHADED AREAS.

Figure 57-20-601 — Inspect — Wing Tips, Wing Front Spar, Leading Edge Skin, Rib and Stiffeners

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FLIGHT SURFACES - DESCRIPTION

1. Wing Slats

A set of five slats is provided over the intermediate wing leading edge assembly. Each slat is cantilevered from the leading edge ribs by two track and roller assemblies. The five slats are interconnected and operate as a single aerodynamic unit.

Slat construction consists of ribs, a sheet-metal beam, a honeycomb sandwich trailing edge, and upper and lower skins. Provisions are made for installation of inflatable deicer boots on the upper surface of the slats.

2. Ailerons

The aileron is a single-cell torque box consisting primarily of a sheet-metal beam, transverse ribs, upper and lower skins, and a honeycomb sandwich panel trailing edge. A trim tab of honeycomb sandwich construction is provided on the left aileron.

The aileron is mounted to the wing rear spar by two hinges. Aerodynamic balance is obtained by a series of steel weights attached inside the leading edge skin.

The trouble shooting and maintenance practices for the ailerons are covered in Chapter 27.

3. Flaps

The flap is a two-cell torque box consisting primarily of a spar, a rear beam, and transverse ribs enclosed in upper, lower, and leading edge skins. The rear beam incorporates a sheet-metal channel and honeycomb sandwich trailing edge. The flap operates on two tracks cantilevered from the wing rear spar.

The trouble shooting and maintenance practices for the flaps are covered in Chapter 27.

FLIGHT SURFACES - TROUBLE SHOOTING

1. Trouble Shooting Wing Slats

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. SLATS BIND, STICK, OR DO NOT OPERATE FREELY.		
(1) Slat tracks dirty.	Check "free-fall" of slat assembly by manually pushing slats closed, then releasing them. Push slats perpendicular to leading edge.	Clean all track and roller surfaces and recheck operation.
(2) Slat deformed.	Visually check for damage and alignment.	Replace damaged sections.
(3) Binding at inboard or outboard ends.	Check all clearances. (See figure 57-50-201.)	Maintain clearances given in figure 57-50-201.
(4) Slats improperly adjusted, or loose, on adjustment bushing.	Check slat alignment and forward attachment points.	Adjust slat alignment. (See figure 57-50-201.)
(5) Obstruction in slat track cup.	Check slat track cups for presence of loose hardware or other objects.	Remove foreign material.
(6) Inoperative main load or side load rollers.	Check rollers for freedom of rotation and corrosion.	Remove, clean, and lubricate rollers. (Refer to "Cleaning Slat Tracks and Slat Track Rollers.")
B. ANTIABRASION COMPOUND WORN OFF LOWER SURFACE OF TRAILING EDGE OF SLATS.		
(1) Loose screw or burr on screw directly under slat trailing edge on wing leading edge.	Check screws for proper torque and burrs on edges. Screws should be installed 0.00 to 0.007 inch below surface.	Torque screws or remove burr and reapply antiabrasion compound (Everlube Corp solid-film lubricant or equivalent.)
(2) Slat improperly adjusted.	Check slat alignment. (See figure 57-50-201.)	Adjust slat alignment and replace antiabrasion compound.
(3) Slat deformed.	Visually check for damage and alignment.	Replace damaged sections.

FLIGHT SURFACES - MAINTENANCE PRACTICES

1. Removing Wing Slats

A. The procedures for removing the right and left wing slats are the same. The parts referred to in the following procedure are shown in figure 57-50-201. Proceed as follows:

(1) Remove interconnecting fittings and disconnect deicer boot interconnecting hoses between slat sections.

(2) On left and right No. 2 slat sections, disconnect deicer boot air supply fitting from hose inside wing leading edge.

(3) Remove attaching screws securing slat section to slat track, and remove slat section.

NOTE: The No. 2 slat sections must be rotated to an angle nearly perpendicular to the wing, so that deicer boot air supply fitting can be withdrawn from the wing leading edge.

- Mark position of eccentric adjustment bushings if slats are to be reinstalled, so they will not have to be rerigged. Tape or wire adjustment bushings in place to avoid loss of rigging.

2. Installing and Adjusting Wing Slats

A. For procedures for installing and adjusting wing slats, see figure 57-50-201.

3. Removing Wing Slat Tracks, Rollers, and Supports

A. Each slat track is supported and guided by pairs of roller bearings mounted inside rib assemblies, at two locations within the wing leading edge structure. (See figure 57-50-202.) At each location, the track rides inside two vertical-load rollers and two side-load rollers. Each side-load roller is mounted to a webbed support fitting. The two support fittings and the vertical-load rollers are positioned at each location by two bolts secured to the rib assemblies. Removal procedures are the same for all slat tracks, rollers, and supports, as follows:

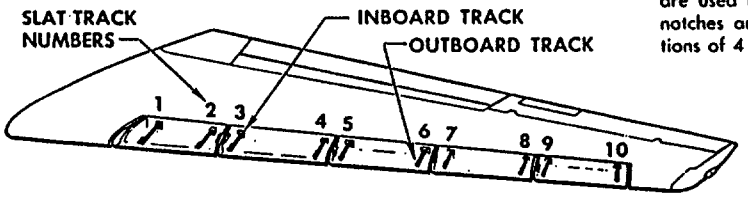
(1) Remove wing slats.

(2) Remove access panel above applicable slat track.

(3) Cut lockwire that safeties upper and lower roller bolts. Remove upper roller bolt, washer, and upper vertical-load roller at aft and forward track-mounting locations within wing leading edge structure.

NOTE The left and right slat installation procedures are the same.

- The object of this procedure is to fair the slat break point and trailing edge with the basic wing contour within tolerances, along the entire slat span. The slat sections should fair, each with the other and with the fixed inboard and outboard leading edges. The slats should extend and retract over the full length of travel, yet not rack or bind.
- The outboard section should be installed first and installation should proceed inboard. Each slat section should be adjusted individually before interconnectors are installed.



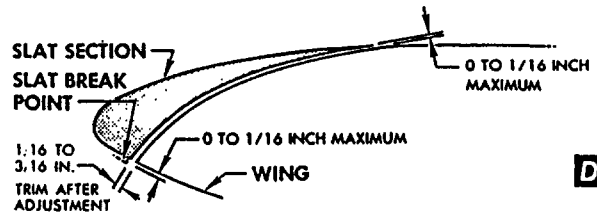
1 Place outboard slat section on tracks and install forward attaching screws and eccentric adjustment bushings only, at this time. Engage notch 1 or 11 with slat track pin for first trial fit. (See detail A.)

NOTE Install attaching screws and eccentric adjustment bushings with flange and screwhead on inboard side of odd-numbered tracks (1, 3, 5, 7, and 9). Install bushings and screws with flange and screwhead on outboard side of even-numbered tracks (2, 4, 6, 8, and 10).

2 Fully close slat section, allowing trailing edge to rest on wing.

3 Check whether slat break point aligns with wing leading edge. If it does not align within 1/16 inch, remove slat section from tracks and rotate forward adjustment bushings to positions necessary to vertically align slat break point with wing leading edge. The slat leading edge skin should not be trimmed at this time.

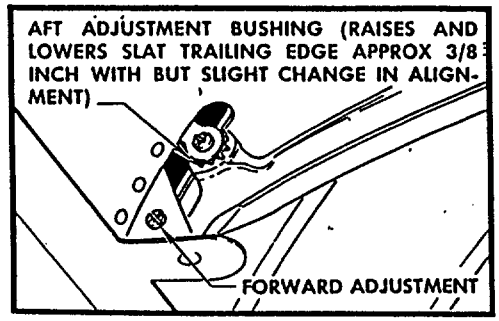
4 Install aft outboard eccentric adjustment bushing in positions necessary to align slat trailing edge with wing as indicated.



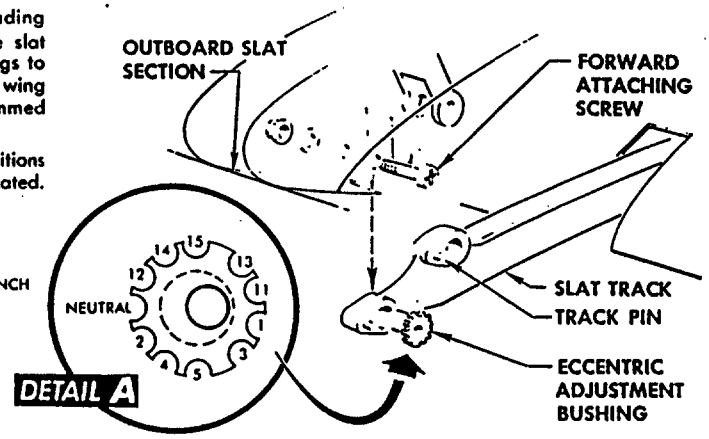
- Slat section not being adjusted should be taped to the closed position.
- Spare slat assemblies furnished have a small amount of excess material on the ends and at the break point on the underside of the leading edge. This excess material is to be trimmed from the slats after they are installed. The excess skin material will not interfere with slat assembly installation.

MOVEMENT OF SLAT BY AFT BUSHING

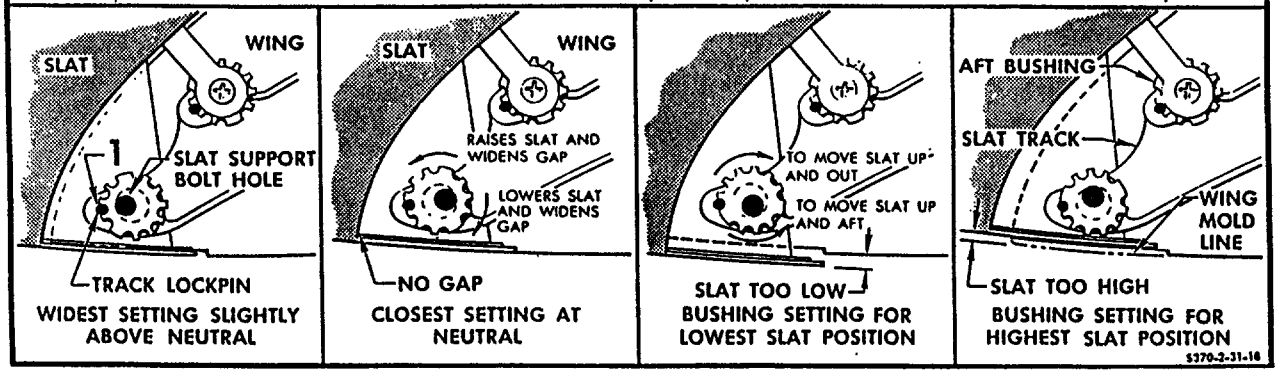
When engaged with the slat track pin, starting with N as neutral, notches 1 through 5 move the bolt hole progressively down and notches 11 through 15 move it progressively up when the notches are used in numerical sequence. Note that progressively numbered notches are not physically adjacent to each other, with the exceptions of 4 and 5 and 14 and 15.


AFT ADJUSTMENT BUSHING (RAISES AND LOWERS SLAT TRAILING EDGE APPROX 3/8 INCH WITH BUT SLIGHT CHANGE IN ALIGNMENT)
FORWARD ADJUSTMENT

NOTE Immediately before installing eccentric adjustment bushings, clean holes with solvent and coat adjustment bushings with wet zinc-chromate primer.



MOVEMENT OF SLAT BY FORWARD BUSHING ADJUSTMENT POSITIONS ROTATE BUSHING TO OBTAIN BEST POSITIONING OF SLAT.
ALL VIEWS LOOKING AT OUTBOARD SIDES OF OUTBOARD TRACKS, LH WING, AND INBOARD SIDES OF INBOARD TRACKS, RH WING



Installing and Adjusting Wing Slats
Figure 57-50-201 (Sheet 1 of 2)

5 Recheck slat break point adjustment given in step 3. Repeat steps 3 and 4 if necessary. The slat section should extend and retract over its full length of travel, yet not rack or bind.

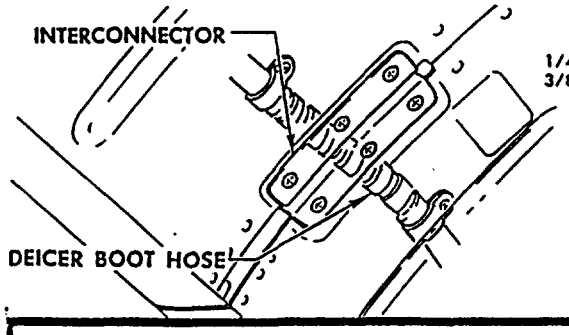
6 With the two forward and the outboard aft bushings rigged and connected, set aft inboard bushing to that notch which most perfectly aligns bushing bolt hole to slat fitting bolt hole and which permits attach bolt to be inserted without requiring force to further align holes. If force is required to align the bolt holes, slat racking or binding may result.

7 Install the remainder of slat sections according to procedures in steps 1 through 5, working from outboard to inboard.

8 Install interconnectors between slat sections. Hand-form (open or close) interconnectors as necessary to obtain installation screw hole alignment.

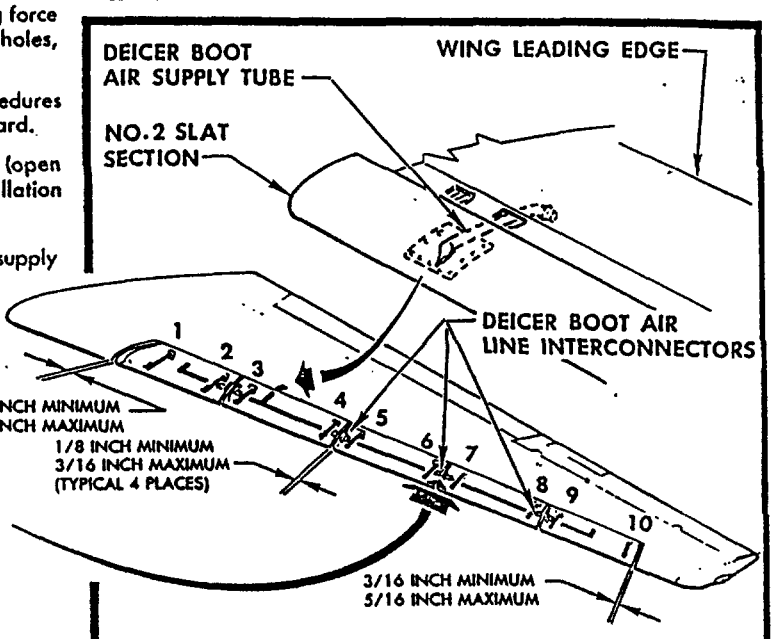
NOTE When installing No. 2 slat section, deicer boot air supply tube must be guided into wing leading edge.

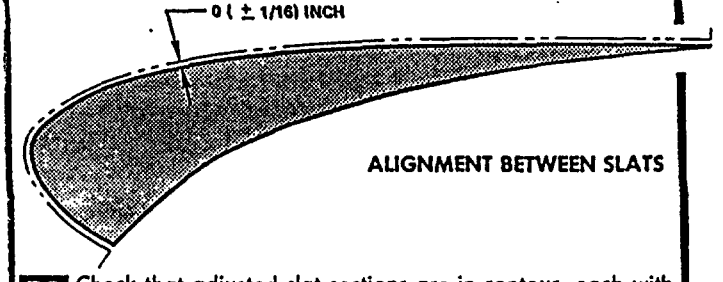
9 Connect deicer boot interconnect hoses.



NOTE Combinations of adjustment bushing locations must be found that align each slat section as noted in steps 3, 4, 9, and 10 and that permit slats to extend and retract over their full length of travel, yet not rack or bind.

12 Torque all attaching screws to standard values.



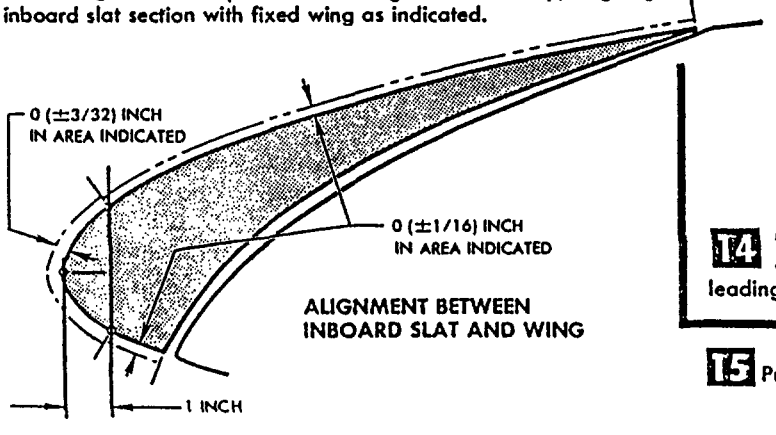


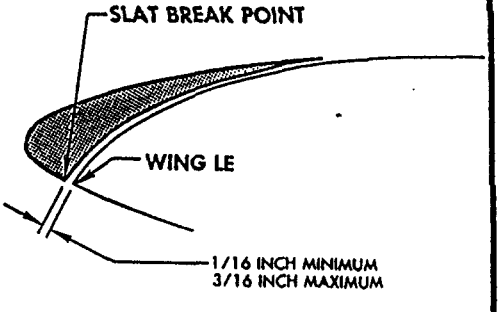
10 Check that adjusted slat sections are in contour, each with the other. Rotate adjustment bushings as necessary, so that the contour of each slat section is in alignment, each with the the other, as indicated.

13 Retract slats; then check clearances between slat sections at their leading edges only. Also check clearances between outboard slat section and wing and between inboard slat section and wing. Do not check clearances at trailing edges. Trim excess material from slat leading edge ends.

NOTE The slats have honeycomb trailing edges and are pretrimmed to the correct dimension at the time of manufacture and do not require additional trimming.

11 Check that inboard slat section is in contour with the fixed wing. Rotate adjustment bushings as necessary, aligning inboard slat section with fixed wing as indicated.





14 Check clearances along slat break point between slat and wing leading edge. Trim excess material from slat leading edge skin.

15 Proceed according to instructions in "Checking Wing Slats."

S370-2-31-19A

(4) Remove aft support fittings and lower vertical-load aft roller while withdrawing lower aft bolt from rib assemblies.

CAUTION: EXERCISE CARE TO PREVENT ROLLER, SUPPORTS, AND TRACK FROM FALLING DOWN UNSUPPORTED.

(5) Withdraw slat track from wing leading edge structure.

(6) Remove forward support fittings and lower vertical-load forward roller while withdrawing lower forward bolt from rib assemblies.

(7) If necessary, remove side-load roller from applicable support fitting by removing bolt, washer, and nut that secure roller to web of support fitting.

4. Installing Wing Slat Tracks, Rollers, and Supports

A. For procedures for installing wing slat tracks, rollers, and supports, see figure 57-50-202.

5. Applying Plastic Abrasion-resistant Coating to Wing Slats

A. All areas of the slat and wing leading edge which contact each other during operation are protected from wear by an abrasion-resistant plastic coating. Areas to be coated are as follows:

(1) Coat length of trailing edge of slat from 1/2 inch in on top side to 2 inches in on bottom.

(2) Coat a 2-inch strip along wing leading edge where slat trailing edge mates when slats are retracted.

(3) Coat a 3/8-inch strip around inside perimeter of slat track close-out tabs at slat track adjustment points.

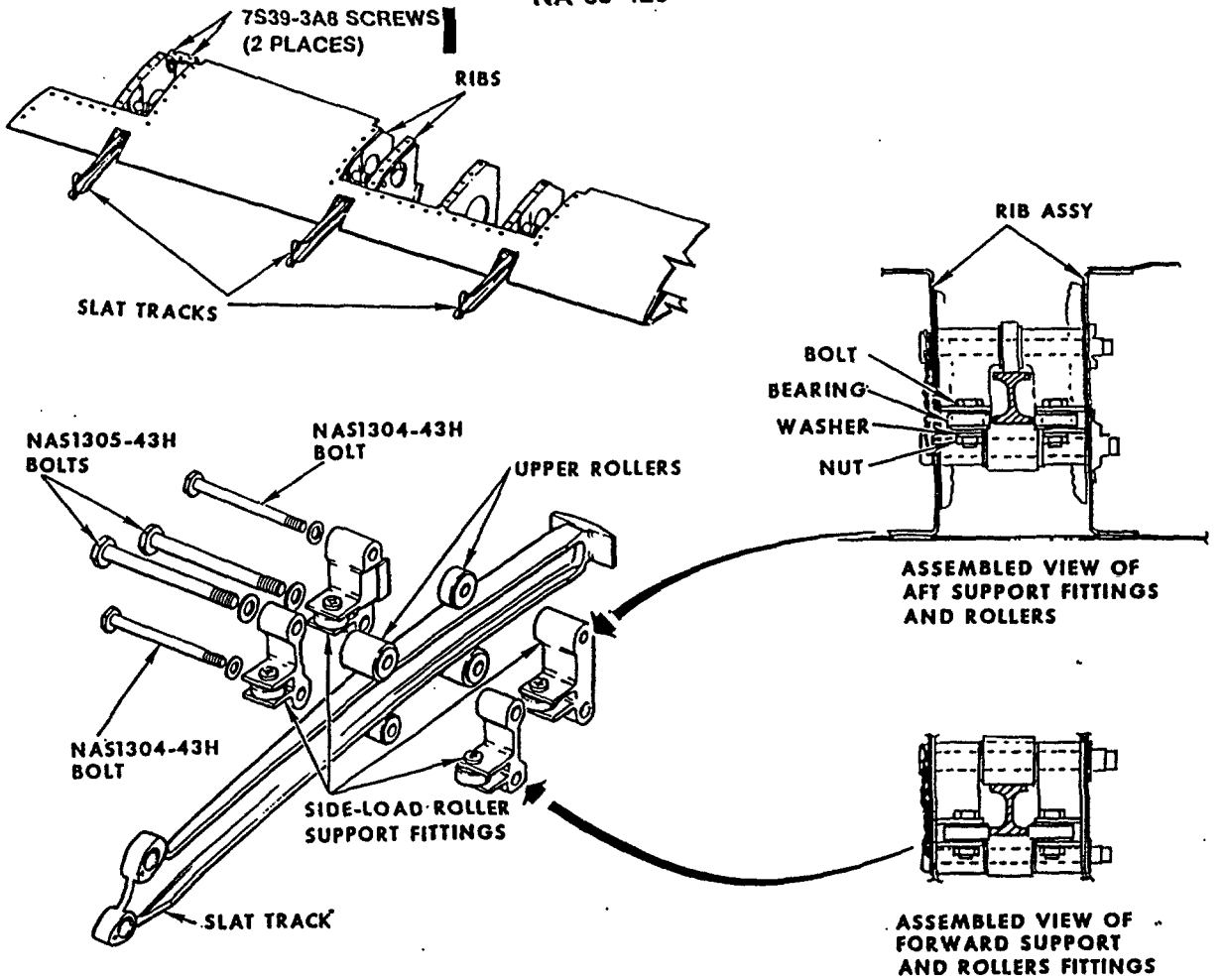
(4) Coat slat bumper plates on end of slats.

B. Make sure areas are free of paint, primer, and grease.

C. Clean areas with a cloth saturated with Naphtha (Federal Specification TT-N-95). Then apply one coat of wash primer (Specification MIL-C-8514.)

D. Mix catalyst with antiabrasion compound (Everlube Corp solid-film lubricant or equivalent) as directed.

E. Apply a 2-1/2 to 3-mil (0.0025- to 0.003-inch) film to areas described in step A. The coating may be applied by spraying or brushing.



1 If side-load rollers are being replaced, install roller bearing in support fitting web.

2 Install forward support fittings and lower vertical-load forward roller between rib assemblies with one NAS1304-43H bolt and one LD153-0002-1204 washer. Do not tighten bolt.

3 Install aft support fittings and upper vertical-load aft roller between rib assemblies with one NAS1304-43H bolt and one LD153-0002-1204 washer. Do not tighten bolt.

4 Position slat track in wing leading edge structure, over lower forward roller and under upper aft roller. Position upper forward roller between forward support fittings and over track, and install one NAS1305-43 bolt and one 2W18-516M washer.

5 Tighten upper forward bolt (NAS1305-43H) to 100-140 inch-pounds. Tighten lower forward bolt (NAS1304-43H) to 50-70 inch-pounds. Safety with MS20995F41 lockwire.

Caution Before tightening bolts, make sure track is properly positioned between side-load rollers in support fittings, and track stop is beyond aft fittings.

6 Position remaining lower aft roller between aft support fittings and under track, and install one NAS1305-43H bolt and one 2W18-516M washer.

7 Tighten upper aft bolt (NAS1304-43H) to 50-70 inch-pounds. Tighten lower aft bolt (NAS1305-43H) to 100-140 inch-pounds. Safety with MS20995F41 lockwire.

8 Check track for smooth sliding and roller action.

Caution Ensure P/N 7S39-3A8 screws are properly installed on either side of the inboard leading edge hoist point. Use of 7S39-3A3 screws may result in screws coming loose and being ingested into the engine.

9 Install access panel.

10 Install wing slats.

5265 2 52 49 A

Installing Wing Slat Tracks, Rollers, and Supports
Figure 57-50-202

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F. Allow 6 to 16 hours air-drying time before handling, and 7 days for hardening. (The film should resist a fingernail test.)

NOTE: Areas may be air-dried one hour and baked 10 minutes at 250°F as an alternate method of drying.

6. Checking Wing Slats

Before check-out, the wing slats must be properly installed and adjusted. (See figure 57-50-201.) If slats do not meet check-out requirements, refer to 57-50-1.

A. Ground-check wing slats by manually applying pressure at lower leading edge of each slat section, so that tracks ride on upper forward and lower aft rollers.

B. With slats held in this position, push slats to full-retract position and then to extend position. Check freedom of rollers, and check that operating characteristics of slats on one wing are the same as those on the other wing. If slats do not seem to operate freely enough, check each track station carefully to see that there is no interference with structure or any other items. Visually check that antiabrasion compound, on lower surface of trailing edge of slats and on wing leading edge under trailing edge of slats, has not been rubbed away at any point.

NOTE: When slats are completely adjusted with all the interconnectors installed and the central sections held fully closed, the unsupported end sections are partially open. The extent of this opening is not critical. However, it should be possible to fully close the partially open sections without mechanical binding.

- When slats are completely adjusted with all interconnectors installed, it is possible and normal that one or more slat sections are not bottomed out on the tracks. This condition is not critical. However, it should be possible to bottom the sections without mechanical binding.

7. Cleaning Slat Tracks and Slat Track Rollers

Cleanness of the slat tracks will greatly increase the service life of the slat track rollers. If the slat tracks and/or rollers require cleaning, use a cloth dampened in solvent (Federal Specification P-D-680) such as Stoddard solvent, mineral spirits, dry-cleaning solvent, or equivalent. Pressure spraying of the solvent should only be used in severe cases of roller binding when mission requirements will not permit use of normal cleaning procedures. Pressure spraying with solvent tends to wash the lubricant out of the roller bearings. Severely binding rollers, and rollers which have been cleaned with a pressure spray, should be replaced, or cleaned and relubricated as soon as possible, using the following procedures:

A. Remove rollers. (Refer to "Removing Wing Slat Tracks, Rollers, and Supports.")

WARNING: SOLVENTS USED IN THIS PROCEDURE ARE TOXIC AND MAY BE INJURIOUS TO PERSONNEL. USE SOLVENTS IN A WELL-VENTILATED AREA AND WEAR RUBBER OR PLASTIC GLOVES.

B. Suspend rollers vertically in a container of chlorinated solvent, such as trichlorethylene or 1,1,1 trichloroethane (methyl chloroform), for 10 minutes.

C. Move roller rapidly up and down in solvent for 2 to 3 minutes; then using clean solvent, rotate roller in solvent for one to 2 minutes.

D. Slosh roller in clean solvent for another 2 to 3 minutes; then dry thoroughly.

E. Lubricate cleaned rollers. (See figure 57-50-203.)

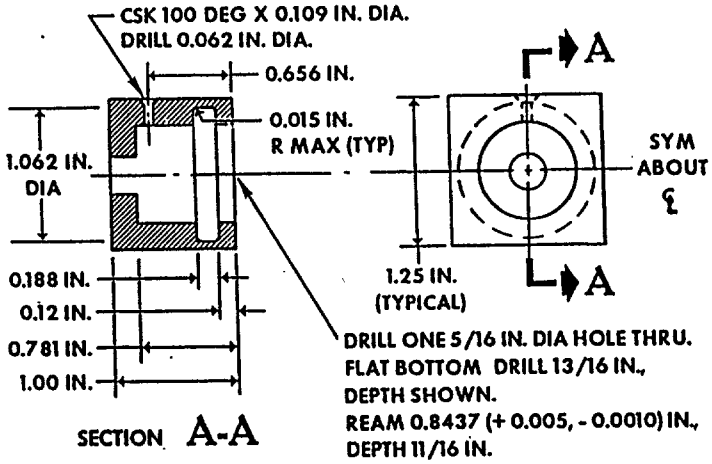
F. Install rollers. (Refer to "Installing Wing Slat Tracks, Rollers, and Supports.")

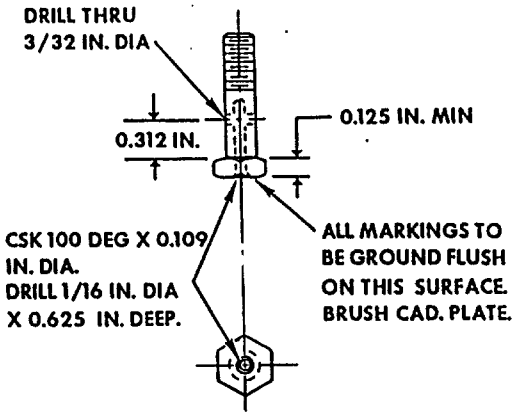
8. Checking Slat Tracks for Wear

Slat tracks having grooves or wear through the nickel plating but less than a total depth of 0.010 inch should be replated as a protection against corrosion. Tracks having grooves or wear exceeding 0.010 inch in depth should be replaced.

NOTE This procedure requires the use of a locally manufactured tool to accomplish lubrication of sealed slat track rollers.

1 Fabricate special lubrication tools as shown.





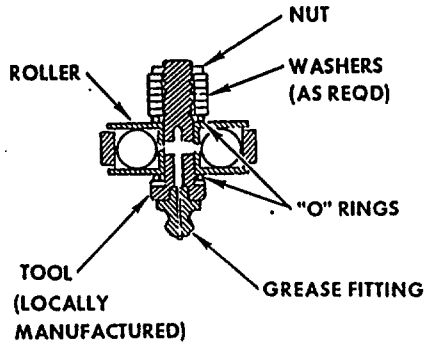
PARTS LIST
2W18-516M WASHER (2 REQD) MS28775-011 "O" RING (1 REQD) MS28775-211 "O" RING (1 REQD) MS21042L5 NUT (1 REQD) AN5-15A BOLT (1 REQD) MS15001-1 FITTING (1 REQD) NAS1515D5 WASHER (1 REQD)
Make from 2024-T42 Al extr bar or Al extr shape, Fed. Spec QQ-A-200/3A, or 2024-T4 Al rolled bar, Fed. Spec QQ-A-225/6A. No finish is required.

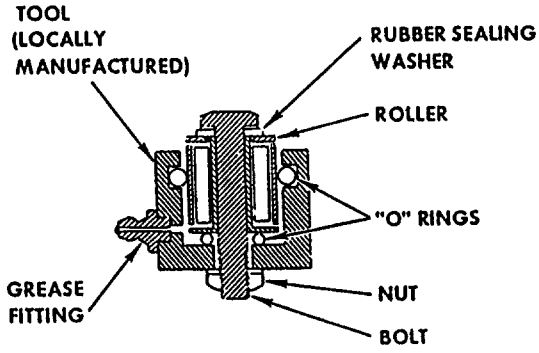
For roller Part No. AA-17407 (Torrington) and YR-137-CP-X or YR213 (McGill)

PARTS LIST
2W18-416 WASHER (2 REQD) MS28775-010 "O" RING (2 REQD) MS21042L4 NUT (1 REQD) MS15001-1 FITTING (1 REQD)
Make from AN4-10A bolt* (1 reqd)

For roller Part No. 4NBF614YJ or 3NBF512YJ (Torrington and McGill)

* IF AN4-10A BOLT IS USED, USE OF WASHERS WILL BE NECESSARY TO ACCOMMODATE ROLLER; HOWEVER, A BOLT OF SHORTER LENGTH (AS SMALL AS AN4-6A) MAY BE USED WITH FEWER WASHERS.





NOTE Similar tools can be made for other rollers.

2 Install roller in tool and lubricate with any one of the following greases:

- Supermil Grease No. 72832, American Oil Co
- Cosmolube 678, E. F. Houghton and Co
- Royco 27, Royal Lubricants Co
- Aeroshell Grease 7A, Shell Oil Co
- Mobilgrease 27, Socony Mobil Co, Inc
- Uni Temp EP, Texaco, Inc
- Sinclair 3827 Grease, Sinclair Refining Co
- Ped-3527, Standard Oil of Calif.
- Castrolease A1, Castrol Oils, Inc

5370-2-31-21

9. Inspection and Repair of Inboard Slat, Inboard Bumper

A. Inspect and repair inboard slat, inboard bumper as follows:

(1) Inspect inboard edge of inboard slat for apparent movement or looseness of attached bumper. If this condition exists, proceed to step 2.

(2) Drill out two attaching rivets with a 5/32 (#22) dia. drill.

NOTE: Remove the rivet upset trails thru the rib corner relief hole.

(3) Remove the inboard bumper and shim.

(4) Inspect inboard bumper and shim for cracks, corrosion, and hole diameters for distortion. Hole diameter requirement range .160-.164 inch.

(5) If hole diameters fall within the required range, reinstall the bumper and shim using two (2) each of either of the following fasteners:

CR 2162-5-3 (NAS1399D-5-3)
CR 3212-5-3

NOTE: If hole diameters in either the bumper or shim exceed .164 inch, proceed with step 6.

(6) Drill a #16 (.176-.180) diameter hole in each part and attaching point on the slat.

(7) Install the bumper and shim using two (2) of either of the following fasteners:

CR 2248-5-3 (NAS1739B-5-3)
CR 3242-5-3

(8) Secure airplane as desired.

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CHAPTER

71

POWERPLANT

NOTICE

The following information applies to NA 265-70 airplanes only.

CHAPTER 71

POWER PLANT — GENERAL

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POWER PLANT — GENERAL

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* The asterisk indicates pages revised, added, or deleted by the current revision.

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POWER PLANT

DESCRIPTION

GENERAL

This airplane is powered by two pod-mounted JT12A-8 engines. The engine is a continuous-flow, gas turbine power plant incorporating a single-rotor, axial-flow compressor; an eight-unit can-anular combustion section; a two-stage reaction turbine; and an exhaust pipe with a fixed nozzle. The power take-off section is waist-mounted. No units are located in the engine nose section. The accessories mounted on the engine by the engine manufacturer are a fuel control, a fuel pump, a fuel filter, a fuel heater, a fuel oil-cooler, an oil tank, ignition units, a power control cross-shaft, and a fuel pressurizing and dump valve. Additional equipment added by the airframe manufacturer at the time of engine installation is the starter generator, engine electrical harnesses, fuel flowmeter tachometer generator, oil pressure transmitter, low oil-pressure warning switch, transverse firewall, tail pipe thrust reverser assembly, and engine air inlet pod assembly. The right and left engines are identical in build-up, and can be installed in either engine pod by installing the throttle connection to the left or right inboard side of the cross-shaft, as required.

The multistage axial-flow compressor, of moderate compression ratio, consists of a nine-stage rotor and eight stator stages. The rotating parts of the compressor are connected to the first- and second-stage turbines by the turbine shaft and shaft lock. The engine has an anti-icing air system to prevent icing of the compressor inlet surfaces. Compressor discharge air is directed into the hollow compressor inlet guide vanes for this purpose. High-pressure air bled from the diffuser section, directly behind the compressor section, is also used for anti-icing and other airframe functions. The combustion section, which houses the combustion chambers and fuel manifolds, is to the rear of the diffuser section. Eight separate combustion chambers are arranged annularly (can-annular) and are connected by flame tubes. High-energy ignition units (continuous ignition) are connected to spark igniters installed in combustion chambers No. 3 and 6. The engine fuel system filters, pressurizes, meters, and delivers the fuel to the fuel nozzles in the burner cans where it is sprayed and consumed.

The turbine section is to the rear of the combustion section. The turbine case houses the turbine rotor assembly. The turbine exhaust case is fastened to the rear flange of the turbine case, and contains the exhaust cone and strut assembly. The engine exhaust nozzle is attached to the rear flange of the turbine exhaust case.

The engine is supported within the engine pod at three points: the left and right sides of the engine diffuser case and the top of the front flange of the compressor housing.

ENGINE DRAIN LINES

The drain lines from the engine components and systems are routed to a central drain manifold attached to the lower side of the engine, forward of the engine firewall. The central drain manifold has a short pylon which extends downward, through the outside mold line of the lower engine access door. The four drain lines terminating at the drain manifold, from forward to aft in order, are as follows:

1. Oil tank scupper drain line
2. Pressurizing and dump valve drain line (line capped on airplanes changed by S/B 74-5)
3. Oil system breather
4. Combustion chamber drain line

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POWER PLANT GENERAL - TROUBLE SHOOTING

1. Trouble Shooting Engines

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. NO RPM WHEN ATTEMPT IS MADE TO START ENGINE.		
(1) No electrical power to starter system.	Check source of electrical power.	
(2) Engine is "frozen."	Attempt to rotate by hand.	Replace engine.
(3) Starter drive shaft in component drive gearbox sheared.	If sounds of starter rotation are heard during attempted start, but engine does not rotate, shaft is sheared.	Replace engine.
(4) Starter shaft sheared.	Starter rotates, but does not rotate engine; starter shaft is sheared.	Replace starter-generator.
B. INSUFFICIENT RPM WHEN ATTEMPT IS MADE TO START ENGINE.		
(1) Excessive turbine and compressor blade rubbing.	Listen for loud scraping noise while engine is rotating.	Replace engine.
(2) Defective starter system.	Check tachometer for maximum speed at which starter will rotate engine.	Replace starter.
(3) Inadequate voltage for starter.	If engine starts, but fails to accelerate because of lack of help from the starter, auxiliary power unit is inadequate.	Use large capacity auxiliary power unit.
C. COMBUSTION DOES NOT OCCUR WHEN ATTEMPT IS MADE TO START ENGINE.		
(1) No fuel to engine.	Check fuel supply and position of fuel shutoff valve.	
(2) Defective ignition system.	Turn on system and listen for ignition noise.	Refer to Chapter 74.
(3) Defective engine fuel system.	Check fuel flow indicator for minimum starting flow.	Refer to Chapter 73.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
(4) Defective starter system.	Check tachometer for maximum speed at which starter will rotate engine.	Replace starter.
(5) Throttle linkage improperly adjusted.		Adjust throttle linkage.
(6) Faulty cutoff valve in fuel control.	If there is fuel from airplane fuel system and throttle linkage is rigged correctly, cutoff valve in fuel control is not opening.	Replace fuel control.
(7) Defective engine-driven fuel pump.	If fuel flowmeter indicates some fuel flow, but not enough for normal start, pump may be defective.	Replace engine fuel pump.
D. ENGINE RPM AND TURBINE EXHAUST TEMPERATURE FLUCTUATES.		
(1) Defective engine fuel system.		Refer to Chapter 73.
(2) Defective instruments and/or instrument circuits.	Use instruments with units of known accuracy and check engine operation.	Replace faulty instruments.
E. ENGINE ROUGHNESS OCCURS.		
(1) Internal interference between rotors and stators.	Check for scraping noise as engine slows down or at shutdown. Check oil strainer for metal particles.	Replace engine.
(2) Defective accessories.	Check for unusual noises coming from accessories, or improperly functioning system.	Replace defective accessories.
F. ENGINE SURGES DURING ACCELERATION.		
(1) Defective compressor bleed actuator.	Perform operational check of engine, noting compressor bleed valve and actuator operation.	Replace fuel control.
(2) Defective fuel control unit.	If bleed system is satisfactory, fuel control unit is faulty.	Replace fuel control.

POWER PLANT GENERAL

MAINTENANCE PRACTICES

PREVENTION OF FOREIGN-OBJECT DAMAGE

To prevent foreign-object damage to the engine, the engine inlet duct cover and the engine exhaust cover should be installed at all times during non-maintenance periods. The engine inlet duct and tail pipe must be covered during movement of the airplane form one area to another. The engine air inlet duct must be inspected for foreign objects, loose fasteners, ect, after the airplane is moved. Under no circumstances should the inlet duct or the inlet guide vanes be used as a resting place for tools, parts, nuts and bolts, etc, during maintenance operations. An inventory of all tools, parts, and equipment must be conducted after each engine maintenance operation is completed.

INSPECT POWER PLANT

To inspect power plant, proceed as follows:

1. Remove upper and lower access doors.
2. Inlet guide vanes for nicks, dents, cracks, distortion, FOD, corrosion and cleanliness.
3. First stage blades for nicks, dents, cracks, distortion, discoloration, FOD and corrosion.
4. Inboard and outboard engine mounts for broken lockwire, proper cotter pin installation, corrosion security.
5. Forward engine mount installation for security of adapter, adapter bolts and nuts. Excess free play.
6. Engine heat and vent manifold for cracks and attaching points and connections for evidence of hot air leakage.
7. Anti-icing holes for obstructions.
8. Compressor inlet case for pin holes, pitting and porosity. Voids and coarse pitted surface to any degree on aluminum braze material is acceptable, provided they exist in fillets but do not extent into joints.
9. Inlet pressure sensing tube and probe for damage and security.
10. Engine exterior for cleanliness, evidence of fuel and oil leakage; loose or missing rivets, bolts, studs and clamps; and proper lockwiring.
11. Fuel flow transmitter for evidences of leakage and inspect support bracket for cracks.
12. Fuel differential pressure switches and mounts on nacelle nose for damage and security.
13. Fuel oil cooler for evidence of leakage.
14. Oil tank and lines for dents, cracks, leakage and security.
15. Rotate compressor in direction of normal rotation at least one full turn and check for free rotation.
16. Diffuser case for bulged or discolored areas and for cracks at exciter mount and igniter bosses.
17. Accessory and component drives gearbox for evidence of leakage, external cracks and security.
18. Gearbox fuel, oil, and drain lines for evidence of leakage, chafing, loose connections, proper positioning and security.
19. Gearbox drain line outlets for obstructions.
20. Fuel heater for evidence of leaks and security.
21. Oil pressure transmitter and attached lines and electrical leads for deterioration and security.
22. Tachometer generator for damage and electrical leads for deterioration and security.
23. Fuel head shutoff valve and electrical leads for deterioration and security.
24. Fuel control for leakage and security. Fuel control pressure lines for security, routing, chafing and signs of leakage.
25. Fuel control mechanical linkage for wear and security.
26. Fuel control linkage cross shaft bearings for wear.
27. Starter-generator flexible inlet duct and air inlet housing for cuts, tears and security.
28. Starter-generator terminal block for cracks, damaged threads and broken insulation. Rubber hood for deterioration and electrical connector for security.
29. Starter-generator overheat switch for security.
30. Lower air inlet duct baffles for cracks and proper position.
31. Exhaust cone struts for cracks, damage and supporting bolts for security and applicable lockwire.

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MAINTENANCE PRACTICES (Continued)

INSPECT THRUST REVERSER AS FOLLOWS:

1. Inspect the following thrust reverser components for security, cracks, hot spots, warping, loose fasteners and applicable safeties.
2. Upper and lower doors.
3. Reverser door drive links and mounting bolts.
4. Bogies and bogie guide rods.
5. Door striker plates, stop bolts, hatch hooks and springs.
6. Interior and exterior surface of exhaust nozzle and reverser doors.
7. Exhaust nozzle longerons.
8. Reverser door attach fittings and bolts.
9. Inboard and outboard stang (fairing) brackets.
10. Exhaust nozzle ring segments (mice).
11. Forward inboard stang bracket assembly (Rohr P/N 140-5421-17).
12. Visible portion of upper and lower reverser door longerons (Rohr P/N 140-5437-1, -2 and 140-5438-1 and -2).
13. Outboard reverser door support fitting (Rohr P/N 140-5439-1).
14. Inboard reverser door support fitting (Rohr P/N 140-5440-1).
15. Hydraulic actuator, fittings, check valves and connecting plumbing for leaks, chafing and security.
16. Secondary latch solenoid, stow and deploy switches and electrical wiring for security, chafing, cleanliness and safeties.

NOTE

Steps 17 and 18 are applicable to thrust reversers with the 140-5401-501 and -502 nozzle assembly installed.

17. Fittings and pins at forward end of longeron for wear, cracks and distortion.
18. Hydraulic line support brackets for loose fasteners and cracks.

ENGINE EXHAUST FLANGE INSPECTION AS FOLLOWS:

1. Inspect exhaust case rear flange of engines with tailpipe clamp, or bolt and nut installation for:
   A. Cracks
   B. Security
2. Thrust reverser installation — Inspect exhaust case rear flange for:
   A. Cracks
   B. Security

NOTE

Rework cracked exhaust case rear flange in accordance with the Pratt and Whitney Service Bulletin No. 1798, Rev. 4.

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1. Removal of Engine

A. The following equipment will be necessary to remove the engine and support it after removal:

(1) Engine hoist adapter (265-810039) or a suitable crane.

(2) Engine lifting sling (265-810030).

(3) Bomb hoist, Aero Model 14-C.

(4) Engine transportation stand or dolly (265-810063).

B. When necessary equipment is available, proceed as follows:

(1) Remove cowling from engine and applicable pylon access panels. Close fuel shut-off valve.

(2) Turn electrical power off and dump hydraulic system accumulators.

(3) Disconnect the following:

(a) Heat and vent duck at engine manifold.

(b) Fire extinguisher line.

(c) Fuel supply line.

(d) Total discharge pressure line (Pt 5) at engine.

(e) Thrust reverser electrical plug.

(f) Pylon firewall-to-engine thermocouple lead. Retain clamp for reinstallation.

(4) Disconnect thrust reverser flexible hoses as follows:

(a) Remove clamps supporting thrust reverser hydraulic line restrictors (see figure 71-00-401).

(b) Remove safety wire from both ends of both firesleeves.

(c) Remove heat reflective tape (if necessary) to allow the firesleeves to be moved.

(d) Slide the firesleeves back as required to uncover the flexible hoses to restrictor valve "B" nuts.

(e) Using two wrenches (one on restrictor) loosen "B" nuts. Disconnect both hydraulic flexible hoses from restrictor valves. Cap restrictors and flexible hoses as required.

(5) Remove starter-generator electrical leads from terminal block. Identify leads for reinstallation. On left engine, remove clamps securing leads to aft bulkhead of pod nose and at engine cross shaft bracket.

(6) Open "FUEL FLOW" circuit breaker in cockpit.

CAUTION: IF AC POWER IS APPLIED TO THE FUEL FLOW INDICATOR CIRCUIT WHEN THE FUEL FLOW TRANSMITTER IS DISCONNECTED, THE FUEL FLOW INDICATOR AMPLIFIER WILL BE DAMAGED.

(7) Disconnect right and left electrical leads.

(8) Disconnect throttle linkage at engine cross-shaft arm.

(9) Attach hoist adapter to fuselage over engine pod and use bomb hoist (Aero Model 14-C) in conjunction with engine hoist or position a suitable crane over engine.

(10) Attach lifting sling to engine and, using a bomb hoist or crane, take engine weight off trunnions.

(11) Remove lockwire and loosen left and right engine mount support strap nuts. Swing "T" bolts free and open support straps.

(12) Remove cotter pin, nut, washer, and lock from engine forward steady mount shaft and remove shaft from forward engine flange.

(13) Lower engine down and back to clear wing and lower engine into engine dolly.

(14) Remove engine lifting sling.

(15) Cover all openings on engine.

(16) Inspect engine well electrical wiring, plumbing, and structure for cleanliness condition and security.

(17) Inspect yoke and pylon structure for cracks, buckles, dents, corrosion, and loose rivets.

(18) Inspect main inboard and outboard mount strap "T" bolt and nut for worn or damaged threads.

(19) Inspect engine trunnions and flex mounts for corrosion, damage, and cracks.

(20) Inspect steady engine mount attach bolt and nut for worn or damaged threads.

(21) Inspect engine compartment firewall for cracks and cleanliness; seal for cuts, tears, and deterioration.

2. Installation of Engine

A. To install engine, proceed as follows:

(1) Support engine as required and install aft engine mount trunnions on engine. Install four bolts (306-329010) and torque to 100-140 inch-pounds. Lockwire bolts.

(2) Make sure all throttle linkages and all lines and electrical wiring are safely stowed to sides of engine.

(3) Install forward steady mount and retaining shaft. Torque retaining bolt to 100-140 inch-pounds and safety with cotter pin.

NOTE: Make sure nut tightens on engine flange and not on bolt shoulder. Use washers as necessary under bolthead only.

(4) Position engine under engine pod.

(5) Attach engine sling to engine.

CAUTION: TO PREVENT POSSIBLE DAMAGE TO ENGINE COMBUSTION CHAMBER CASE, DO NOT LIFT ENGINE STAND WITH ENGINE.

(6) Using hoist and hoist adapter or crane, take engine weight off dolly mount lockpins.

(7) Raise engine into position in pod.

(8) Position engine trunnion mounts into pod supports and lock with support straps. Torque "T" bolts to 50-70 inch-pounds and lockwire.

(9) Align forward mount with pod attach point and install retaining shaft with washer through support beam and eccentric. Install lock washer and nut on shaft. Do not torque until completion of engine alignment.

(10) Relieve tension on hoist and disconnect engine sling.

(11) Temporarily install engine pod doors and check engine alignment. Align engine in pod by adjusting outboard trunnion and forward steady-support mount.

(12) Connect heat and vent duct to engine bleed manifold at top of engine.

(13) Connect fuel supply line. Tighten steel "B" nut to 900 to 1100 inch-pounds or aluminum "B" nuts to 650 to 800 inch pounds torque.

(14) Connect thermocouple lead from pylon fire wall to engine.

(15) Connect starter generator electrical leads. On left engine install clamps securing leads to aft bulkhead of pod nose assembly, and at engine cross-shaft bracket. Ensure that leads are installed properly. Torque 3/16" stud nut to 20-30 inch-pounds and 3/8" stud to 120-130 inch-pounds.

(16) Connect throttle linkage at engine cross-shaft arm and safety.

(17) Connect engine total discharge pressure line (Pt 5) from pylon fire wall to engine.

(18) Connect fire extinguisher line.

(19) Connect thrust reverser electrical plug.

(20) Connect thrust reverser flexible hoses to restrictors. Torque the 1/4" "B" nut to 135-185 inch-pounds and the 3/8" "B" nut to 270-345 inch-pounds. Do not install firesleeves and reflective tape at this time.

CAUTION: DO NOT ALLOW FLEXIBLE HOSE TO TWIST WHEN TORQUING "B" NUTS.

(21) Connect and safety electrical leads.

(22) Rig engine controls as required.

(23) Service engine oil system.

(24) Perform Ground Operational Check-out of thrust reverser system.

(25) Push in FUEL FLOW circuit breaker.

(26) Accomplish preoperational check of newly installed engine if required.

(27) Perform engine run-up, thrust reverser check, and engine trim check.

CAUTION: DO NOT RUN ENGINE AT TAKEOFF THRUST WITH THE ENGINE COWLING REMOVED.

(28) After engine run make final check of all hoses, cables, control rods, and electrical leads for leaks, chafing, and proper safety.

(29) Install ground maintenance locks on thrust reversers.

(30) Install firesleeves over thrust reverser restrictors as follows:

(a) Slide firesleeves over restrictors and flexible thrust reverser hydraulic hose "B" nuts.

(b) Wrap heat reflective tape over firesleeve if required. Each wrap should overlap by 1/4" with three full wraps at each end of firesleeve.

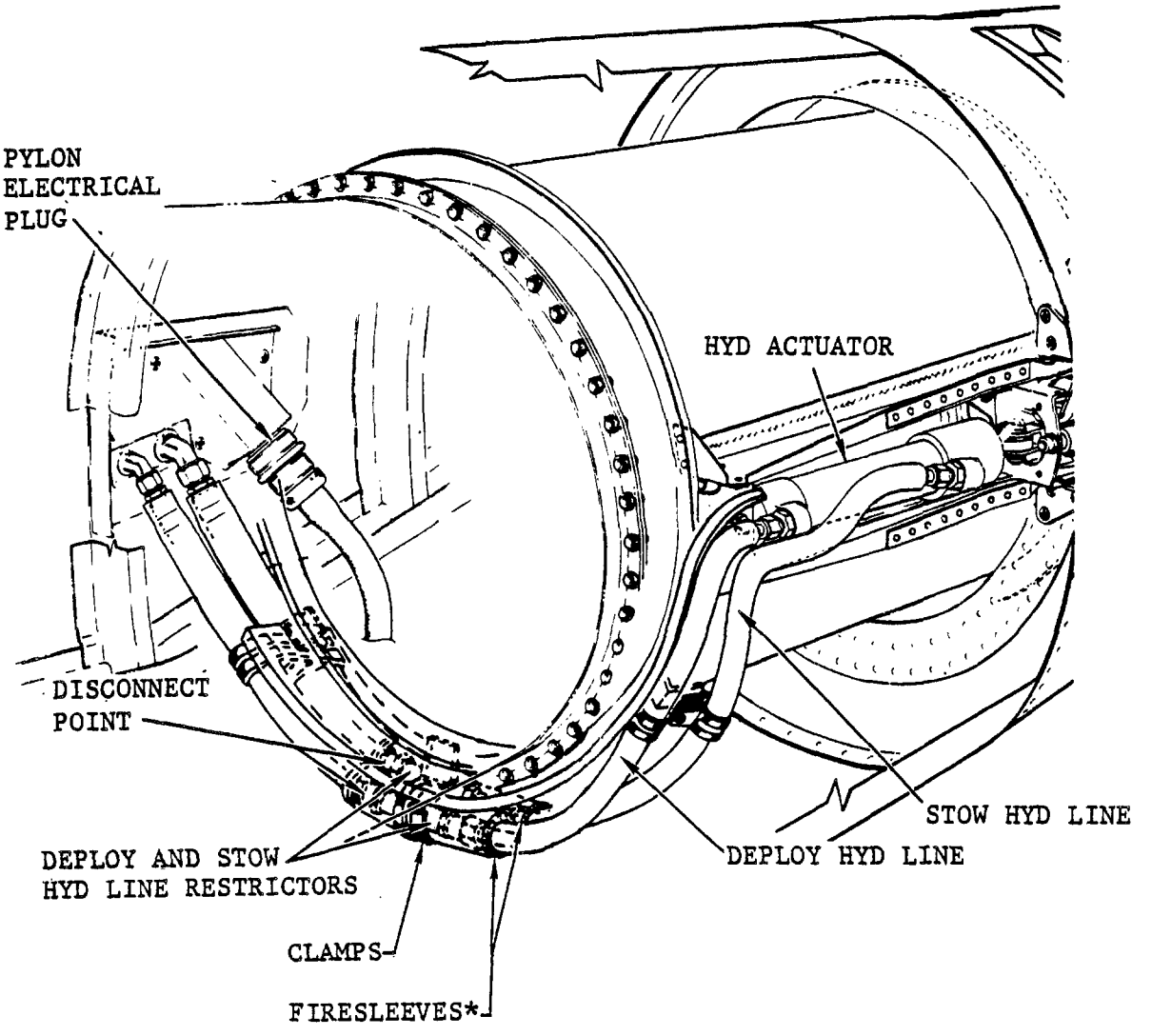
(c) Install safety wire at both ends of firesleeves. The ends of reflective tape should be under the safety wire.

(d) Install clamps securing restrictors to engine.

(31) Inspect applicable engine screens as required. Perform leak check as required.

(32) Install engine access doors.

NOTE: If engine alignment requires additional adjustment after final installation of engine access doors, check engine power lever rigging after accomplishing engine alignment procedures.



*P/N AE102-14 AEROQUIP CORP (APPROX. 5 INCHES EACH)
REFLECTIVE TAPE P/N PE100, MAY
BE PROCURED FROM PERMACEL
CORP. BRUNSWICK, N.J.

Thrust Reverser Flexible Hose Connections
Figure 71-00-401

MAINTENANCE PRACTICES (Continued)

ALIGNING ENGINE

When the engine is being aligned, the trailing edge of the pod doors should be used as a reference mold line.

1. Measure distance between thrust reverser doors (targets) and trailing edge of pod doors. Clearance should be within 1/4 inch of the trailing edge mold line, with a gap of 1/32 inch minimum to 3/16 inch maximum between the butting edges.

NOTE

Deleted.

2. Remove pod doors. Support engine with a suitable hoist and engine lifting sling.
3. Remove lockwire and loosen outboard trunnion mount support clamp; then loosen outboard trunnion support fitting retaining bolts.
4. Move outboard trunnion support the proper number of serrations (forward or aft) as required to align engine horizontally, as determined in step 1.

NOTES

- ***Each 1/32-inch serration equals about 3/32-inch movement of the thrust reverser doors.***
- ***Moving the trunnion support forward moves the thrust reverser doors outboard.***

5. Tighten and torque outboard trunnion support fitting retaining bolts to 120 inch-pounds (maximum). Tighten and torque trunnion mount support clamp nut to 50-70 inch-pounds, and safety with lockwire.
6. Install aft upper and lower pod doors and check horizontal alignment of thrust reverser doors. Repeat steps 1 through 5, as necessary, to obtain proper horizontal alignment between doors and pod.
7. Check vertical alignment and if adjustment is necessary, through upper forward access, remove cotter pin from eccentric retaining shaft.
8. Loosen retaining nut and disengage lock from eccentric retaining bolt of forward steady support.
9. Rotate eccentric retaining shaft, as required, to bring thrust reverser doors into desired vertical alignment.
10. Place lock on retaining shaft splines (remove, rotate, and reinstall as necessary) to match clutch teeth with locking plate.
11. Install nut, torque nut to 50-70 inch-pounds and safety with cotter pin.
12. Check engine to fuel control rigging.
13. Install pod access doors.

ENGINE STARTING PROCEDURES

Before the initial start of a replacement or repaired engine, a preliminary cold run of the engine is required. Perform a preoperational check of newly installed engine as follows:

1. Service engine oil system. (Refer to 12-16-00.)
2. Remove protective covers from engine inlet duct and tail pipe. Check for foreign material at engine air inlet and at starter-generator cooling-air inlet and exit. Remove any foreign material, oil, or fuel from inlet duct, engine compartment, and engine tail pipe.

CAUTION

Remove any objects on wing upper surfaces and ground which may be drawn into the engine air inlet.

3. Remove fuel pressurizing and dump valve inlet screen. (Refer to "Removing and Cleaning Fuel Pressurizing and Dump Valve Inlet Screen") in chapter 73.

3A. Place container under fuel pressurization and dump valve to collect fuel which will flow from valve.

4. Install chocks at wheels.
5. Connect external power source to external power start receptacle.

NOTE

When an external power unit is used for starting the engines, the unit must be capable of providing not less than 24 VDC and 800 amps, and not more than 30 VDC and 1200 amps.

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MAINTENANCE PRACTICES (Continued)

ENGINE STARTING PROCEDURES (Cont'd)

6. Pull out IGN & BOOST circuit breaker applicable to engine being checked. Check that all other applicable circuit breakers are engaged (pushed in).
7. Move ELECTRIC MASTER switch to ON.
8. Move battery switch to ON.
9. Move INVERTER switch to NO. 1.

WARNING

Make certain the area behind, and in front of, engine being checked is clear of personnel and equipment.

10. Move ENGINE MASTER switch for engine being checked to ON.

CAUTION

Check that applicable fuel shutoff valve is open. The engine fuel control unit is lubricated by fuel, and it will be damaged in operation if the fuel control valve is closed.

11. Momentarily depress engine starter switch for engine being checked. Check for proper engine rotation (clockwise as viewed from rear). Listen for noises in engine which may indicate rubbing of compressor rotor or turbine wheels, or any other malfunction. If any indication of engine malfunction is observed, pull out starter switch button, position all previously energized switches to OFF, and proceed with recommended corrective action. (Refer to 71-00-01.) If there is no indication of malfunction proceed with step 12.
12. With starter switch held depressed, check that engine speed reaches 10% rpm within 10 to 20 seconds.

WARNING

If engine speed does not reach 10% rpm within 20 seconds after starter switch is depressed, pull out starter switch button, as engine is not safe to start.

CAUTION

To prevent damage to starter switch and associated starting system, never hold starter switch depressed for more than 120 seconds (total time on) in any 20-minute period.

NOTE

When starting engine, it is recommended that 10% to 12% rpm be obtained before throttle lever is advanced to IDLE. However, if rpm stabilized at 10% rpm, advance throttle lever to IDLE without further delay.

13. If engine speed reaches 10% rpm within 20 seconds, advance throttle lever (for engine being checked) to IDLE.
    A. Check that engine oil pressure is within 1 to 5 psig. If no indication of oil pressure is obtained, proceed with preoperational check of engine if no other discrepancy is noted during checks in steps B and C.
    B. Check that fuel control shutoff valve is open as indicated by flow of fuel from the fuel pressurizing and dump valve inlet screen opening.
    C. Check that fuel flow rises within 20 seconds to at least 300 pounds per hour (pph) but to not more than 600 pph at 10% rpm. If fuel flow is not within 300 and 600 pph, move throttle lever back to OFF, pull out starter switch button, and proceed with recommended corrective action. (Refer to 71-00-01.)
14. If fuel flow is within 300 and 600 pph, move throttle lever back to OFF, pull out starter switch button, and check that fuel flow immediately drops to zero and that fuel immediately stops flowing from fuel pressurizing and dump valve inlet screen opening.

WARNING

If fuel flow does not drop to zero immediately following positioning of throttle lever to OFF and release of starter switch, the engine is not safe to start.

15. Repeat steps 11 through 14.

NOTES

- ***A minimum of two cold starts (20 seconds minimum each start) must be made to clear the engine fuel system of as much preservative oil as possible.***
- ***Subsequent to the second cold start, if fuel is noticed in the tail pipe, the engine may be motored with power lever at OFF to dry tail pipe of any collected fuel.***

16. Following completion of at least two cold starts, push in IGN & BOOST circuit breaker and move ENGINE MASTER switch to OFF.

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MAINTENANCE PRACTICES (Continued)

ENGINE STARTING PROCEDURES (Cont'd)

17. Check engine carefully for fuel and oil leaks. Ensure that all drains from engine are open and working properly. Remove any puddles of fuel or oil which may have accumulated in engine tail pipe.
18. Disconnect external electrical power source.
19. Install fuel pressurizing and dump valve inlet screen. (Refer to Removing and Cleaning Fuel Pressurizing and Dump Valve Inlet Screen in chapter 73.)
20. Determine starting characteristics of engine when powered only by airplane batteries as follows:
    A. Check that airplane batteries are fully charged. (Refer to 12-18-00.)
    B. Check that throttle levers for both engines are positioned at OFF.
    C. Check that battery switch is at ON.
    D. Depress starter switch for engine being checked. Pull out starter switch button when engine rpm stabilizes. Make a note of stabilized engine rpm and length of time required to attain stabilized engine speed.

CAUTION

Do not exceed starter limitations. (Refer to Step 12.)

21. Following satisfactory completion of preoperational check of newly installed engine, position all previously actuated switches to OFF.
22. Perform post-engine-installation inspection. (Refer to 71-00-05.)

OPERATIONAL CHECK OF ENGINES

An operational check of a newly installed engine is mandatory following the required preoperational check. The procedure herein described is applicable to newly installed engines or during postflight checks or scheduled maintenance of the airplane. In the case of a newly installed engine, only the switches and controls applicable to that particular engine need to be actuated. When both engines are replaced or newly installed, and during maintenance of the airplane, the right engine should be started first. To start engine, proceed as follows:

CAUTION

To prevent a possible overtemperature during start, if the status of the airplane to engine fuel control rigging is unknown, the engine control system rigging shall be checked prior to engine start.

1. Connect external power source to external power start receptacle.

NOTE

When an external power unit is used for starting the engines, the unit must be capable of providing not less than 24 VDC and 800 amps, and not more than 30 VDC and 1200 amps.

2. Move right and left DC generator switch to ON.
3. Move ELECTRIC MASTER switch to ON.
4. Move battery switch to ON.
5. Move HYDRAULIC PUMP switch to AUTO. Check system pressure.
6. Move INVERTER switch to NO. 1.
7. Move applicable ENGINE MASTER switch (for right or left engine) to ON.

NOTE

When both engines are to be checked, the right engine should be started first to prevent left engine operation from endangering personnel entering or leaving the airplane, or disconnecting the external electrical power source. If only one engine is to be checked, only the switches and controls applicable to that engine (right or left) need be actuated unless otherwise specified.

8. Depress engine starter switch momentarily.

NOTES

- ***The depressed starter switch energizes the starter and arms the ignition circuit.***
- ***When the external power source is used, the starter should accelerate the engine to 10% rpm in 10 to 15 seconds and should motor the engine at a steady-state speed of 12% to 15% rpm.***
- ***If the airplane battery power is used, the starter should accelerate the engine to 10% rpm in 10 to 20 seconds and should motor the engine at a steady-state of speed of 10% to 12% rpm. For a battery start on low batteries, the engine starter switch must be held depressed until the engine reaches approximately 30% rpm.***

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MAINTENANCE PRACTICES (Continued)

OPERATIONAL CHECK OF ENGINES

NOTES (Cont'd)

- *During the initial phase of the starting cycle (0% to 10% rpm) on the right-hand engine, a negative fuel flow (as much as -100 pounds per hour) may be indicated. This negative fuel flow indication is normal because of the effect of the heavy current passing through the starter leads adjacent to the fuel flow transmitter, and does not indicate a faulty fuel flow transmitter.*

9. Advance throttle to IDLE at 10% to 12% rpm.

CAUTION

If light-off has not occurred within 20 seconds after throttle is advanced, abort start by retarding throttle to cutoff. Motor engine for 20 seconds to remove as much fuel as possible; then pull out starter switch. Wait at least 30 seconds for combustion chambers to drain before attempting another start. Remove any puddles of fuel which may have accumulated in tail pipe.

NOTE

Throttle advancement energizes the ignition and fuel boost pump, and opens the fuel control shutoff valve, admitting fuel to the engine.

WARNING

If an engine fire occurs, follow procedures given for motoring engine and for a fire in the engine.

CAUTION

Excessive fuel flow will cause a hot start.

10. If fuel is less than 250 pounds per hour when light-off occurs, or exhaust gas temperature exceeds 480°C and is still increasing at a rapid rate during the start, retard throttle to OFF position and pull out starter switch. If exhaust gas temperature limit of 525°C is exceeded, refer to 71-00-05, exhaust gas temperature limitations table.

NOTES

- *Insufficient fuel flow will result in improper burning and unsatisfactory acceleration, ending in a "hung" or false start.*
- *Retarding the throttle to the OFF position de-energizes the ignition system, closes the fuel control shutoff valve, and de-energizes the fuel boost pump.*
- *Pulling out the starter switch de-energizes the starter.*

11. After engine rotation has stopped, check engine tail pipe for puddles of fuel. Remove fuel as required.

NOTE

Operation of the starter is limited to a total of 120 seconds in any 20-minute period.

12. If a successful light-off occurs, the starter should continue to operate, assisting the engine to accelerate until the start controller relay opens and causes the starter to be de-energized and the ignition to stop. The starter switch automatically releases (pops up) at about 25% to 35% rpm, and the ignition-on light goes out. If ignition-on light remains on after idle rpm is reached, manually pull starter switch.

NOTES

- *When battery power only is used, the ignition-on light will go out at an engine speed of about 35% to 40% rpm.*
- *When starting with an external electrical power source, the engine should accelerate to idle speed in about 35 to 40 seconds from the time the starter switch is depressed.*

13. Note that engine oil pressure rises immediately on light-off. If a rise of at least 5 psi is not obtained by the time engine speed has reached 35% rpm, abort start.

14. At idle rpm, move right generator switch to RESET and then to ON to bring right generator on the line, if it is not already on the line.

NOTE

This brings the generator on the line and restores the cabin air conditioning system to operation. The generator caution lights should be extinguished.

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MAINTENANCE PRACTICES (Continued)

OPERATIONAL CHECK OF ENGINES (Cont'd)

WARNING

If the throttle is inadvertently retarded to the off position during engine operation, there will be an immediate flame-out, which will be impossible to catch regardless of how quickly the throttle is advanced. Do not try to regain engine operation by advancing throttle, since raw fuel will be sprayed into the hot engine and create a dangerous condition. To restart engine, wait at least 2 minutes for engine to stop rotating. Remove any fuel that may have collected in tail pipe prior to next start.

15. Make engine idle speed check.

NOTE

When a start is made with an engine which has been cold-soaked to a temperature of -35°C (-31°F), operate engine for 2 minutes at idle speed to warm engine and accessories before advancing throttle lever above IDLE.

16. If the running engine is the right-hand engine, disconnect external electrical power source.

WARNING

If the running engine is the left engine, shut down engine before attempting to disconnect external power source.

17. If an engine is to be started while other engine is running, throttle running engine to idle rpm before depressing starter switch for second engine.

CAUTION

Always make certain that running engine (whether right or left) is throttled to idle rpm before starting other engine. Generator output of the running engine (operating above idle) plus battery power may impose damaging loads on the starter shaft of the second engine.

ENGINE SHUTDOWN

1. Retard throttle to IDLE position and hold for 2 to 5 minutes or until exhaust gas temperature stabilizes at value observed in 71-00-05, idle speed check.

CAUTION

Do not shut down engine by use of airframe fuel shutoff valve, as engine-driven fuel pump is lubricated by fuel and will be damaged by operation without fuel.

2. Advance throttle until engine speed reaches 75% rpm and hold for 30 seconds; then move throttle rapidly past idle stop to full cutoff position.
3. Note that fuel flow immediately drops to zero.
4. Observe coast down for any unusual noises (scraping or grinding) at low speeds and especially during final rotations.
5. Move ENGINE MASTER switch to OFF after engine has stopped.
6. Move all switches to OFF.

WARNING

If there is any evidence of fire within the engine after shutdown, follow the procedure given for a fire in the engine.

7. Check entire engine for fuel and oil leaks.
8. Check engine oil level.

MOTORING ENGINE

1. Check that LH ENGINE — MASTER, RH ENGINE — MASTER, BOOST PUMP — LH FUEL, BOOST PUMP — RH FUEL circuit breakers are pushed in.
2. Move ELEC MASTER switch to ON.
3. Move battery switch to ON.
4. Check that throttles are in OFF position.
5. Move ENGINE MASTER switch for selected engine to ON. This opens the airframe fuel shutoff valve.
6. Depress selected engine starter switch. Observe starter limitations.
7. To stop motoring, pull out starter switch.
8. Move all switches to OFF.

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MAINTENANCE PRACTICES (Continued)

FIRE IN ENGINE

NOTE

Black smoke coming from the tail pipe after shutdown is an indication of burning oil or fuel which will cause damage to the engine.

1. Move throttle to OFF if it is not already in this position.
2. Motor engine with starter for 30 seconds. If fire persists, continue motoring for an additional 60 seconds.
3. If fire persists after 90 seconds of motoring, apply a fire extinguisher to engine inlet and continue motoring until fire is extinguished.

CAUTION

Use a fire extinguisher only if motoring fails to extinguish fire.

FIRE IN ONE ENGINE POD

1. Pull out proper FIRE PULL handle.
2. Position extinguisher discharge switch to EXT NO. 1. The extinguisher agent will be discharged into right or left engine pod, depending on which FIRE PULL handle was pulled out.
3. Move applicable throttle to cutoff position.
4. Move applicable ENGINE MASTER switch to OFF.
5. If fire has been extinguished, return FIRE PULL handle to neutral position.
6. If pod requires a second charge of fire-extinguishing agent, leave FIRE PULL handle pulled out and position extinguisher discharge switch to EXT NO. 2.

FIRE IN BOTH ENGINE PODS

1. Pull out left FIRE PULL handle. Move extinguisher discharge switch to EXT NO. 1. The extinguishing agent will be discharged into the left engine pod.
2. Move left throttle to cutoff position and left ENGINE MASTER switch to OFF.
3. Pull out right-hand FIRE PULL handle and position extinguisher discharge switch to EXT NO. 2. The extinguishing agent will be discharged into the right engine pod.

NOTE

The FIRE PULL handles are mechanically interlocked so that only one handle at a time can be pulled out, and consequently, only one pod at a time can be sprayed with extinguisher agent.

4. Move right throttle to cutoff position and right ENGINE MASTER switch to OFF. Return right-hand FIRE PULL handle to neutral position.

POSTFIRE PROCEDURE

NOTE

Normally the fire-extinguishing agent (bromotrifluoromethane) evaporates completely without leaving any deposit, and it is not necessary to perform any cleaning operation in the pod sprayed by the agent. However, the agent might puddle, particularly on cold days, and cause corrosion. Therefore, the pod should be inspected and, if any puddles of agent are found, they should be blown out with compressed air.

1. The fire-extinguishing system lines and valves should be serviced. (Refer to chapter 26.)
2. The engine, pod, and airframe should be inspected for fire damage.
3. The fire-extinguishing and detection system should be inspected as directed in chapter 26.

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1. Idle Speed Check

A. With throttle at IDLE, allow engine to stabilize. Then check the following:

(1) Check that engine idle speed is 42% (±1%) rpm. If this speed is not obtained, check that engine control lever is at idle detent and that pointer on fuel control shaft is at "IDLE" mark on quadrant. Then turn idle adjustment screw on fuel control to obtain proper engine idle speed. To correct insufficient (low) engine idle speed, turn idle adjustment screw counterclockwise. To correct excessive (high) engine idle speed, turn idle adjustment screw clockwise and bring engine speed slightly under 42% (±1%) rpm; then turn screw counterclockwise to the point where engine speed is 42% (±1%) rpm.

NOTE: Always make final adjustments of engine idle speed in increase rpm direction.

(2) Actual exhaust gas temperature: 515°C maximum.

(3) Fuel flow: 300 to 600 pounds per hour.

(4) Lubrication oil pressure: 35 psi minimum.

NOTE: If oil pressure increases or decreases 10 psi or more from normal stabilized value (at any rpm), investigate cause.

- On a cold-temperature start, the oil pressure may reach as high as 60 psi before stabilizing in the required range.

(5) Lubrication oil pressure fluctuation: 5 psi maximum.

(6) Check that appropriate engine low oil pressure caution light has not come on.

(7) If fuel temperature is at, or below, 0°C, operation of the fuel heater may be necessary.

WARNING: OPERATION OF THE FUEL HEATER IS LIMITED TO ONE MINUTE ON, 29 MINUTES OFF, IN ANY 30-MINUTE PERIOD.

(8) Check engine for fuel, oil, and exhaust gas leaks. A maximum total fuel drainage, from engine overboard drain lines, of 5.5 cc per minute per engine is allowed.

(9) Check hydraulic pressure indicators. Normal hydraulic system pressure should be 2650 to 3200 psi. Auxiliary hydraulic system pressure should be 2800 to 3600 psi.

(10) Check electrical system for proper operation with both engines operating at IDLE.

2. Engine Interstage Bleed Valve Check

A. Check interstage bleed valve as follows:

(1) Slowly accelerate engine from idle speed and note rpm at which bleed valve closes (as indicated by a jump in rpm and turbine discharge pressure, P_{t5}).

(2) Check to make sure bleed valve closes within 78.6% and 81% rpm.

NOTE: When the bleed valve closes, the increase in engine rpm and engine pressure ratio (EPR) may be observed on the applicable engine indicating instruments.

(3) Advance throttle to 85% to 90% rpm. From this point, slowly retard throttle to IDLE and check to make sure that during deceleration bleed valve opens within 81% and 78.6% rpm.

NOTE: When the bleed valve opens, a slight decrease will be noted in engine rpm and engine pressure ratio (EPR).

3. Part-Power Trim Check

A. Check part-power trim as follows:

(1) Remove both lower and upper aft engine access doors.

NOTE: Before operation of an engine with the lower engine access door open or removed, the upper aft engine access door must also be removed. This is necessary to maintain sufficient cooling when the engine is in operation.

(2) Connect a 0- to 80-inch mercury manometer or pressure-sensitive gage to monitor turbine discharge pressure (P_{t5}) as follows:

(a) Remove pressure cap from turbine discharge pressure line "T" fitting at pylon fire wall.

(b) Connect manometer or pressure gage hose to "T" fitting.

(3) Install part-power mechanical stop in part-power operating position.

(4) Just before making a part-power trim check, compute maximum allowable turbine discharge pressure (P_{t5}). Use curve from figure 71-00-601.

(5) After completing engine idle speed check and engine interstage bleed valve check, slowly advance throttle until fuel control lever contacts part-power mechanical stop. Make a minimum of three slow accelerations from idle to part power stop. Stabilize engine for 3 minutes or, if more than 3 minutes is required, until P_{t5} stabilizes.

CAUTION: DO NOT EXCEED REQUIRED TURBINE DISCHARGE PRESSURE (P_{t5}) BY MORE THAN 0.5 IN. HG. (SEE FIGURE 71-00-601.)

(6) Check that cabin air-conditioning manifold bleed-air, engine anti-icing, and engine fuel heater systems are off.

CAUTION: IF FUEL TEMPERATURE IS AT, OR BELOW, 0°C, OPERATION OF THE FUEL HEATER MAY BE NECESSARY. DO NOT OPERATE FUEL HEATER WHILE ACTUALLY CHECKING OR TRIMMING ENGINE P_{t5}. NONESSENTIAL AIR EXTRACTION DURING TRIMMING WILL RESULT IN OVER-TRIMMED ENGINE.

- OPERATION OF THE FUEL HEATER IS LIMITED TO ONE MINUTE ON, 29 MINUTES OFF, IN ANY 30-MINUTE PERIOD.

(7) Make sure proper turbine discharge pressure (P_{t5}) for ambient atmospheric temperature and pressure conditions is obtained. (See figure 71-00-601.)

(8) If necessary, adjust P_{t5} within allowable limits by means of maximum rpm adjustment screw (adjacent to idle adjustment screw on fuel control). The Allen-head adjustment screw has serrations or detents and can be felt to click into these detents. Seven clicks of adjustment change P_{t5} by about one in. Hg. Adjust P_{t5} as follows:

NOTE: Final part-power trim P_{t5} should be no more than target value plus 0.5 in. Hg. (See figure 71-00-601.)

- The part-power stop adjustment screw is set and safetied during a flow bench adjustment of the fuel control. Under no circumstances should this setting be changed in the field.

- If engine does not have to be trimmed, proceed with steps (12) through (20).

(a) To increase P_{t5}, turn maximum rpm adjustment screw counterclockwise on fuel control a sufficient number of clicks to obtain required P_{t5}.

(b) To decrease P_{t5}, turn maximum rpm adjustment screw clockwise until P_{t5} gage reading is below desired P_{t5} value. Then turn adjustment screw counterclockwise until required P_{t5} value is obtained.

NOTE: Always make final adjustments of engine speed in the increase rpm (or increase P_{t5}) direction.

(9) Correct engine speed to Standard Day (59°F), using figure 71-00-602.

(10) Compare corrected engine speed obtained [corrected for ambient temperature (figure 71-00-602)] with value stamped on engine data plate. Make sure corrected engine speed falls within +1.872%, -0%

(+300, -0) rpm of the data-plate engine speed. To correct data-plate speed to a percent value, assume that 1% equals 160 rpm.

(11) Check engine pressure ratio (EPR) indicator for accuracy, using curve in figure 71-00-603.

(12) Check exhaust gas temperature. Exhaust gas temperature must not exceed 655°C.

(13) Engine oil pressure should be 45 (±5) psi. If oil pressure reading is out-of-limits, shut down engine (refer to 71-00-4) and follow procedure outlined in Pratt & Whitney engine maintenance manual, PWA 435107.

NOTE: Oil pressure fluctuation: 5 psi maximum.

- If a sudden rise of 10 psi or more occurs, shut down engine and investigate even if increased pressure does not exceed 50 psi.

(14) Check that appropriate engine low oil pressure caution light has not come on.

(15) Check hydraulic pressure indicators. Normal hydraulic system pressure should be 2650 to 3200 psi. Auxiliary hydraulic system pressure should be 2800 to 3600 psi.

(16) Check air conditioning and pressurizing system for proper operation.

(17) If it was necessary to adjust P_{t5} as outlined in step (8), retard throttle to IDLE. Recheck engine idle speed as outlined in procedure for idle speed check.

(18) Shut down engine. Remove part-power mechanical trim stop. Reinstall it, in stowed position, on engine fuel control body, and safety with lockwire.

(19) Make take-off thrust check. Perform thrust reverser check if applicable.

4. Thrust Reverser Check

NOTE: During this check, the nacelle doors (upper and lower) may be installed or removed.

A. To perform thrust reverser check, proceed as follows:

(1) Remove ground maintenance safety lock from thrust reversers.

(2) Just before making a thrust reverser check, compute desired reverse thrust EPR. Use curve of figure 71-00-606.

(3) Start engines. With engines operating at idle speed, check that both "ARM" indicator lights (amber) are on.

(4) Move both reverser control levers up and aft. The reverser doors (targets) should deploy (reverse thrust position) within 2 seconds. Check that "UNLOCK" (red) and "DEPLOY" (blue) indicator lights are on.

NOTE: The "DEPLOY" indicator lights should illuminate within 2 seconds after the "UNLOCK" indicator lights illuminate.

(5) Move both reverser control levers forward and down. The reverser doors should stow within 2 seconds.

(6) Check that both "DEPLOY" and "UNLOCK" indicator lights go off.

NOTE: The "UNLOCK" indicator lights should go out within 2 seconds after the "DEPLOY" indicator lights go out.

(7) Move "CABIN AIR" switch to BOTH ENG.

(8) Deploy thrust reversers. After thrust reversers are deployed, move reverser control levers aft until engine speed reaches 45 to 50%.

(9) Move "THRUST REVERSER" control switch to EMER STOW. Check that "DEPLOY," "ARM," and "UNLOCK" indicator lights are off.

(10) Move thrust reverser control levers aft until EPR computed in step (2) is obtained.

(11) Rotate both stop bell-crank adjustment knobs (figure 76-00-2) until stop bell-cranks contact thrust reverser control lever stops.

(12) Check that intersection of temperature and elevation lines corresponds to ambient (field) conditions.

(13) If corresponding lines in step (12) do not intersect, loosen temperature scale retaining screws and adjust scale to obtain proper temperature-elevation relationship; then tighten scale retaining screws.

(14) Move thrust reverser control levers full forward and down.

(15) Move "THRUST REVERSER" control switch to ON and "CABIN AIR" switch to OFF. Check that both "ARM" indicator lights are on.

(16) Shut down engine if take-off thrust check is not required. Install ground maintenance safety locks on thrust reversers.

5. Take-off Thrust Check

A. Take-off thrust check may be made by use of either engine pressure ratio (EPR) or turbine discharge pressure (P_{t5}); however, for accurate take-off thrust check, use P_{t5} method, making use of mercury manometer or pressure-sensitive gage. Check take-off thrust as follows:

(1) Install all engine access doors.

NOTE: When checking take-off thrust by P_{t5} method, route turbine discharge pressure line through cooling air scoop on lower engine access door. Secure line so that paint on cowl door is not damaged.

(2) Compute maximum allowable engine pressure ratio (EPR) or turbine discharge pressure (P_{t5}). Use EPR or P_{t5} curves from figure 71-00-604 or 71-00-605.

(3) Start engine. Apply and hold brakes. Slowly advance throttle until computed maximum allowable engine pressure ratio or turbine discharge pressure is obtained.

WARNING: HOLD BRAKES APPLIED EVERY TIME ENGINE IS RUN AT SPEEDS ABOVE IDLE. ADVISE PERSONNEL WORKING AROUND AIRPLANE THAT ENGINE WILL BE RUN AT HIGH POWER SETTINGS.

CAUTION: DO NOT EXCEED COMPUTED MAXIMUM ALLOWABLE EPR OR P_{t5} VALUE.

- DO NOT EXCEED RPM SPEED LIMITATIONS OF STEP (4) (a).
- DO NOT EXCEED EXHAUST GAS TEMPERATURE LIMITS OF STEPS (4) (b).

NOTE: Advancement of the throttle at this point does not constitute an acceleration check. If instability is encountered, retard throttle to 65% rpm, and after engine has stabilized, again slowly accelerate to computed maximum allowable EPR or P_{t5}.

- Check to be certain that cabin air control is at BOTH ENG and engine anti-icing system, windshield anti-icing system, and engine fuel heater system are off.

CAUTION: DO NOT OPERATE ENGINES AT TAKE-OFF THRUST FOR AN EXTENDED PERIOD. EXTENDED OPERATION AT THIS POWER SETTING MAY CAUSE POSSIBLE DISTORTION AND/OR BUCKLING TO THE ENGINE ACCESS DOORS.

(4) Stabilize engine at Take-off Thrust EPR or P_{t5}; then check the following:

(a) Engine speed should be below 104.2% (16,700) rpm.

CAUTION: IF ENGINE SPEED EXCEEDS 104.2% (16,700) RPM, IMMEDIATELY SHUT DOWN ENGINE. IF ENGINE SPEED EXCEEDED 105.4% RPM, REPLACE ENGINE. IF ENGINE SPEED WAS LESS THAN 105.4% RPM, BUT GREATER THAN 104.2% RPM, A VISUAL INSPECTION MUST BE PERFORMED. IF ANY ABNORMAL CONDITION IS EVIDENT, REPLACE ENGINE.

(b) The exhaust gas temperature should stabilize below 718°C.

CAUTION: IF MAXIMUM OVERTEMPERATURE LIMITS ARE EXCEEDED, REFER TO EXHAUST GAS TEMPERATURE LIMITATIONS TABLE.

(c) (Deleted)

(d) Engine oil pressure should be in the range of 40 to 50 psi.

NOTE: Oil pressure fluctuation: 5 psi maximum.

(e) If oil pressure reading is out-of-limits, refer to Pratt & Whitney engine maintenance manual, PWA 435107.

(5) Check that appropriate low oil pressure caution light is not on.

(6) Check hydraulic pressure indicators. Normal hydraulic system pressure should be 2650 to 3200 psi. Auxiliary hydraulic system pressure should be 2800 to 3600 psi.

(7) If fuel temperature is at, or below, 0°C, operation of fuel heater may be necessary.

WARNING: OPERATION OF THE FUEL HEATER IS LIMITED TO ONE MINUTE ON, 29 MINUTES OFF, IN ANY 30-MINUTE PERIOD.

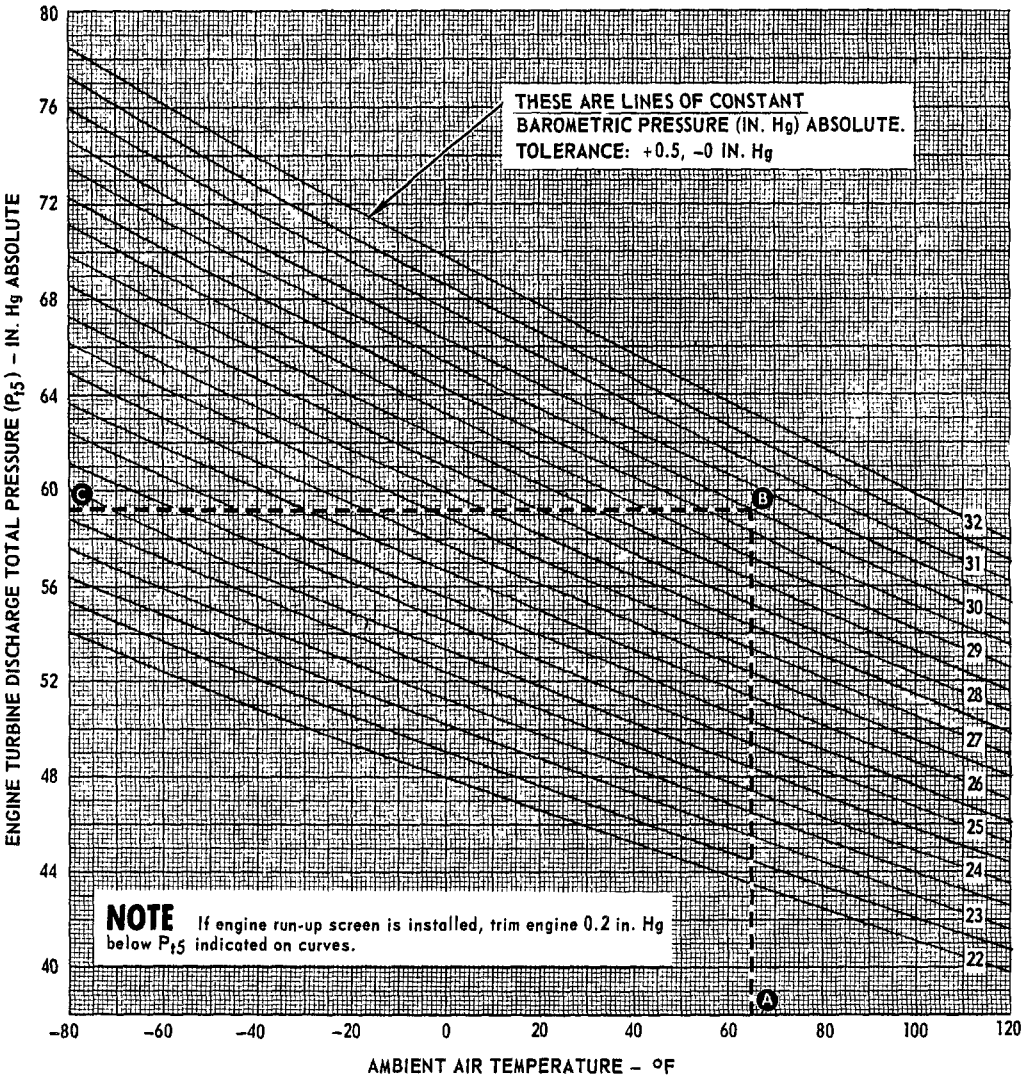
(8) Place "CABIN AIR" selector switch at BOTH ENG position and sniff-check cabin conditioning air. If fuel or oil fumes are detected, investigate source.

6. Acceleration and Deceleration Checks

A. After it has been determined that engine operates properly up to and at Take-off Thrust, make the following acceleration check:

(1) With engine operating at Take-off Thrust, place finger on throttle quadrant in front of, and against, throttle lever. While holding finger firmly at this position on quadrant, retard throttle to IDLE and allow engine to stabilize. Advance throttle rapidly (one second or less) to finger (Take-off Thrust position). Watch tachometer and make sure that acceleration time does not exceed 8 seconds when tachometer reading is 1-1/4% (200) rpm less than Take-off Thrust reading at start of this check. The exhaust gas temperature must

NOTE These curves are for engine operation with air inlet duct (with splitter vanes) and tail pipe installed, nacelle doors removed, no inlet screen, no compressor air bleed or power extraction, and fuel control lever against part-power mechanical trim stop.



EXAMPLE:

To find computed part-power trim P_{t5} using these curves:

1. Enter ambient temperature (taken in airplane nose wheel well) at a point on the total temperature axis. In this example, 65 °F at point "A"
2. From point "A" extend a vertical line to intersect station barometric pressure line (in this example, 30 in.Hg) at a point "B."
3. From point "B," extend a horizontal line to the left and read value of P_{t5} at a point "C," where the horizontal line intersects the P_{t5} scale. In this example, P_{t5} is then 59.20 in. Hg.

S370-2-42-3

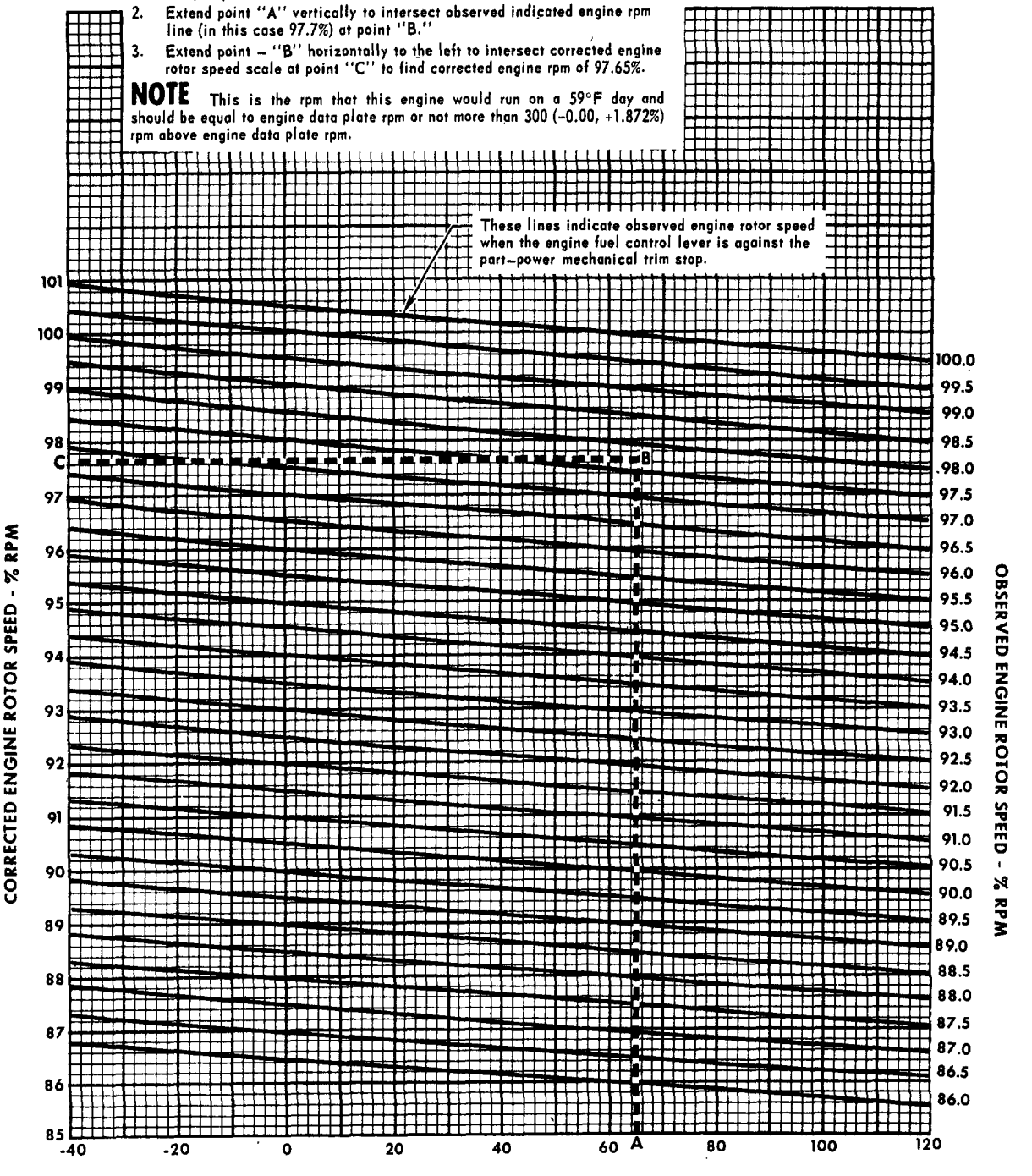
Part-power Trim P_{t5} Curve
Figure 71-00-601

EXAMPLE
AMBIENT TEMPERATURE– 65°F

To find corrected engine speed:

1. Enter ambient temperature (65 °F, reading taken in airplane nose wheel well) at point "A."
2. Extend point "A" vertically to intersect observed indicated engine rpm line (in this case 97.7%) at point "B."
3. Extend point – "B" horizontally to the left to intersect corrected engine rotor speed scale at point "C" to find corrected engine rpm of 97.65%.

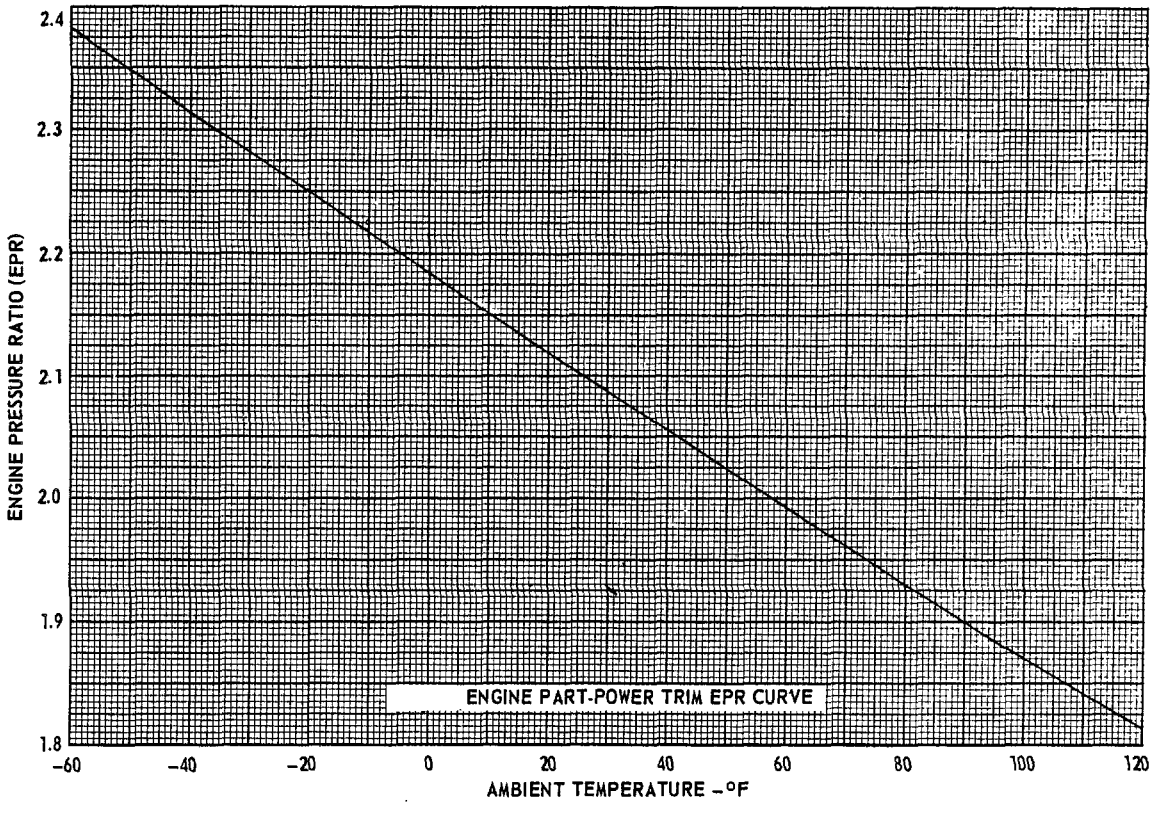
NOTE This is the rpm that this engine would run on a 59°F day and should be equal to engine data plate rpm or not more than 300 (-0.00, +1.872%) rpm above engine data plate rpm.



S370-2-42-4

Engine Speed Temperature Correction Curve
Figure 71-00-602

1034-1 NEW 12-69

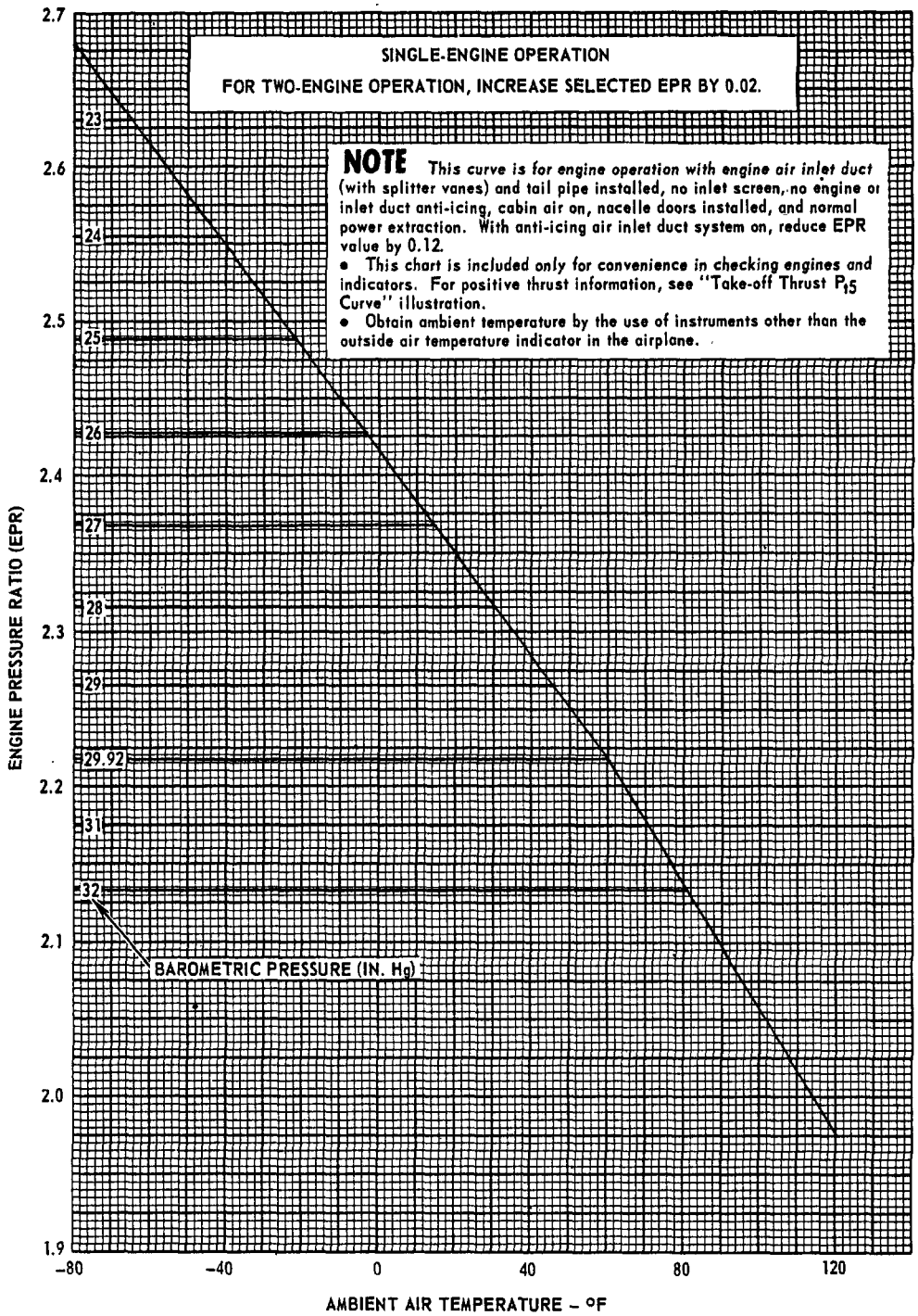


NOTE The charts are for engine operation with engine fuel control lever against the part-power mechanical trim stop, engine inlet duct and tail pipe installed, nacelle doors removed, no inlet screen, cabin air OFF, no power extraction, and no engine or inlet duct anti-icing.

- These charts are included only for checking engines and indicators. Do not use for trimming engine. For positive trim information, refer to applicable "Part-power Trim P_{t5} Curve" illustration.
- Obtain ambient temperature by the use of instruments other than the outside air temperature indicator in the airplane.

S370-2-42-5

Part-power Trim EPR Curve
Figure 71-00-603

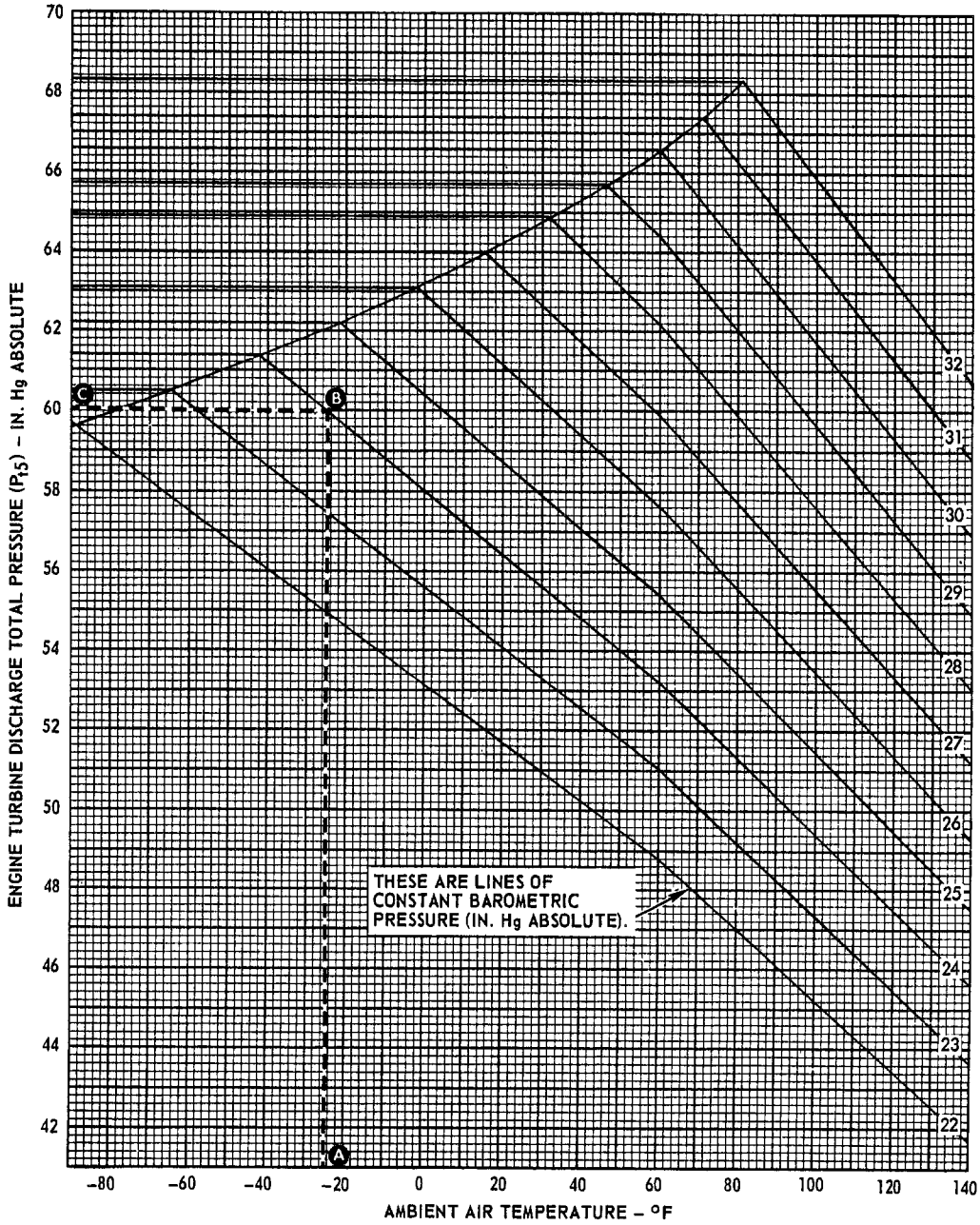

SINGLE-ENGINE OPERATION
FOR TWO-ENGINE OPERATION, INCREASE SELECTED EPR BY 0.02.
NOTE This curve is for engine operation with engine air inlet duct (with splitter vanes) and tail pipe installed, no inlet screen, no engine or inlet duct anti-icing, cabin air on, nacelle doors installed, and normal power extraction. With anti-icing air inlet duct system on, reduce EPR value by 0.12.
• This chart is included only for convenience in checking engines and indicators. For positive thrust information, see "Take-off Thrust P15 Curve" illustration.
• Obtain ambient temperature by the use of instruments other than the outside air temperature indicator in the airplane.
ENGINE PRESSURE RATIO (EPR)
2.7
2.6
2.5
2.4
2.3
2.2
2.1
2.0
1.9
23
24
25
26
27
28
29
29.92
31
32
BAROMETRIC PRESSURE (IN. Hg)
-80
-40
0
40
80
120
AMBIENT AIR TEMPERATURE - °F

Select engine pressure ratio as follows:
Enter graph at ambient temperature and project upward to diagonal or barometric pressure line, whichever occurs first. Project to left, from intersection, and read engine pressure ratio.

5370-2-42-6

Take-off Thrust EPR Curve
Figure 71-00-604

NOTE These curves are for single-engine operation with engine inlet duct (with splitter vanes) and tail pipe installed, no inlet screen, no engine or inlet anti-icing, cabin air on, nacelle doors installed, and normal power extraction. For two-engine operation, increase computed P_{t5} by 1.0 percent (multiply P_{t5} by 1.01).



EXAMPLE: To find Take-off Thrust P_{t5}:

A. Enter ambient temperature (in this example, –25°F, reading taken in nose wheel well) at point "A".

B. Extend point "A" vertically to intersect either the diagonal or the horizontal portion of ambient barometric pressure line (in this example, 24.0 in. Hg) at point "B."

C. Extend point "B" horizontally to the left to intersect P_{t5} scale at point "C" to find computed P_{t5} of 60.0 in. Hg.

S370-2-42-7

Take-off Thrust P_{t5} Curve
Figure 71-00-605

remain within limits during the entire check. Acceleration time is measured from initial throttle movement until fuel flow drop-off occurs.

CAUTION: FOR MAXIMUM OVERTEMPERATURE LIMITS, REFER TO EXHAUST GAS TEMPERATURE LIMITATIONS TABLE.

- MAXIMUM ALLOWABLE ENGINE SPEED IS 16,700 (104.2%) RPM. IF MAXIMUM OVERSPEED LIMITS ARE EXCEEDED, THE ENGINE MUST BE SHUT DOWN IMMEDIATELY AND INSPECTED AS OUTLINED IN THE PROCEDURE FOR ENGINE OVERSPEED INSPECTION.

NOTE: During the preceding check, EPR may overshoot past the Take-off Thrust value and then settle back down. This overshoot is normal and will not be considered as part of the acceleration time.

- 1-1/4% rpm less than Take-off Thrust percent is about 95% thrust, or 200 rpm less than Take-off Thrust rpm.

B. Check engine for proper deceleration as follows:

(1) Move throttle rapidly from Take-off Thrust to IDLE. Make sure deceleration time does not exceed 15 seconds.

(2) Upon reaching idle speed, make sure fuel flow does not exceed 600 pounds per hour.

7. Exhaust Gas Temperature Limitations

TYPE OF ENGINE OPERATION	TEMPERATURE - °C	DURATION OF TEMPERATURE	NUMBER OF OCCURRENCES	INSPECTION REQUIRED
Starting	Up to 525	No limit	No limit	None
	525 to 595	No limit	Each occurrence	*
	525 to 630	5 seconds or less	Each occurrence	*
	595 to 630	Over 5 seconds	Each occurrence	†
	630 to 700	5 seconds or less	Each occurrence	†
	630 to 800	Over 5 seconds	Each occurrence	Hot section
	700 to 800	5 seconds or less	Each occurrence	Hot section
	Over 800	Momentary	One	Overhaul

TYPE OF ENGINE OPERATION	TEMPERATURE - °C	DURATION OF TEMPERATURE	NUMBER OF OCCURRENCES	INSPECTION REQUIRED
Normal operation	Up to 718	5 minutes or less	No limit	None
	718 to 730	No limit	Each occurrence	*
	718 to 760	5 seconds or less	Each occurrence	*
	760 to 770	5 seconds or less	Each occurrence	†
	730 to 760	Over 5 seconds, but less than 2 minutes	Each occurrence	†
	730 to 800	Over 2 minutes	Each occurrence	Hot section
	760 to 800	Over 5 seconds, but less than 5 minutes (3).	Each occurrence	Hot section
	770 to 800	5 seconds or less	Each occurrence	Hot section
	Over 800	Momentary	One	Overhaul

*Refer to step (2) of engine overspeed inspection procecure.

†Refer to step (3) of engine overspeed inspection procedure.

8. Engine Overspeed Inspection

A. The following inspection procedures are to be used in conjunction with the preceding exhaust gas temperature limitations table when an engine overspeed condition is encountered.

(1) Perform a visual inspection, as outlined in step (3), whenever the observed engine speed exceeds 104.2% (16,700) rpm, but does not exceed 105.4% (16,900) rpm. Visual inspection is also required after the occurrences of a "hung" start or an aborted start during which residual fuel was burned within the engine. If inspection indicates engine can be considered satisfactory, continue engine in service. If any abnormal condition is evident, ship engine to overhaul. If observed engine speed exceeds 105.4% (16,900) rpm, ship engine to overhaul.

(2) Check engine as thoroughly as possible and determine cause of overtemperature.

(3) Perform a visual inspection as follows:

(a) Inspect inlet and exhaust ducts for foreign particles, evidence of rubbing, or damaged blades or vanes.

(b) Inspect compressor rotor for free rotation with no audible evidence of circumferential contact and for nicks, dents, or distortion.

(c) Inspect burner section, nozzle guide vanes, front of turbine section and visible turbine blades for evidence of rubbing, burns, nicks, dents, distortion, cracks, and other damage. Inspect rear of turbine for apparent damage.

(d) Inspect combustion chamber covers for cracks, heat damage, and excessive buckling or warping.

(e) Inspect accessory gearbox for evidence of oil leakage, security, and external damage.

9. Post-engine-installation Inspection

A. The following steps should be accomplished after an engine installation is completed.

(1) Ensure that all air intake and exhaust openings are free of obstructions.

(2) Check for fuel or oil leaks.

(3) Examine all engine accessories for security of mounting.

(4) Check all electrical leads for security and for possibility of chafing.

(5) Inspect all engine controls for freedom of movement, full travel, and security.

(6) Inspect engine mounting bolts for security.

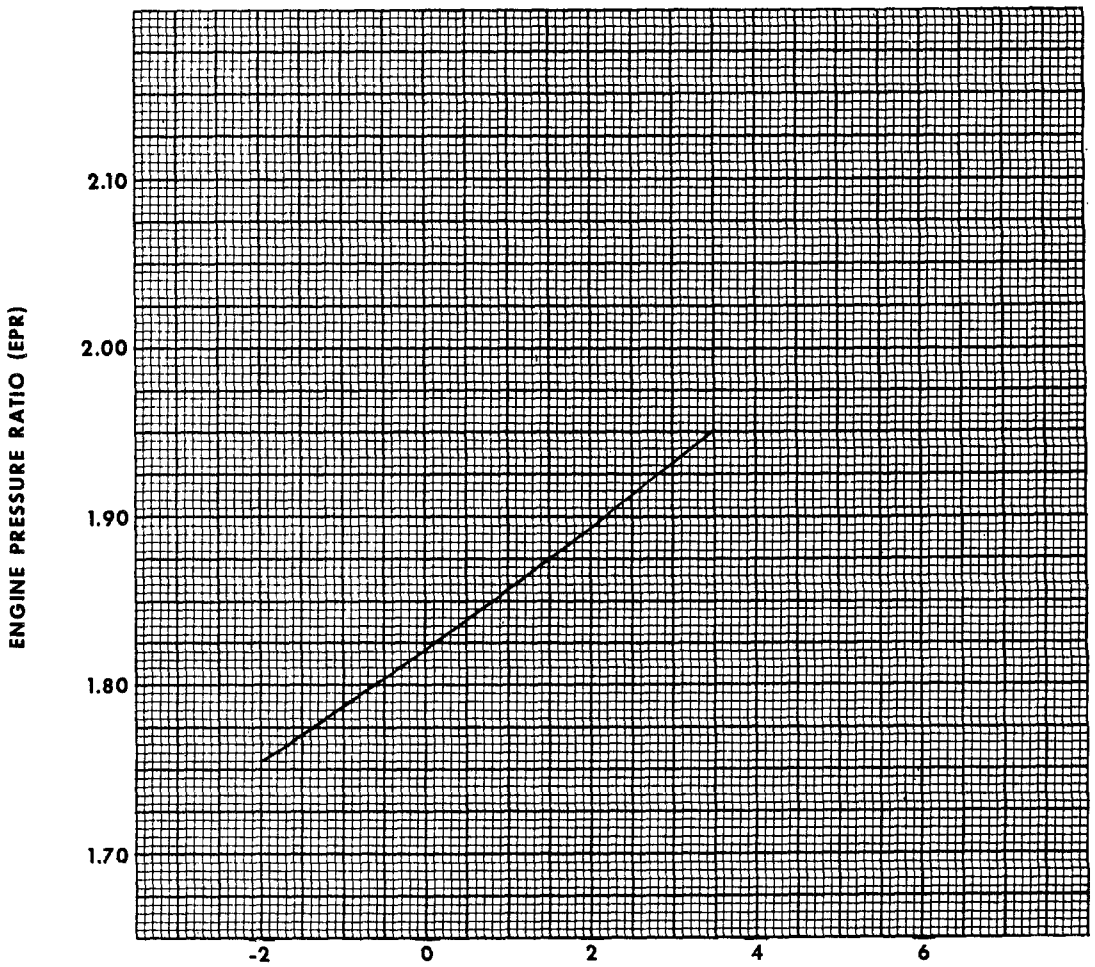
(7) Check engine oil level.

(8) Inspect engine exhaust pipe for cracks and security of installation.

TWO ENGINE OPERATION

NOTE This curve is for engine operation with thrust reverser and engine inlet duct (with splitter vanes) installed, nacelle doors installed, no inlet screen, no engine or inlet duct anti-icing, normal power extraction, cabin air on, and reverser doors (targets) stowed.

- This curve is applicable to all ambient temperatures up to 115°F.



Select engine pressure ratio as follows:

Enter graph at pressure altitude and project upward to curve. Project to left, from intersection, and read engine pressure ratio.

S370-2-42-8

Reverser Thrust Setting EPR Curve
Figure 71-00-606

ENGINE NACELLES - DESCRIPTION

1. General

For a description of engine nacelles, refer to Chapter 54.

ENGINE NACELLES - MAINTENANCE PRACTICES

1. Removal of Engine Air Intake Duct

A. To remove engine air intake duct, proceed as follows:

(1) Disconnect lower forward fire extinguisher nozzle line at upper end of flexible line.

(2) Disconnect anti-icing sensing line.

(3) Disconnect anti-icing electrical lead.

(4) Disconnect anti-icing ducts.

(5) Disconnect starter-generator cooling-air duct.

(6) Remove fuel pump differential switch from aft bulkhead of engine air intake duct.

(7) On left engine only, remove dc generator electrical lead clamps.

(8) Remove bolts attaching engine air intake duct to forward flange of engine.

(9) Remove engine air intake duct.

2. Installation of Engine Air Intake Duct

A. The installation procedure for the engine air intake is essentially the reverse of the removal procedure.

3. Opening Lower Engine Access Door

A. To open lower engine access door, proceed as follows:

(1) Latches are numbered 1 through 4, from front to rear, both inboard and outboard. Release and unhook No. 1, 3, and 4 outboard latches.

(2) Release and unhook No. 1, 3, and 4 inboard latches.

(3) Release, but do not unhook, No. 2 inboard latch.

(4) Release and unhook No. 2 outboard latch.

(5) Unhook No. 2 inboard latch released in step (3).

(6) Lift outboard side of door and push in on trigger hook to release door; then lower door to open position.

(7) Push door back toward fuselage until locking sleeve on sway brace can be slid over hinged joint in brace.

4. Closing Lower Engine Access Door

A. To close lower engine access door, proceed as follows:

(1) Push door gently toward fuselage until locking sleeve on sway brace can be pulled from hinged joint unlocking sway brace.

(2) Raise outboard side of door until trigger hook is engaged.

(3) Push up on inboard side of door and hook No. 2 inboard latch, but do not close latch.

(4) Push up on outboard side of door and hook and close No. 2 outboard latch.

(5) Close No. 2 inboard latch.

(6) Hook and close remaining latches on inboard and outboard sides.

5. Removing Lower Engine Access Door

A. To remove lower engine access door, proceed as follows:

(1) Open lower engine access door.

(2) Disconnect the two fire detector wires at inboard side of door. The disconnects pull apart under a pull of about 20 pounds.

(3) Remove lockpin from fire wall end of sway brace. Stow lockpin in free end of sway brace.

(4) Slide locking sleeve from joint, fold sway brace, and stow in bottom of lower access door.

(5) With door in two-thirds full-open position, raise entire door slightly and move it inboard to lift it off hinges.

6. Installing Lower Engine Access Door

A. To install lower engine access door, proceed as follows:

(1) While holding door in two-thirds full-open position, raise door until hinge hooks engage in hinge clevises.

(2) Attach free end of sway brace to engine fire wall bracket, using lockpin stowed in sway brace.

(3) Connect the two fire detector wires at disconnects on inboard side of door.

(4) Close lower engine access door.

7. Removing Upper Engine Access Door

A. To remove upper engine access door proceed as follows:

(1) Through fire detector disconnect access door, disconnect the two fire detector wires.

(2) Release and unhook latch on inboard side of door.

(3) Unlock door fasteners and lift door off engine pod.

(4) Unlock door fasteners and lift outboard engine access door off engine pod.

(5) Unlock door fasteners and lift inboard engine access door off engine pod.

8. Installing Upper Engine Access Door

A. To install upper engine access door proceed as follows:

(1) Place inboard engine access door in position on engine pod, and lock door fasteners.

(2) Place outboard engine access door in position on engine pod and lock door fasteners.

(3) Install upper engine access door as follows:

(a) Place door in position on engine pod and hook latch on inboard side of door.

(b) Lock door fasteners and close latch.

(c) Through fire detector disconnect access, connect the two fire detector wires at disconnects; then close fire detector disconnect access door.

ENGINE MOUNT SYSTEM - DESCRIPTION

1. General

The engine is supported within the engine pod at three points. The supports are at each side of the diffuser case and at the steady-support mount at the top of the front flange of the compressor housing. The outboard engine supports of both engine mounting components are adjustable fore and aft for horizontal positioning of the engine in the engine pod. The forward steady-support mount of the engine does not carry the weight of the engine, but is primarily used for vertical alignment of the engine. An eccentric bolt in the forward steady-support mount positions the engine to the correct vertical alignment.

ENGINE MOUNT SYSTEM - MAINTENANCE PRACTICES

1. Removal of Aft Engine Mount Fittings on Pylon Yoke

A. To remove engine aft mount, proceed as follows:

(1) Remove engine from engine pod and place in a suitable stand or container.

(2) Remove the four NAS1104 bolts securing engine mount to engine saddle.

(3) Remove engine mount.

1A. Removal of Aft Engine Mount Trunnions

A. To remove engine aft trunnions, proceed as follows:

(1) Hoist engine.

(2) Remove lockwire from 306-329010 bolts.

(3) Remove bolts and trunnion(s) from engine.

2. Removal of Engine Forward Steady-support Mount

A. To remove engine forward steady-support mount, proceed as follows:

(1) Remove engine access doors.

(2) Remove top segment of anti-icing duct forward of steady-support mount.

(3) Remove nut, washer, eccentric lock, and eccentric shaft securing mount to nacelle structure.

(4) Hold nose of engine down and remove nut, washers, and bolt securing mount to engine, and remove mount.

3. Installation of Engine Aft Mount Fittings on Pylon Yoke

A. To install engine aft mount, proceed as follows:

(1) Place engine mount and serrated shims (if required) in position.

(2) Install four NAS1104 bolts through engine mount into engine saddle.

(3) Torque NAS1104 bolts to 120 inch-pounds.

3A. Installation of Aft Engine Mount Trunnions

A. To install engine aft trunnions, proceed as follows:

(1) Install flex mount (K740-4SA-1) on trunnion. Install bolt (265-329041) through trunnion and install nut (265-329043) on bolt. Install roll pin (NAS561P3-11) through trunnion and bolt. Torque nut to 80-140 inch-pounds. Install cotter pin.

(2) With engine suspended from suitable hoist, install trunnion assembly on engine.

(3) Install four (306-329010) bolts. Torque bolts to 100-140 inch-pounds.

(4) Lockwire bolts.

4. Installation of Engine Forward Steady-support Mount

A. To install engine forward steady-support mount, proceed as follows:

(1) Hold nose of engine down and position mount on engine.

NOTE: Washers should be installed as necessary under bolthead, to ensure that nut does not bottom on shoulder of bolt when torqued.

(2) Install bolt, washers, and nut securing mount to engine. The bolt must be installed with the head forward. Torque nut to 100-140 inch-pounds, and safety with cotter pin.

(3) Install eccentric shaft, eccentric lock, washer, and nut securing mount to nacelle structure. Align engine; then torque nut to 50-70 inch-pounds. Install cotter pin.

(4) Install anti-icing duct segment and install engine access doors.

ENGINE MOUNT SYSTEM

INSPECTION/CHECK

INSPECTION OF ENGINE MOUNT SUPPORT BOLTS — S/N 370-2 thru -6, -8, and -9.

NOTES

- ***Initial compliance with this inspection requires a magnetic particle inspection or replacement of bolts with new bolts.***
- ***Visually inspect for loose engine mount support bolts and/or torque striping every 300 hrs. after initial compliance with this inspection. DO NOT REMOVE BOLTS EVERY 300 HRS.***
- ***For missing torque striping or loose bolts proceed with steps 1 thru 4.***

1. Remove engine mount support bolts following engine removal procedures in chapter 71.
2. Magnetic particle inspect (Magna-Glo) support bolts for cracks or replace with new bolts. Replace any bolts not passing inspection.
3. Install serviceable support bolts and torque to 90 inch-pounds.
4. Apply torque striping.

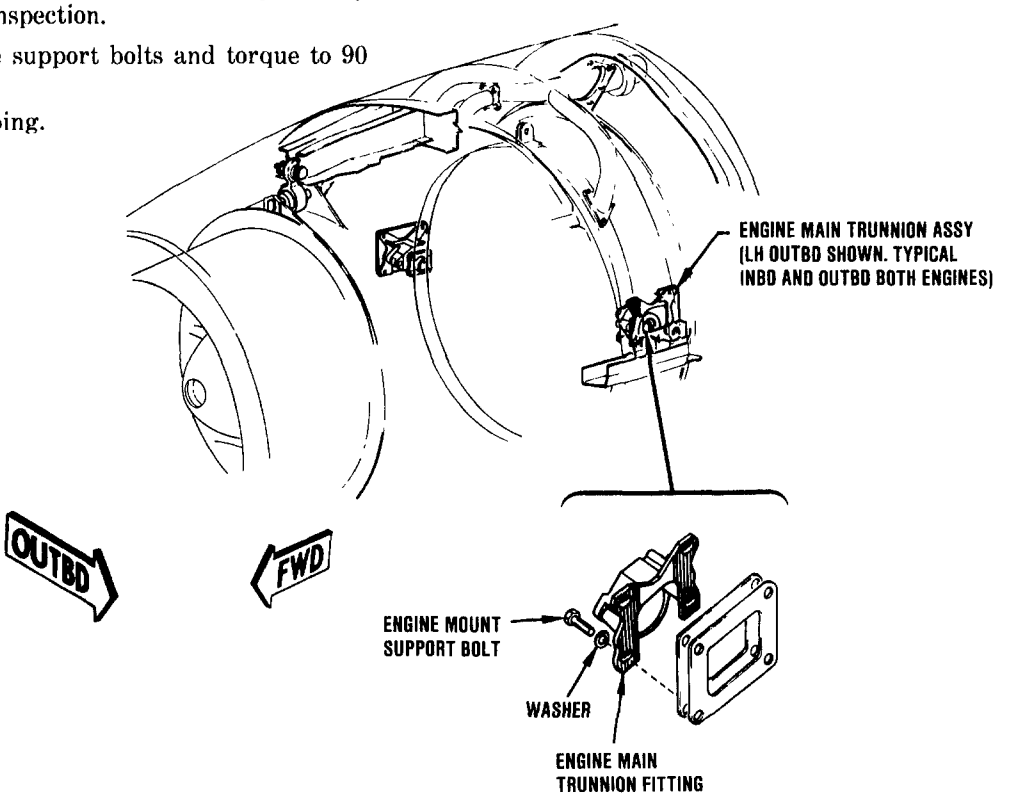


Figure 71-20-601 — Torqueing Engine Mount Support Bolts

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ENGINE FIRE SEAL - DESCRIPTION

1. General

The engine fire seal is composed of three major sections of formed stainless steel. A one-inch-diameter seal is attached to the outer edge of the fire seal by means of retaining strips. The diaphragm assembly is attached to a flange attached to the aft face of the engine diffuser case. The seal of the diaphragm assembly mates with the upper and lower engine access doors, forming an effective fire seal between the forward portion of the engine and the aft hot section of the engine.

NOTICE

The following information applies to NA 265-80 airplanes only.

CHAPTER 71

POWER PLANT — GENERAL

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CHAPTER 71

POWER PLANT — GENERAL

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POWER PLANT — GENERAL

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POWER PLANT GENERAL - DESCRIPTION

1. General

The NA-370-1, -7 and NA-380-1 and later airplanes are powered by two pod-mounted CF700-2D-2 turbofan engines. The engine is a continuous-flow, gas turbine power plant incorporating a variable geometry-air inlet section; an eight stage single-rotor axial-flow compressor; a single can-annular combustion section; a two-stage reaction turbine coupled to the compressor; an exhaust pipe with a fixed nozzle; a free-floating single stage aft fan section, and an accessory gear box with its associated power takeoff drive shaft and transfer gearbox. Replacement engines shall have a "B" suffix letter following the engine serial number on the data plate.

VARIABLE GEOMETRY SYSTEM

The variable geometry system consists of a fuel-operated variable vane actuator, an actuation bellcrank, a synchronizing cable, fifteen variable inlet guide vanes, an actuator ring, and two compressor bleed valves. Operation of the variable inlet guide vanes and bleed valves is scheduled as a function of engine RPM and compressor inlet temperature (CIT). These two engine parameters are provided by the CIT system and engine tachometer and are processed by the main fuel control, which in turn provides high pressure fuel to the vane actuator for positioning of the variable inlet guide vanes and bleed valves.

COMPRESSOR SECTION

The multistage axial-flow compressor, with a compression ratio of 6.8 to 1, consists of an eight stage rotor and seven stator stages. The rotor of the compressor is connected to the first- and second-stage turbines by a splined compressor drive shaft. A bleed air manifold on the outer portion of the compressor case, bleeds compressor air from the third, fourth, and fifth stages. This manifold is terminated near the horizontal split line of the compressor. The bleed valves are installed over these bleed ports, thus allowing the bleed valves to control the amount of bleed air that is extracted from the engine. Bleed air is also extracted from this manifold for cooling the various engine and/or aft fan components. Anti-icing and environmental system bleed air is extracted from the main frame assembly. This eight stage bleed air is ducted to the refrigeration unit and to the compressor inlet guide vane manifold. Bleed air from the manifold is directed through the compressor inlet guide vanes and back in the airstream. A portion of the bleed air is also routed to the bullet nose to prevent ice from forming on the nose.

COMBUSTION SECTION

The combustion section consists of an outer and inner casing, and a combustion liner. The annular inner combustion liner is a one piece fabrication with openings provided for the fuel nozzles, ignition plugs and air flow passages. Fuel enters the combustion liner through twelve fuel nozzles

which protrude into the forward end of the liner. Each fuel nozzle incorporates a flow divider, a primary and secondary flow passage, which ensures a proper fuel spray pattern and flow through the engine operating speed range. Electrical energy for the ignitor plugs are provided by a solid state capacitor discharge ignition unit installed at approximately the 12:00 position. The ignition unit requires 28-volts dc input which in turn converts this voltage to 3000 volts ac with a sufficient amplitude to fire the ignitor plugs.

TURBINE SECTION

The turbine section consists of the turbine casing, shrouds, nozzles and rotor. The turbine rotor extracts energy from the exhaust gas flow, which in turn drives the compressor rotor. The turbine rotor consists of two turbine wheel assemblies, a turbine interstage seal, and a torque ring assembly. The first stage turbine wheel consists of an internally splined shaft which provides the mounting provisions for the compressor rotor drive shaft.

AFT FAN SECTION

The aft fan is attached to the aft flange of the turbine casing and consists of a single stage, free floating rotor, which is part compressor and part turbine. The aft fan provides approximately 45% of the total thrust of the engine. Exhaust gasses (primary air flow) from the turbine section flow into the fan nozzle which directs the gasses to the turbine airfoil of the fan rotor blades, resulting in fan rotor rotation (power extraction). Secondary air flow enters the aft fan inlet and flows through the inlet duct to the rotating compressor airfoils of the aft fan rotor blades which compress (approximately 1.6 to 1) the secondary air flow. This air flow is then expelled into the exhaust stream. Fan rotor rpm is detected by a coil and alnico magnet mounted inside the fan rear frame. This coil and magnet produces a pulsating dc voltage during fan rotor rotation. This pulsating voltage is then converted to the actual fan rotor rpm.

ACCESSORY DRIVE GEAR BOX

The accessory drive gear box is mounted on brackets below the engine front frame. It has a total of six pads (three forward and three aft) on which the engine accessories are mounted. Power to drive the accessory drive gear box is extracted from the engine through the power takeoff shaft and a transfer gear box. The transfer gearbox is mounted on the main frame and is driven by a radial drive shaft. Power from the transfer gear box is transmitted to the accessory drive gear box through a transfer drive shaft. Lubrication for the gear boxes and drive shaft is provided by the engine lubrication system.

POWER PLANT GENERAL - TROUBLE SHOOTING

NOTE: Additional trouble shooting information may be found in the CF700 Maintenance Manual, Report No. SEI-187.

1. Trouble Shooting Engines

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. NO RPM WHEN ATTEMPT IS MADE TO START ENGINE.		
(1) No electrical power to starter system.	Check source of electrical power.	
(2) Engine is "frozen."	Attempt to rotate by hand.	Replace engine.
(3) Starter drive shaft in component drive gearbox sheared.	If sounds of starter rotation are heard during attempted start, but engine does not rotate, shaft is sheared.	Replace engine.
(4) Starter shaft sheared.	Starter rotates, but does not rotate engine; starter shaft is sheared.	Replace starter-generator.
B. INSUFFICIENT RPM WHEN ATTEMPT IS MADE TO START ENGINE.		
(1) Excessive turbine and compressor blade rubbing.	Listen for loud scraping noise while engine is rotating.	Replace engine.
(2) Defective starter system.	Check tachometer for maximum speed at which starter will rotate engine.	Replace starter.
(3) Inadequate voltage for starter.	If engine starts, but fails to accelerate because of lack of help from the starter, auxiliary power unit is inadequate.	Use large capacity auxiliary power unit.
C. COMBUSTION DOES NOT OCCUR WHEN ATTEMPT IS MADE TO START ENGINE.		
(1) No fuel to engine.	Check fuel supply and position of fuel shutoff valve.	

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
(2) Defective ignition system.	Turn on system and listen for ignition noise.	Refer to Chapter 74.
(3) Defective engine fuel system.	Check fuel flow indicator for minimum starting flow.	Refer to Chapter 73.
(4) Defective starter system.	Check tachometer for maximum speed at which starter will rotate engine.	Replace starter.
(5) Throttle linkage improperly adjusted.		Adjust throttle linkage.
(6) Faulty cutoff valve in fuel control.	If there is fuel from airplane fuel system and throttle linkage is rigged correctly, cutoff valve in fuel control is not opening.	Replace fuel control.
(7) Defective engine-driven fuel pump.	If fuel flowmeter indicates some fuel flow, but not enough for normal start, pump may be defective.	Replace engine fuel pump.
D. ENGINE RPM AND TURBINE EXHAUST TEMPERATURE FLUCTUATES.		
(1) Defective engine fuel system.		Refer to Chapter 73.
(2) Defective instruments and/or instrument circuits.	Use instruments with units of known accuracy and check engine operation.	Replace faulty instruments.
E. ENGINE ROUGHNESS OCCURS.		
(1) Internal interference between rotors and stators.	Check for scraping noise as engine slows down or at shutdown. Check oil strainer for metal particles.	Replace engine.
(2) Defective accessories.	Check for unusual noises coming from accessories, or improperly functioning system.	Replace defective accessories.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
F. ENGINE SURGES DURING ACCELERATION.		
(1) Defective fuel control unit.	If bleed system is satisfactory, fuel control unit is faulty.	Replace fuel control.

POWER PLANT GENERAL - MAINTENANCE PRACTICES

1. Prevention of Foreign-object Damage

To prevent foreign-object damage to the engine, the engine inlet, aft fan inlet and exhaust covers should be installed at all times during non-maintenance periods. The covers must be installed during movement of the airplane from one area to another. The engine and aft fan inlet ducts must be inspected for foreign objects, loose fasteners, etc., after the airplane is moved. Under no circumstances should the inlet ducts or the inlet guide vanes be used as a resting place for tools, parts, nuts and bolts, etc., during maintenance operations. An inventory of all tools, parts, and equipment must be conducted after engine maintenance operation is completed.

2. Inspect Power Plant

A. To inspect power plant, proceed as follows:

(1) Open the upper and lower access doors.

(2) Inlet duct and forward frame (casing) for deterioration, damage, loose fasteners, and for presence of foreign objects.

(3) Bulletnose for dents, nicks, cracks and security, and anti-icing holes for obstructions.

(4) Front frame struts for dents, nicks and cracks.

(4A) Front frame casing for cracks.

(5) Variable vanes for nicks, dents, cracks, excessive movement and security.

(6) Missing pin and clips from visible variable vane levers.

(7) First stage blades for nicks, dents, cracks, distortion, discoloration, FOD and corrosion.

(8) Missing cotter pins from visible variable vane outer shanks.

(9) Variable vane linkage for security and worn components; vanes for nicks, dents and cracks.

(10) Rubs between variable vanes and shrouds. Check for distortion of the variable vane actuator ring.

(11) Check for free rotation of compressor rotor, broken rotor studs (tinkling sound during rotation), blades for nicks, dents, cracks and tip curl.

(12) Bleed valves for security, loose or broken linkage. Stall warning inhibit switches and support bracket for security and damage.

(13) Ignition unit, leads, ignitor plugs for security, chafing and proper safeties.

(14) Electrical harness for security, chafing and proper routing.

(15) Fuel manifold for security, cracks, dents and leakage.

(16) Fire detection loop for proper routing, dents, chafing, clamps for rubber insert, security of couplings and proper safeties.

(17) Fuel hoses for chafing, proper routing and leaks.

(18) Fire extinguishing manifold for dents, security and outlet holes for obstructions.

(19) Accessory drive gear box and transfer gear box for leaks, security and broken attachment lugs or brackets.

(20) Engine external fuel, oil pressure, oil scavange lines for leaks, dents, cracks, chafing and security.

(21) Engine anti-ice valves and connecting lines for air leaks, security, dents and chafing. Anti-icing (engine to cowling doors) rubber seals for deterioration.

(22) Engine cooling air lines for leaks, dents, chafing and security.

(23) Exhaust thermocouple harness for dents, chafing, cracks and security.

(24) Combustion and turbine section for hot spots, bulges and cracks. Turbine guard for cracks and security.

(25) Engine mounts and attaching hardware for security, cracks and proper safeties.

(26) Bleed air extraction duct for cracks, and attaching points and connections for evidence of hot air leakage.

(27) Fuel oil cooler for evidence of leakage.

(28) Oil tank and line for dents, cracks, leakage and security.

(29) Engine accessories for damage, overheating, security leaks and proper safeties.

(30) Gang drain block and lines for security, leaks and obstructions.

(31) Engine cowling for dents, cracks, and seals for damage. Bonding straps for security. Latches and hinges for security and cracks.

(32) Fuel heater for security, leaks (air & fuel), and outlet for obstructions.

(33) Fuel control linkage for worn components, freedom of operation, security and proper safeties.

(34) Starter/generator cooling air muff for security, air leaks, and cowling seal for condition.

(35) Aft fan frame for cracks, distortion, struts for dents, cracks, and loose or missing fasteners.

(36) Fan rotor blades for nicks, dents, cracks, and security.

(37) Fan rotor for free rotation.

(38) Fan rotor exit vanes for nicks, cracks, dents, and distortion.

(38A) Exit guide vane outer band abradable coating for missing patches of material. If corrective action is required, refer to G.E. Maintenance Manual, SE-187.

(39) Engine tail pipe for dents, bulges, distortion, or hotspots.

(40) EPR probe for security and obstructions.

(41) Engine shroud (firewall) for security, missing or loose fasteners, cracks, and hot spots.

(42) Oil tank for security, leaks, and cracks.

(43) Engine inhibit switches (engine stall warning system) for security, damage, and support bracket for cracks.

(44) Tail pipe for dents, bulges, distortion, or hot spots. Seal for condition.

(45) Electrical power supply . . . . . on

(46) Circuit breakers:

(a) LH and RH ENG MASTER . . . . . out

(b) LH and RH FUEL BOOST PUMP. . . . . out

(c) LH and RH VIBRATORS. . . . . out

(d) EQUIP GRD BLO-SECOND CONT. . . . . out

(47) Engine ignition system check as follows:

(a) Switches:

LH and RH ENGINE MASTER. . . . . on
IGNITION . . . . . .CONTINUOUS
AIRSTART . . . . . OFF

(b) LH power lever . . . . . .IDLE
Check for audible firing of continuous igniter plugs and IGNITION light on.

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(c) IGNITION switch . . . . . NORM
Check igniter firing ceases.

(d) AIRSTART switch . . . . . LEFT HAND ON
Check both igniters for firing.

(e) LH power lever. . . . . .CLOSE
Check that igniter firing ceases.

(f) IGNITION switch . . . . . CONTINUOUS

(g) RH power lever. . . . . IDLE
Check for audible firing of continuous igniter plugs and IGNITION light on.

(h) IGNITION switch . . . . . NORM
Check igniter firing ceases.

(i) AIRSTART switch . . . . . .RIGHT HAND ON
Check both igniters for firing.

(j) RH power lever. . . . . .CLOSE
Check that igniter firing ceases.

(k) Switches:

LH and RH ENGINE MASTER . . . . . .OFF
AIRSTART. . . . . .OFF

(48) Circuit breakers:

(a) LH and RH ENG MASTER. . . . . in

(b) LH and RH FUEL BOOST PUMP . . . . . in

(c) LH and RH VIBRATORS . . . . . in

(d) EQUIP GRD BLO-SECOND CONT . . . . . in

(49) Electrical power supply. . . . . .off

(5) Airplane . . . . . .secure as desired

MAJOR CHANGE

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INSPECTION/CHECK (Continued)

REMOVAL OF ENGINE

The following equipment will be necessary to remove the engine and support it after removal.

1. Engine lifting sling (RI P/N 13785-135D or G.E. P/N 2C5327), engine attach fittings (RI P/N 13785-139D).
2. Hydraulic Hoist.
3. Engine transportation stand or dolly (RI P/N 13785-132G or G.E. P/N 2C5347).

To remove the engine, proceed as follows:

NOTE

Cap all open lines and electrical plugs as they are disconnected. All disconnect points are painted international orange.

1. Position all switches OFF or in their normal airplane parked position and remove electrical power from airplane.
2. Remove engine cowling doors.
3. Remove engine aft fairing (refer to "Removing Engine Aft Fairing").
4. Disconnect bleed air duct at fitting adjacent to engine air bleed line fitting to fuel heater and bleed air line (forward inboard side of engine above de-icing valve) to pylon.
5. Disconnect fuel heater bleed air line.
6. Disconnect fire extinguishing line at reducer fitting.
7. Disconnect fuel supply line at engine.
8. Disconnect engine power lever control coupling. Coupling is installed between engine and pylon.
9. Disconnect EPR flex line at pylon.
10. Disconnect electrical ground cables (2) at forward engine mount.
11. Disconnect fan speed sensor electrical plug. Electrical plug is adjacent to EPR flex line.
12. Disconnect electrical plugs (6) at forward end of pylon. Check complete installation to ensure that electrical wiring and lines are disconnected.
13. On airplanes with fan sync system on left engine, disconnect trimmer assembly (synchronizer unit) from the fuel control. The flex cable and components stay with the aircraft.
14. Attach lifting sling and adapter to engine. Hook sling to lifting hoist.
15. Using hoist or crane, take weight of engine off engine mounts.
16. Remove nut, washer and bolt (MY205501302) from lower end of main engine mount. (See Figure 71-00-1.) Ensure that electrical bonding is free.
17. Remove nut, washers, and bolt (NAS464-PT-46) from forward engine mount. (See Figure 71-00-1.) Disconnect electrical bonding from mount to damper.
18. Remove nut, washer and bolt (MY205501519) from aft engine frame fitting. Ensure that electrical bonding cable is free.
19. Remove pin (MY205501306) from main engine mount top fitting. Remove nut (MY205501311) from top of main engine mount. (See Figure 71-00-1.) Remove electrical bonding from engine mount.
20. Ensure that all lines and electrical leads are disconnected, then slowly lower engine until top fitting of main engine mount is free. Then move engine outboard and down.
21. Remove pin from link (MY2055013850) at bottom fitting on aft fan forward flange. Remove link fitting. Install engine dolly support fitting where link was removed.
22. Install engine dolly support fitting in forward frame of aft fan housing (opposite fitting installed in step 21).
23. Lower engine into transportation dolly. Install bolts in the two aft fittings. Install clevis bolt in forward engine mount.
24. Lower hoist and remove engine sling.

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INSPECTION/CHECK (Continued)

Inspect — Top Main Engine Mount Pin Bushing

NOTE

The Top Main Engine Mount Pin Bushing, which is located in the Fan Inlet Aft Cowl Support Ring, can wear excessively over a period of time. During an engine change, while the engine is removed, it is recommended that this bushing be inspected for excessive wear (aircraft without SOK-19 only).

If bushing is excessively worn, replace as follows:

1. While engine is removed from aircraft — remove old bushing. See Figure 71-00-3.
2. Prior to engine installation — install new bushing slug (P/N 234-22509-3).
3. Install engine on aircraft per Maintenance Manual Chapter 71.
4. With engine in place — align top "C-Ring" mount hole with top engine mount hole.
5. Using "C-Ring" and engine mount hole as a drill guide — drill bushing slug (.379 max./.374 min.).
6. Install Top Main Engine Mount Pin per Maintenance Manual.

Installation of Engine

The following equipment is necessary to install the engine:

1. Engine lifting sling (RI P/N 13785-135D or G.E. P/N 2C5327) and engine attach fittings (RI P/N 13785-139D).
2. Hydraulic Hoist.

NOTE

Ensure that all caps and plugs are removed from electrical plugs and lines just prior to connecting each line or electrical lead.

To install the engine, proceed as follows:

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(1) Ensure that all switches are in their OFF or airplane parked position and that electrical power is removed from the airplane.

NOTE: The numbers in parenthesis correspond to the items called out in figure 71-00-1. Under each callout, the applicable part numbers of the bolt, washer, etc., are also depicted. This illustration should be used in conjunction with the installation procedures to ensure that the proper installation of all assemblies is accomplished.

(2) Install the upper main engine mount fitting (1) on top of fitting of fwd fan housing. Install spacer (2) on inbd side of fitting. Measure gap on opposite side of fitting. If gap is 0.059 in. or greater, install a MY205501309 shim (laminated) and adjust as required to obtain 0.017 in. max gap. If measured gap is less than 0.059 in., install a MY205501308 spacer. Check for 0.017 in. max gap. (See figure 71-00-1.) Install pin (4) through fitting and secure with two screws (5). Ensure that electrical bonding is installed under bottom screw.

(3) Position a suitable crane or hydraulic hoist over engine.

(4) Attach lifting sling (RI P/N 13785-135D) to engine. Connect sling hook to hoist.

CAUTION: To prevent possible damage to compressor or fan housing do not raise engine stand with engine.

(5) Slowly raise engine until weight is removed from engine stand.

(6) Remove the pins that secure engine to stand.

(7) Raise engine from stand. Remove the engine to stand adapter fittings on forward flange of aft fan housing.

(8) Install the lower link (6) in bottom fitting of main engine mount. Install pin (7) through link fitting. Secure with two screws. Ensure that electrical bonding is installed under screws.

(9) Slowly raise engine to mount assembly. Align bolt (1) with the top mounting hole of the main engine mount. Install washer (8), and nut (9). Do not tighten nut at this time.

(10) Install (head fwd) pin (10) through the top main engine mount fitting, then install washer and nut on pin. Do not tighten nut.

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(11) Align link (6) between the lower mounting hole bracket of the main engine mount. Install bolt (11) (head fwd) electrical bonding (12), washer (13) and nut as shown. Do not tighten nut.

(12) Align and install bolt (14) in aft engine mount fitting. Install electrical bonding (15) and nut (17). Do not tighten nut.

(13) On S/N 370-1 and -7, align and install bolt (18) with link (19) and damper, washers (20, 24 and 26) shims (21, 22 and 23) in fwd engine mount as shown on the applicable drawing of figure 71-00-1. *Install nut (25). Torque nut (25) to 8-11 ft-lbs.* Install cotter pin.

NOTES: • If a replacement engine is installed, steps (14), (15), and (16) are accomplished during the engine alignment procedure.

• If new damper is being installed, adjust rod end to obtain 5.875 (±0.090) in. (center of mounting holes) free length.

(14) On S/N 380-1 and subs, install bolt (18A) with link (19A) and damper, washers (20A), shims (21A and 22A) and laminated washer (23A).

(15) On S/N 380-1 and subs, remove laminations from washer (P/N 56C121018-7) as required to obtain 0.010 in. max gap on assembled assembly.

(16) On S/N 380-1 and subs, the link should be square with the axis of the bolt; however, the link may slope from 0.00 to 0.12 in. as shown on applicable drawing of figure 71-00-1. The link position may be obtained by changing the location of the laminated washer (P/N 56C121018-7) and the AN960-716L washers. Install nut (24A) and torque to 8-11 ft-lbs. Install cotter pin.

NOTE: On RH engine installation, prior to installing nut (24A) *install fuel line support bracket.*

(17) Torque aft engine mount nut (17) to 8-11 ft-lbs. Install cotter pin.

(18) Torque top center engine mount nut (9) to 69-79 ft-lbs. Install cotter pin.

(19) Torque nut (on bolt 11) on bottom main (center) engine mount fitting to 21-25 ft-lbs.

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NOTE: After pin nut (10) is tightened, the pin will have a fore and aft movement of approximately 1/16 to 1/8 in.

(20) Tighten nut on pin (10). Check complete engine mount installation for security and applicable safeties.

(21) Align engine as necessary. (Refer to 71-00-4.)

(22) Connect all hoses, anti-icing lines, electrical plugs and controls. (Refer to Removal of Engine for connecting points.)

(23) Rig power lever as required. (Refer to chapter 76.)

(24) Check complete engine installation for security, chafing hoses or lines, missing fasteners, loose hardware and applicable safeties.

(25) Install engine aft fairing and engine cowling.

(26) Perform an operational check of engine. (Refer to 71-00-4.)

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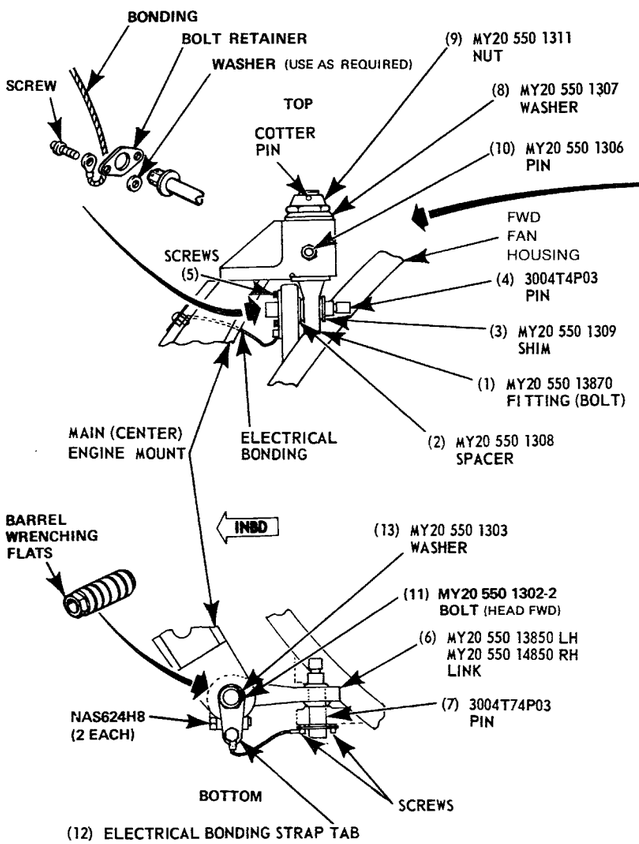
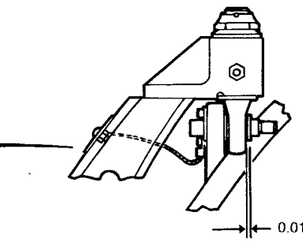
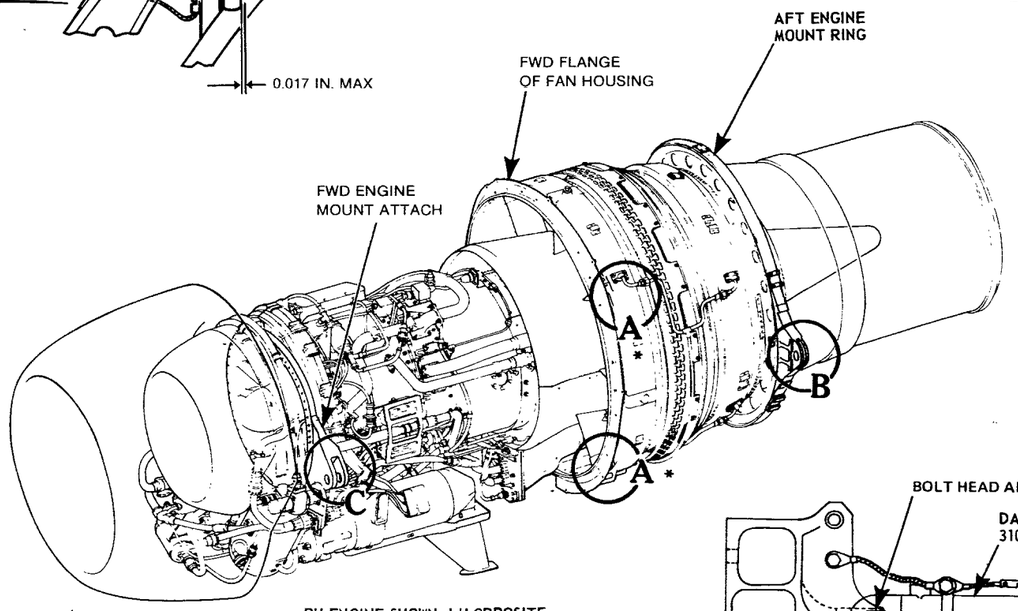


VIEW **A** MAIN (CENTER) ENGINE MOUNT INSTALLATION



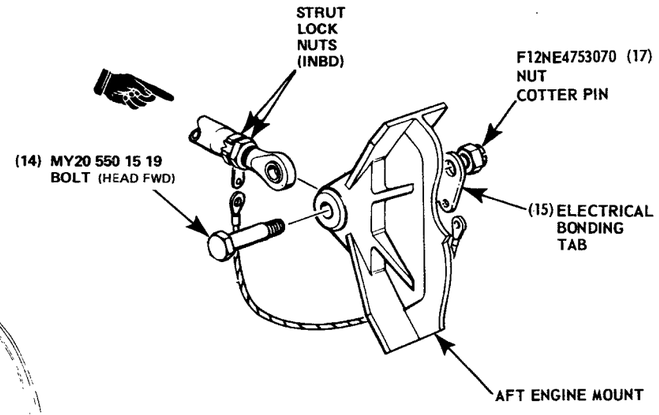


RH ENGINE SHOWN, LH OPPOSITE

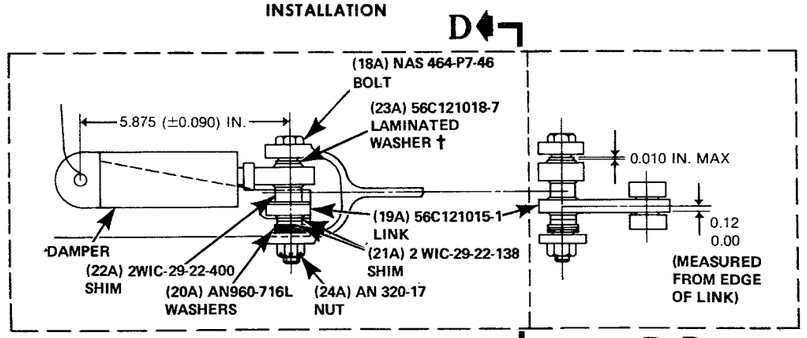
* RH ENGINE MAIN MOUNT FITTINGS DEPICTED. LH ENGINE FITTINGS ARE ON THE OPPOSITE SIDE OF FAN HOUSING.

† REMOVE LAMINATIONS AS REQUIRED TO OBTAIN 0.010 IN. GAP OF ASSEMBLED ASSEMBLY

NOTE MOST PART NUMBERS DEPICTED ARE INTERCHANGEABLE WITH NA380 PART NUMBERS. REFER TO ILLUSTRATED PARTS CATALOG NA-69-424 FOR INTERCHANGEABLE ITEMS.

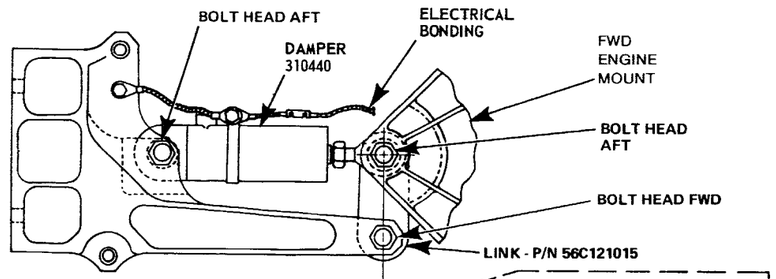

STRUT LOCK NUTS (INBD)
F12NE4753070 (17) NUT COTTER PIN
(14) MY20 550 15 19 BOLT (HEAD FWD)
(15) ELECTRICAL BONDING TAB
AFT ENGINE MOUNT

VIEW **B** AFT ENGINE MOUNT INSTALLATION



VIEW **C** FWD ENGINE MOUNT INSTALLATION S/N 380-1 AND SUBS

VIEW **D-D**



VIEW **C** FORWARD ENGINE MOUNT INSTALLATION NA370-1 AND-7

Engine Mount Fittings
Figure 71-00-1

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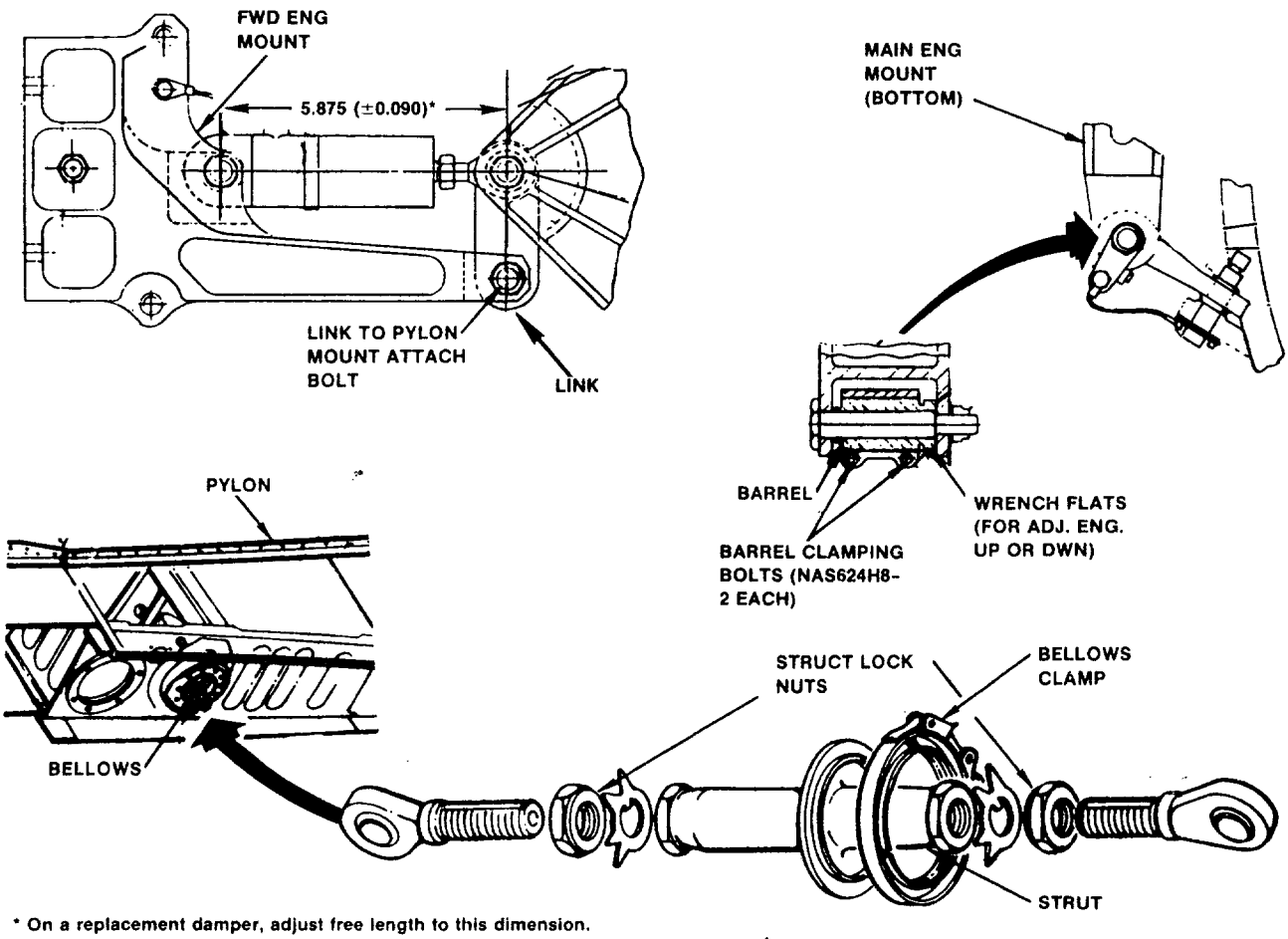
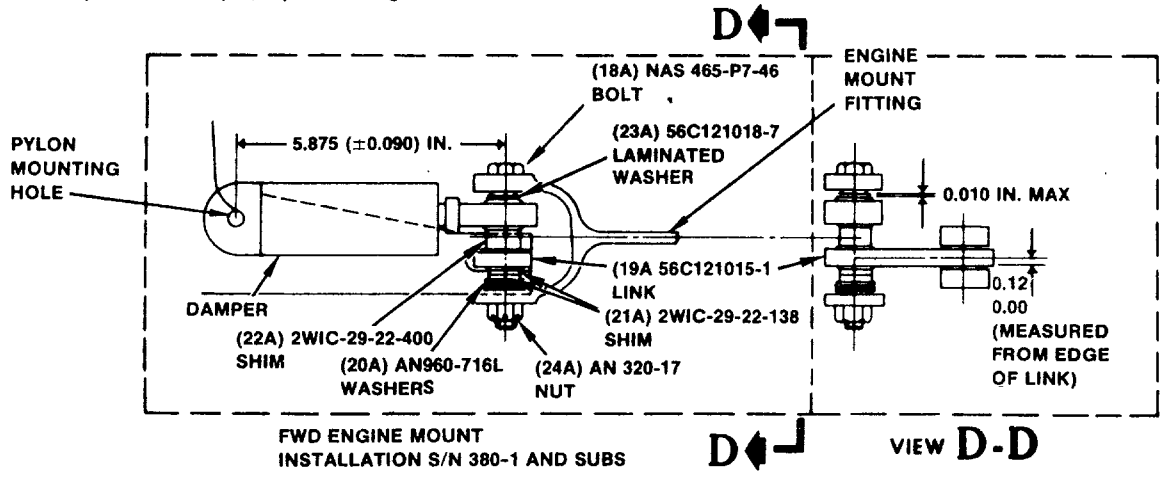
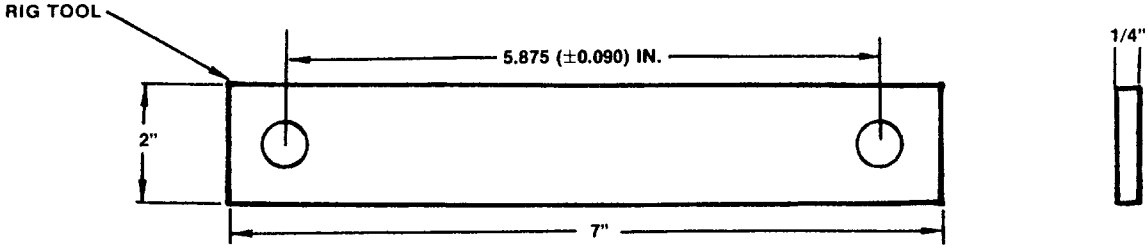


* On a replacement damper, adjust free length to this dimension.





Engine Alignment
Figure 71-00-2

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INSPECTION/CHECK (Continued)

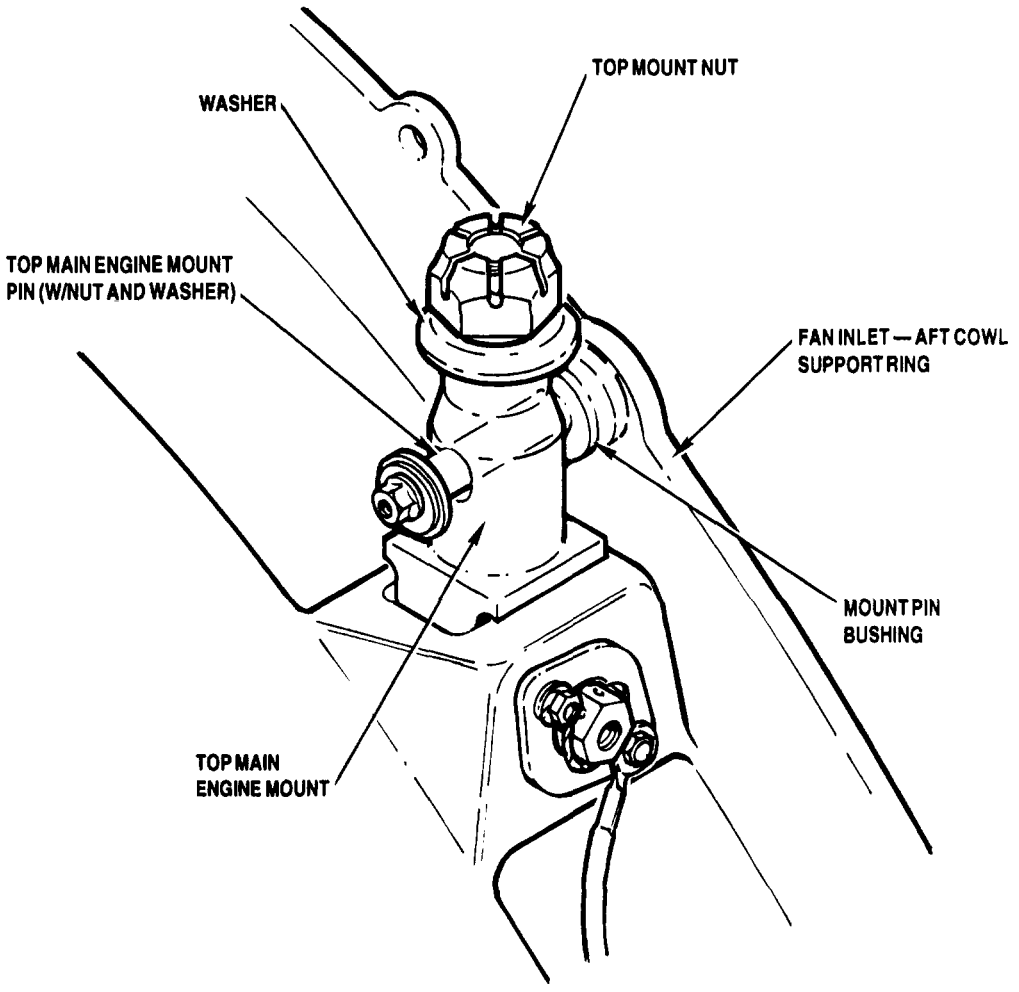

Figure 71-00-3 — Inspect Top Main Engine Mount Pin Bushing

NEW

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1. Aligning Engine

A. To align engine, proceed as follows:

(1) Fabricate a rig tool as depicted in illustration. Remove damper and link on forward engine mount. With link removed, install rig tool in place of damper. Do not install nuts on bolts (see figure 71-00-2).

(2) Remove pylon access door 5 or 56.

(3) Loosen strut bellows clamp in pylon (rear mount).

(4) Loosen inboard and outboard strut rod end locknuts.

(5) Turn strut as required to obtain a slip fit of bolt through the engine front mount fitting and rig tool. This provides a yaw angle of 4° ± 0° .06' and the 5.875 inch dimension.

(6) Tighten strut locknuts securely.

(7) Bend tab of lockwasher over strut locknuts.

(8) Tighten strut bellows clamp.

(9) Remove bolts from engine mount fitting and install link. Reinstall bolts.

(10) Loosen the two bolts (barrel clamping bolts - P/N NAS624H8) on the bottom fitting of the main (center) engine mount.

(11) Turn the barrel (using the wrench flats on barrel) as required to obtain a slip fit of bolt through link and pylon mount. This establishes a 2° incidence angle. Install and tighten nut on pylon mount to link attach bolt.

(12) Tighten barrel clamping bolts - P/N NAS624H8 and torque to 3.5 to 4 foot-pounds. Lockwire bolts.

(13) Remove rig tool and link bolts.

(13A) Install damper attach bolt (inboard bolt) and bonding cable. Install cotter key.

(14) Install bolt (18A) with link (19A) and damper, washers (20A), shims (21A and 22A) and laminated washer (23A).

(15) Remove laminations from washer (P/N 56C121018-7) as required to obtain .010 maximum gap on assembled assembly.

(15A) Install link to pylon mount attach bolt.

(16) The link should be square with the axis of the bolt; however, the link may slope from .00 to .12 inch as shown on the illustration.

The link position may be obtained by changing the location of the laminated washer (P/N 56C121018-7) and the AN960-716L washers. Install nut (24A) and torque to 8-11 foot-pounds. Install cotter pin.

NOTE: On RH engine installation, prior to installing nut (24A) install fuel line support bracket.

(17) Check all mounting points for security and proper safeties.

(18) Install and align engine aft fairing (refer to 71-10).

2. Engine Starting Procedures

A. The following procedure shall be performed whenever any of the following conditions exist:

(1) Prior to the initial start of a new, overhauled or reinstalled engine.

(2) Prior to the first start following installation of any major (fuel control, fuel pump, pressurizing and dump valve, fuel-oil cooler, etc.) fuel system component.

(3) Prior to the first start following installation of a new or reinstalled starter generator.

(4) Prior to starting an engine with a type of auxiliary power unit which has not previously been used with the airplane.

B. Preoperational Check of Engine

To perform a preoperational check (cold run) of engine, proceed as follows:

(1) Open the lower cowl door.

(2) Service engine oil system (refer to Chapter 12).

(3) Remove the protective covers from the engine inlet and exhaust ducts. Check for foreign material at engine and fan air inlets and exits and at the starter-generator cooling air inlet and exit. Check that inlet duct, engine compartment and tailpipe are free of oil and fuel. Remove all objects on the wing upper surfaces and ground around the aircraft which may be drawn into the air inlet.

(4) Install chocks at wheels and connect external power source to external power start receptacle.

NOTE: When an external power unit is used for starting the engines, the unit must be capable of providing not less than 24-volt dc and 800 amperes, and not more than 30-volt dc and 1200 amperes.

(5) Check that fuel density knob on the engine fuel control is indexed for the type fuel being used. If adjusted, ensure that knob is locked.

(6) Pull out "IGN & BOOST" circuit breaker applicable to engine being checked. Ensure that all other applicable circuit breakers are engaged (pushed in).

(7) (Deleted)

(8) Position drain pan under main fuel filter, then remove drain plug from cap of main fuel filter.

(9) Position "ELECTRIC MASTER," "BATTERY" and applicable "ENGINE MASTER" switches to ON.

(10) Allow fuel to drain until a solid stream of fuel emits from the fuel filter drain plug, then, position "ELECTRIC MASTER," "BATTERY" and "ENGINE MASTER" switches to OFF. If a solid stream of fuel does not discharge from filter, energize the applicable boost pump.

(11) Install and tighten the fuel filter drain plug. Lockwire plug. Manually drain fuel collector tank.

(12) Position "ELECTRIC MASTER," "BATTERY" and applicable "ENGINE MASTER" and GENERATOR switches to ON.

CAUTION: ENSURE THAT FUEL SHUTOFF VALVE IS OPEN. THE ENGINE FUEL CONTROL IS LUBRICATED BY FUEL, AND IT WILL BE DAMAGED IN OPERATION IF THE FUEL CONTROL VALVE IS CLOSED.

(13) Momentarily depress "PUSH TO START" switch for engine being checked. Pull out starter button. Check for proper engine rotation (counter-clockwise as viewed from the front). Listen for noises in engine which may indicate rubbing of compressor, turbine rotor and/or fan wheel, or any other malfunction.

NOTE: Certain normal noises are associated with engine starts such as ticking, scraping, and rubbing because of the design of the number 1-stage compressor rotor blades, backlash in the accessory gear train. Any departure from these normal noises are to be investigated.

(14) Depress applicable starter switch. At 10% engine rpm (minimum), advance the power lever to IDLE position. Ensure that the following items do not exist:

(a) If 10% engine rpm is not attained, the engine is not safe to start.

(b) If engine does not reach 12% rpm within 12 seconds, pull out starter button and investigate problem.

CAUTION: OPERATION OF THE ENGINE STARTER IS LIMITED TO THE FOLLOWING LIMITS: 1ST START - 30 SECONDS MOTORING MAXIMUM 2-MINUTE WAIT BEFORE 2ND CYCLE; 2ND START - 30 SECONDS MOTORING MAXIMUM 5-MINUTE WAIT BEFORE 3RD CYCLE; 3RD START - 30 SECONDS MOTORING MAXIMUM 20-MINUTE WAIT BEFORE 4TH CYCLE, 4TH START - 30 SECONDS MOTORING MAXIMUM 1½-HOUR WAIT.

(15) Check that engine oil pressure is within 1 to 10 psig.

NOTE: If no indication of oil pressure is obtained, and if this is the only discrepancy noted during the motoring operation, the prestart check may be continued. However, during the actual engine start, the oil pressure should be monitored for a pressure indication.

(16) Check that fuel flow rises to at least 200 pph, but does not exceed 350 pph when engine rpm reaches 10% rpm. If fuel flow of 200 pph is not obtained, or if fuel flow exceeds 350 pph, retard power lever to OFF and pull the "PUSH TO START" button out. If fuel flow exceeds the stated limits, investigate as required.

CAUTION: IF FUEL FLOW DOES NOT DROP TO ZERO IMMEDIATELY FOLLOWING POSITIONING OF POWER LEVER TO OFF AND RELEASE OF STARTER SWITCH, THE ENGINE IS NOT SAFE TO START.

(17) If fuel flow is within limits, retard power lever to OFF, and pull out "PUSH TO START" button. Ensure that fuel flow immediately drops to zero.

(18) During engine coastdown, listen for unusual noise from the rotating components. Check the engine front frame, mainframe and turbine areas for unusual roughness. Normal rotational noise consists of clicking compressor, turbine and fan blades, gear noise and slight rubbing at the turbine and fan shrouds.

(18A) Manually drain combustion chamber fuel collector tank.

(19) Repeat steps (13) through (18). After the second cold start, continue to motor the engine after the power lever has been positioned to OFF for the time remaining within the starter operating limits. This will purge the combustion and exhaust sections of fuel and vapor. Two cold starts are required to ensure that preservation oil is cleared from the various engine systems.

(20) Following completion of cold starts, push in "IGN & BOOST" circuit breaker and move "ENGINE MASTER" and GENERATOR switches to OFF.

(21) Check the engine for the following items:

(a) Fuel and oil leaks.

(b) Engine drains for proper operation. Drain combustion chamber fuel collector tank.

(c) Tailpipe and thrust reverser for puddles of oil or fuel (remove as necessary).

(d) Interstage bleed valve discharge ports for oil accumulation (remove as necessary).

(22) Disconnect external electrical power from airplane.

(23) Determine starting characteristics of engine when powered only by airplane batteries as follows:

(a) Check that airplane batteries are fully charged.

(b) Ensure that power levers are at OFF.

(c) Position battery and applicable ENGINE MASTER and GENERATOR switches at ON.

(d) Depress "PUSH TO START" switch. Pull out start switch button when engine rpm stabilizes. Record length of time at which engine rpm stabilizes. Do not exceed starter limitations.

NOTE: The starter must be capable of turning the engine rotor to at least 12% rpm within 12 seconds for ambient temperatures down to 0°F in order to achieve normal engine starts.

(24) Following satisfactory completion of preoperational check, position all switches to OFF or the normal airplane parked position.

(25) Check entire engine for fuel and oil leaks, proper saftey of all components, and ensure that all foreign objects are removed from engine and around airplane.

3. Engine Prestart Inspection and Operational Check of Engines

A. To perform an engine prestart inspection, proceed as follows:

(1) Open the lower cowl door.

(2) Check for foreign material at engine and fan air inlet and exits and at the starter-generator cooling air inlet and exit. Check that inlet duct, engine compartment and tailpipe are free of oil and fuel. Remove all objects on the wing upper surfaces and ground around the aircraft which may be drawn into the air inlet.

WARNING: TO PREVENT INJURY TO PERSONNEL, ENSURE THAT INTAKE AND EXHAUST AREAS ARE CLEAR BEFORE ENGINE OPERATION.

CAUTION: TO AVOID OVERSTRESS AND POSSIBLE DAMAGE TO FAN BLADES, DO NOT OPERATE ENGINE ABOVE IDLE SPEED WITH NACELLE DOORS OPEN OR REMOVED.

(3) Check that engine oil system is properly serviced.

(4) Check engine ignition system as follows:

(a) Connect external electrical power to airplane. Manually actuate and hold the external power monitor switch open. This will prevent the engine from rotating during the ignition check.

(b) Station a man in vicinity of engine to audible check ignitor plug operation.

(c) Check that both power levers are positioned at OFF.

(d) Position "ELECTRIC MASTER," "BATTERY" switches at ON. Check that "AIRSTART" switch is OFF.

(e) Open the RH and LH "FUEL-BOOST PUMP" circuit breakers.

(f) Position the applicable "ENGINE MASTER" switch to ON and the "IGNITION" selector switch to CONTINUOUS.

(g) Advance the engine power control lever to the IDLE position. The START IGNITION switch "ON" red light should come on indicating the ignition ignitor plugs are firing. Check for sound of spark.

(h) Return the engine power control lever to the CUT-OFF position. The engine START IGNITION "ON" red light should go out, the engine ignitor plugs will stop firing.

(i) Advance the engine power control lever to the IDLE position. Place the IGNITION selector switch to the "Normal" position. Depress the engine start button. The START IGNITION switch "ON" red light should come on, check for sound of the engine ignition firing.

(j) Return the engine power control lever to the CUT-OFF position. (This will permit the engine start button to release.) The engine START IGNITION "ON" red light should go out, the engine ignition plugs will stop firing.

(k) Advance the engine power control lever to the IDLE position. Place the engine AIRSTART selector switch to the "LEFT HAND ON" position. The engine ignitors should fire with the start button ignition "ON" light illuminated.

(l) Place the AIRSTART switch to the "OFF" position. The engine ignitors should stop firing, and the start button ignition "ON" light should go out.

(m) Position the engine MASTER switch to the "OFF" position. Return the engine power control lever to the CUT-OFF position.

(5) Position and hold fire detector "PUSH TO TEST" button. Check that warning lights in each handle of engine fire pull handle comes on. Release switch. Warning lights should go out.

(6) On airplanes with APU, depress APU FIRE test switch and check that switch cap and master caution APU fire module comes on. Release switch. Warning lights should go out.

(7) Position "BATTERY" and "ELECTRIC MASTER" switches to OFF.

(8) Engage RH and LH "FUEL BOOST PUMP" circuit breakers.

(9) Check that fuel density control is indexed as to the type of fuel being used. If adjusted, ensure that knob is locked.

(10) Release external power monitor switch in external power receptacle.

(11) Close the lower cowl door if engine is to be operated above idle rpm.

B. Operational Check of Engines

To ensure satisfactory engine starts, the external electrical power unit must be capable of providing a minimum of 24 volts dc and 800 amperes at the external power receptacle.

To ensure that the airplane is self-sufficient, at least two engine starts should be made utilizing the batteries. However, the battery start should be made only on engines which have demonstrated its capability of attaining 10% rpm during the starting cycle. The fuel collector tank should be manually drained after each start.

(1) Connect external power source to external power start receptacle. Ensure that all applicable circuit breakers are engaged, thrust reversers stowed, and the "THRUST REVERSER CONTROL" switch is OFF.

(2) Move LH and RH DC GENERATOR switches to ON.

(3) Move "ELECTRIC MASTER" switch to ON.

(4) Move battery switches to ON.

(5) Move "INVERTER" switch to NO. 1.

(6) Move applicable "ENGINE MASTER" switch to ON.

NOTE: The engine master switches open the main fuel shutoff valve and arms the power lever-operated switches which control ignition and fuel boost pump operation.

- On airplanes with an APU installed and operating, when the left engine master switch is turned on, the APU generator will be removed from the dc essential bus as indicated by the illumination of the "APU GEN OFF" indicator light.

(7) Depress engine starter switch momentarily. Advance power lever to IDLE when 10% rpm is obtained.

CAUTION: IF LIGHT-OFF HAS NOT OCCURRED WITHIN 10 SECONDS AFTER THROTTLE IS ADVANCED, ABORT START BY RETARDING THROTTLE TO CUTOFF. MOTOR ENGINE FOR 20 SECONDS TO REMOVE AS MUCH FUEL AS POSSIBLE; THEN PULL OUT STARTER SWITCH. WAIT AT LEAST 30 SECONDS FOR COMBUSTION CHAMBERS TO DRAIN BEFORE ATTEMPTING ANOTHER START. IF ENGINE IS NOT MOTORED, WAIT AT LEAST 2 MINUTES FOR FUEL TO DRAIN FROM COMBUSTION CHAMBER. REMOVE ANY PUDDLES OF FUEL WHICH MAY HAVE ACCUMULATED IN TAIL PIPE. MANUALLY DRAIN FUEL COLLECTOR TANK AFTER EACH START.

NOTE: Throttle advancement energizes the ignition and fuel boost pumps admitting fuel to the engine. The ignition system is armed when the starter button is depressed.

- The starter should accelerate the engine to 12% rpm in 12 seconds and should motor the engine at a steady-state speed of 12% to 15% rpm.

(8) If fuel flow exceeds 350 pph before a light-off occurs, retard the power lever to cut-off and pull out the starter button.

WARNING: IF AN ENGINE FIRE OCCURS, FOLLOW PROCEDURES GIVEN FOR MOTORING ENGINE AND FOR A FIRE IN THE ENGINE.

CAUTION: EXCESSIVE FUEL FLOW COULD CAUSE A HOT START.

(9) If fuel is less than 200 pounds per hour when light-off occurs, or exhaust gas temperature reaches 750°C and is still increasing at a rapid rate during the start, retard power lever to OFF position and pull out starter button. If exhaust gas temperature limit of 910°C is exceeded, refer to Figure 71-00-501, exhaust gas temperature limitations table.

NOTE: Insufficient fuel flow will result in improper burning and unsatisfactory acceleration, ending in a "hung" or false start.

- Retarding the throttle to the OFF position de-energizes the ignition system, closes the fuel control shutoff valve, and de-energizes the fuel boost pump.

- Pulling out the starter switch de-energizes the starter.

(10) After engine rotation has stopped, wait at least 2 minutes to permit fuel to drain from engine before attempting another start.

NOTE: Do not exceed starter limitations.

(11) If a successful light-off occurs, the starter should continue to operate, assisting the engine to accelerate until the start controller relay opens and causes the starter to be de-energized and the ignition to stop. The starter switch automatically releases (pops up) at about 42% rpm, and the ignition-on light goes out. The applicable generator switch will also move to OFF.

WARNING: IF THE STARTER BUTTON RELEASES BEFORE 42% RPM IS REACHED, OR THE GENERATOR SWITCH REMAINS AT ON, BE ALERT TO DETECT THE POSSIBILITY OF A HUNG START AS EVIDENCED BY A RAPID INCREASE IN EXHAUST GAS TEMPERATURE WITH LITTLE OR NO INCREASE IN ENGINE SPEED.

NOTE: If the ignition-on light remains on after idle rpm is reached, pull out starter button and position generator switch to OFF.

- The engine should accelerate to idle speed in about 40 seconds from the time the starter switch is depressed.

(12) Note that engine oil pressure rises immediately on light-off. If a rise of at least 5 psi is not obtained by the time engine speed has reached 45% rpm, abort start.

(13) At idle rpm, position the generator switch to ON.

NOTE: This brings the generator on the line and restores the cabin air conditioning system to operation. The generator caution lights should be extinguished.

WARNING: IF THE POWER LEVER IS INADVERTENTLY RETARDED TO THE OFF POSITION DURING ENGINE OPERATION, THERE WILL BE AN IMMEDIATE FLAME-OUT, WHICH WILL BE IMPOSSIBLE TO CATCH REGARDLESS OF HOW QUICKLY THE THROTTLE IS ADVANCED. DO NOT TRY TO REGAIN ENGINE OPERATION BY ADVANCING POWER LEVER, SINCE RAW FUEL WILL BE SPRAYED INTO THE HOT ENGINE AND CREATE A DANGEROUS CONDITION. TO RESTART ENGINE, WAIT AT LEAST 2 MINUTES FOR ENGINE TO STOP ROTATING. DRAIN FUEL FROM COMBUSTION CHAMBER FUEL COLLECTOR TANK; THEN REPEAT STARTING PROCEDURE.

(14) Check for indication of fan speed rotation. If fan is not rotating, shut down engine and investigate.

(15) Deleted

(16) If the RH engine is to be started with the LH engine running, throttle the LH engine to idle rpm before depressing starter switch for RH engine.

NOTE: Always make certain that running engine (whether right or left) is throttled to idle rpm before starting other engine. Generator output of the running engine (operating above idle) plus battery power may impose damaging loads on the starter shaft of the second engine.

- On airplanes with an APU installed and operating, starting RH engine will result in the APU shutting down automatically approximately 30 seconds after the RH engine power lever is advanced to IDLE. When this occurs, position the APU "BLEED AIR" and turbine "START-RUN-OFF" switches to OFF and the "EGT TRIM" rotary switch to START after shutdown. Check to ensure the APU "GENERATOR" switch is at OFF. Note that the "UNDERSPEED," "LOW OIL PRESSURE" and "LOW FUEL PRESSURE" APU indicator lights illuminate during shutdown.

(17) Make engine idle speed check.

4. Idle Speed Check

A. The engine parameters may be recorded on figure 71-00-508. This form may be used to record all temperatures, pressures, etc., and may be filed with the aircraft maintenance records. To perform an idle speed check, proceed as follows:

(1) With the power lever at IDLE and THRUST REVERSER control switch at OFF, check that engine idle speed is 46% (±1%) rpm. If this speed is not obtained, check that power lever is at IDLE detent and that pointer on fuel control shaft is at 13° to 15° range.

(2) If the idle rpm is not within the specified range, perform the following adjustment procedure after the maximum rpm adjustment is accomplished.

(a) If idle rpm is low, increase rpm by turning the IDLE trim adjustment screw clockwise until rpm is within limits.

(b) If idle rpm is high, decrease rpm by turning the IDLE trim adjustment screw counterclockwise until rpm is within limits.

NOTE: Each click of the trim screw will result in an approximate 55 rpm speed change. To eliminate the effect of backlash, preload the internal linkage of the fuel control by completing all adjustments of idle speed in a counterclockwise direction with 5 clicks. For example: (1) to increase engine speed 1 click, turn the adjusting screw 6 clicks clockwise, then 5 clicks counterclockwise; (2) to decrease engine speed 1 click, turn the adjusting screw 4 clocks clockwise, then 5 clicks counterclockwise. Record idle rpm after adjustment.

(3) Monitor fan speed for a positive rpm indication. If fan speed is not indicated, shutdown engine and investigate.

(4) Actual exhaust gas temperature: 675°C maximum.

(5) Fuel flow: 400 to 550 pounds per hour.

(6) Lubrication oil pressure: 5 to 20 psi.

NOTE: If oil pressure increases or decreases 10 psi or more from normal stabilized value (at any rpm), investigate cause.

(7) Lubrication oil pressure fluctuation: ±2 psi maximum.

(8) Lubrication oil temperature: 185°C maximum.

(9) Check engine for fuel, oil, and exhaust gas leaks. A maximum total fuel drainage, from engine fuel system components of 134 cc per minute per engine is allowed. A maximum total oil leakage from the oil tank vent relief valve and accessory drive gearbox seals of 10 cc per hour and 3 drops in 5 minutes respectively (per engine) is allowed during engine operation.

(10) Inspect thrust reverser for air leaks.

(11) Shut down engine(s) and close and secure engine cowling and applicat access doors (if open). Re-start engine(s) if required to perform remaining operational checks.

5. (Deleted)

6. Take-off Check

A. The engine idle speed check should be performed prior to this check. Refer to figure 71-00-504 for maximum EPR value. Check takeoff thrust as follows:

(1) Position aircraft into the prevailing wind. Wind velocity should not exceed 10 knots.

CAUTION: TO AVOID OVERSTRESS AND POSSIBLE DAMAGE TO FAN BLADES, DO NOT OPERATE ENGINE ABOVE IDLE SPEED WITH NACELLE DOORS OPEN OR REMOVED.

(2) Close all access and cowling doors. Install wheel chocks.

(3) Compute maximum allowable EPR. (See figure 71-00-504.)

(4) Position engine and windshield anti-icing switches at OFF and the air conditioning switches at OFF.

(5) Start engine. Allow engine to stabilize at idle rpm.

WARNING: HOLD BRAKES APPLIED EVERY TIME ENGINE IS RUN AT SPEEDS ABOVE IDLE. ADVISE PERSONNEL WORKING AROUND AIRPLANE THAT ENGINE WILL BE RUN AT HIGH POWER SETTINGS.

CAUTION: DO NOT EXCEED COMPUTED MAXIMUM ALLOWABLE EPR.

- DO NOT EXCEED RPM SPEED LIMITATIONS.
- DO NOT EXCEED EXHAUST GAS TEMPERATURE LIMITS.

(6) Slowly advance throttle until computed maximum allowable EPR is obtained. Record all engine parameters.

NOTE: Advancement of the throttle at this point does not constitute an acceleration check. If instability is encountered, retard power lever to 65% rpm, and after engine has stabilized, again slowly accelerate to computed maximum allowable EPR.

(7) Allow engine to stabilize at take-off thrust, then check the following:

CAUTION: ENGINE OPERATION AT TAKE-OFF THRUST IS LIMITED TO 5 MINUTES (CONTINUOUS OPERATION).

(a) Engine speed should be below 101.2% (16,700) rpm. Fan speed should be below 105% (9,000) rpm.

CAUTION: IF ENGINE SPEED EXCEEDS 101.2% (16,700) RPM, RETARD POWER LEVER TO IDLE AND SHUT DOWN ENGINE. IF FAN SPEED EXCEEDED 113.2% RPM, REPLACE ENGINE. IF FAN SPEED WAS LESS THAN 113.2% RPM, DETERMINE ENGINE OR FAN SERVICEABILITY AS PER FIGURES 71-00-505 OR 71-00-506. IF ANY ABNORMAL CONDITION IS EVIDENT, REPLACE ENGINE.

(b) The exhaust gas temperature should stabilize below 741°C. Record EGT.

CAUTION: IF MAXIMUM OVERTEMPERATURE LIMITS ARE EXCEEDED, REFER TO EXHAUST GAS TEMPERATURE LIMITATIONS TABLE, FIGURE 71-00-501.

(c) Record fuel flow.

(d) Engine oil pressure should be as depicted in figure 71-00-502. If oil pressure is out of limits, refer to G. E. CF700 Maintenance Manual #SEI187.

NOTE: Oil pressure fluctuation: ±2 psi maximum.

(e) Engine oil temperature - 185°C maximum.

(f) If oil pressure reading is out-of-limits, refer to General Electric Engine Maintenance Manual, SEI 187.

(8) Check hydraulic pressure indicators. Normal hydraulic system pressure should be 2650 to 3200 psi. Auxiliary hydraulic system pressure should be 2800 to 3600 psi.

(9) Place "CABIN AIR" and "COCKPIT AIR" selector switch at BOTH ENG position and sniff-check cabin conditioning air. If fuel or oil fumes are detected, investigate source.

7. Maximum Speed Check

A. To perform a maximum speed check, proceed as follows:

(1) Position aircraft into the prevailing wind (maximum 10 knots) and ensure that all access doors and engine cowling are closed.

(2) Install wheel chocks and position engine and windshield system switches OFF. Position cockpit and cabin air conditioning controls OFF.

(3) Start engine and allow to stabilize at idle rpm.

CAUTION: LIMIT ENGINE OPERATION TO 5 MINUTES (CONTINUOUS OPERATION) DURING THIS CHECK.

(4) Slowly advance power lever to maximum position (physical stop of fuel control lever) and check the following items:

(a) Engine rpm 100.7% to 101.2% rpm. Fan rpm should not exceed 105% rpm.

(b) Exhaust gas temperature should not exceed 741°C.

(c) See figure 71-00-502 for oil pressure range.

(d) Oil temperature should not exceed 185°C.

(e) Record all engine parameters.

(5) Retard power lever to IDLE. If engine rpm was over 101.2% rpm, adjust the maximum speed adjustment as follows:

WARNING: DO NOT ADJUST MAXIMUM SPEED WITH ENGINE OPERATING ABOVE IDLE RPM.

(a) Adjust fuel control MAX speed as required to obtain a maximum rpm of 100.7% to 101.2%.

NOTE: One click of the MAX adjusting screw will change engine speed by approximately 30 rpm.

- To eliminate the effect of backlash, preload the internal linkage of the fuel control by accomplishing the adjustment in a counterclockwise direction with 5 clicks. For example to decrease engine speed 1 click (30 rpm), turn the MAX speed adjustment screw 4 clicks clockwise and then 5 clicks counterclockwise.

(6) Repeat steps (4) and (5) until engine rpm is within prescribed limits.

(7) Retard power lever to IDLE position. Idle rpm should be 46% (±1%). If idle rpm is within limits, refer to Idle Speed Check and adjust in accordance with step (2).

(8) If idle rpm was changed, recheck maximum speed adjustment.

(9) Perform thrust reverser operational checks. (Refer to Chapter 78.)

(10) Shut down engine if required.

8. Overspeed Governor Check

A. To perform overspeed governor operation, proceed as follows:

(1) Open the lower cowl doors.

WARNING: TO PREVENT SERIOUS INJURY TO PERSONNEL, USE EXTREME CAUTION WHEN PERFORMING MAINTENANCE ON ENGINE WITH THE ENGINE OPERATING.

(2) Start engine and operate at idle rpm.

(3) Position the overspeed governor lever aft until the rocker arm seats firmly against the stop.

(4) Close the lower cowl doors.

CAUTION: DO NOT EXCEED 101.2% ENGINE RPM.

- DO NOT EXCEED 741°C EXHAUST GAS TEMPERATURE.
- DO NOT EXCEED 185°C ENGINE OIL TEMPERATURE.

(5) Slowly advance the power lever to the fuel control maximum stop. Engine rpm should not exceed 90.5% (±1%) rpm. Record engine rpm.

(6) If 90.5% (±1%) rpm is exceeded, replace overspeed governor and repeat procedures for new governor.

(7) Retard power lever to idle and open the lower cowl doors.

(8) Position the governor speed lever to the full forward position.

(9) Shut down engine and close lower cowl doors.

9. Acceleration and Deceleration Checks

A. After it has been determined that engine operates properly up to and at Take-off Thrust, make the following acceleration check:

(1) Compute take-off thrust EPR.

(2) With engine stabilized at idle rpm, advance throttle rapidly (one second or less) to take-off thrust EPR setting. Watch tachometer and make sure that acceleration time does not exceed 8 seconds when EPR reaches the established setting. The exhaust gas temperature may overshoot 741°C momentarily. If this occurs, see figure 71-00-501 for overtemperature limits.

NOTE: During the preceding check, EPR may overshoot past the take-off thrust value and then settle back down. This overshoot is normal and will not be considered as part of the acceleration time.

(3) Ensure that engine rpm does not exceed 101.2% rpm. Fan speed should not exceed 105% rpm. If either of these engine parameters are exceeded, see figures 71-00-505 or 71-00-506 for allowable limits.

B. Check engine for proper deceleration as follows:

(1) Move throttle rapidly from take-off thrust to IDLE. Make sure deceleration time does not exceed 15 seconds.

(2) Upon reaching idle speed, make sure fuel flow does not exceed 500 pounds per hour.

(3) Shut down engine if required.

10. Engine Suction Fuel Feed Check

A. To perform engine suction fuel feed check, proceed as follows:

(1) Defuel each wing tank to a total of 800 pounds maximum.

(2) Check that fuel crossfeed and tank selector switch is at normal.

(3) With the right engine operating at idle rpm, disengage the "RH FUEL BOOST PUMP" circuit breaker.

(4) Slowly advance power lever to take-off thrust setting (see figure 71-00-504).

(5) Operate engine at take-off thrust for 2 to 3 minutes. Check that no engine malfunctioning occurs as evidenced by a decrease in rpm, fuel flow, engine cycling or surging.

(6) Retard right engine power lever to IDLE. Close "RH FUEL BOOST PUMP" circuit breaker.

(7) Perform step (3) through (6) for left hand engine. The "LH FUEL BOOST PUMP" circuit breaker is used for this check.

(8) Shut down engine if required.

(9) Fuel airplane as required.

11. Engine Shutdown

A. To perform engine shutdown, proceed as follows:

(1) Retard throttle to IDLE position and hold for 2 to 5 minutes.

CAUTION: DO NOT SHUT DOWN ENGINE BY USE OF AIRFRAME FUEL SHUTOFF VALVE, AS ENGINE-DRIVEN FUEL PUMP IS LUBRICATED BY FUEL AND WILL BE DAMAGED BY OPERATION WITHOUT FUEL.

(2) Move power lever rapidly past idle stop to full cutoff position.

(3) Note that fuel flow immediately drops to zero. Exhaust gas temperature and engine rpm should decrease.

(4) Observe "coast down" for any unusual noises (scraping or grinding) at low speeds and especially during final rotations.

NOTE: A normal engine coastdown time is approximately 30 seconds. A slight rotor kickback at the moment the rotor stops turning usually indicates that rotor is free.

(5) Move "ENGINE MASTER" switch to OFF after engine has stopped rotating.

(6) Move all switches to OFF.

WARNING: IF THERE IS ANY EVIDENCE OF FIRE WITHIN THE ENGINE AFTER SHUTDOWN, FOLLOW THE PROCEDURE GIVEN FOR A FIRE IN THE ENGINE.

(7) Check entire engine for fuel, exhaust and oil leaks. Check thrust reverser for proper mating of cowling, blocker doors, etc.

(8) Check engine oil level if required. This should be performed within one-half hour after shut down.

12. Motoring Engine

A. To motor engine, proceed as follows:

(1) Check that "LH ENGINE - MASTER," "RH ENGINE - MASTER," "BOOST PUMP - LH FUEL" and "BOOST PUMP - RH FUEL" circuit breakers are in.

(2) Move "ELECTRIC MASTER" switch to ON.

(3) Move battery and generator switches to ON.

(4) Check that power levers are in OFF position.

(5) Move "ENGINE MASTER" switch for selected engine to ON. This opens the airframe fuel shutoff valve.

CAUTION: OPERATION OF THE ENGINE STARTER IS LIMITED TO THE FOLLOWING LIMITS: 1ST START - 30 SECONDS MOTORING MAXIMUM 2-MINUTE WAIT BEFORE 2ND CYCLE; 2ND START - 30 SECONDS MOTORING MAXIMUM 5-MINUTE WAIT BEFORE 3RD CYCLE; 3RD START - 30 SECONDS MOTORING MAXIMUM 20 MINUTE WAIT BEFORE 4TH CYCLE, 4TH START - 30 SECONDS MOTORING MAXIMUM 1½-HOUR WAIT.

(6) Depress selected engine starter button.

(7) To stop motoring, pull out starter button.

(8) Position all switches to OFF.

13. Fire in One Engine Nacelle

A. If fire occurs in one engine nacelle, proceed as follows:

(1) Ensure that "ELECTRIC MASTER" and "BATTERY" switches are ON, then pull out proper "FIRE PULL" handle.

(2) Position extinguisher discharge switch to EXT NO. 1. The extinguisher agent will be discharged into right or left engine nacelle, depending on which "FIRE PULL" handle was pulled out.

(3) Move applicable power lever to cutoff position.

(4) Move applicable "ENGINE MASTER" switch to OFF.

(5) If fire has been extinguished, return "FIRE PULL" handle to neutral position.

(6) If pod requires a second charge of fire-extinguishing agent, leave "FIRE PULL" handle pulled out and position extinguisher discharge switch to EXT NO. 2.

(7) If fire persists and engine cowling is closed, open cowling doors and direct fire extinguishing agent into engine compartments.

14. Fire in Both Engine Pods

A. If fire occurs in both nacelles, proceeds as follows:

(1) Pull out left "FIRE PULL" handle. Move extinguisher discharge switch to EXT NO. 1. The extinguishing agent will be discharged into the left engine nacelle.

(2) Move left power lever to cutoff position and left "ENGINE MASTER" switch to OFF.

(3) Pull out right-hand "FIRE PULL" handle and position extinguisher discharge switch to EXT NO. 2. The extinguishing agent will be discharged into right engine nacelle.

NOTE: The "FIRE PULL" handles are mechanically interlocked so that only one handle at a time can be pulled out, and consequently, only one nacelle at a time can be sprayed with extinguisher agent.

(4) Move right power lever to cutoff position and right "ENGINE MASTER" switch to OFF. Return right-hand "FIRE PULL" handle to neutral position.

(5) If fire persists in either or both nacelles, open engine access doors and direct fire extinguishing agent into engine compartments.

(6) Perform postfire inspection.

15. Fire in Engine

A. If fire occurs in engine, proceed as follows:

NOTE: Black smoke coming from the tail pipe after shutdown is an indication of burning oil or fuel which will cause damage to the engine.

(1) Move power lever to OFF and "THRUST REVERSER" control switches to OFF if it is not already in this position. Ensure that ELECTRIC MASTER, BATTERY, and applicable GENERATOR switches are ON.

(2) Motor engine with starter for 30 seconds. If fire persists, continue motoring for an additional 60 seconds.

(3) If fire persists after 90 seconds of motoring, apply a fire extinguisher to engine inlet and continue motoring until fire is extinguished.

CAUTION: USE A FIRE EXTINGUISHER ONLY IF MOTORING FAILS TO EXTINGUISHER FIRE.

- DO NOT ENGAGE STARTER UNLESS ENGINE RPM IS BELOW 35% RPM.

(4) After fire is extinguished, pull out engine starter button.

(5) If motoring the engine does not extinguish the fire, proceed as follows:

(a) Retard power lever to OFF.

(b) Position "ENGINE MASTER," "BATTERY" , GENERATOR and "ELECTRIC MASTER" switches to OFF.

(c) Direct fire extinguishing agent into engine tailpipe as necessary.

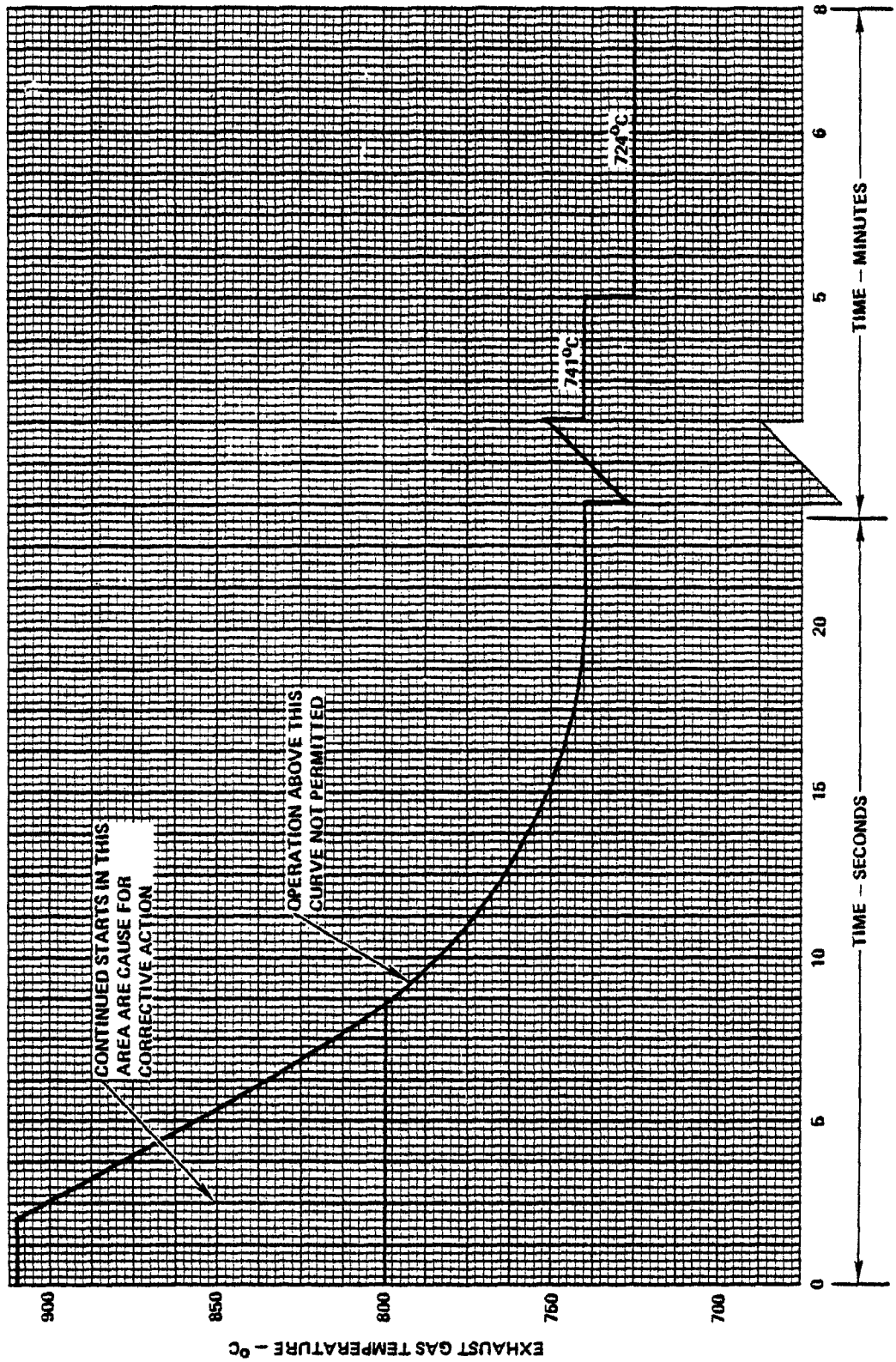
16. Postfire Procedure

A. Normally the fire-extinguishing agent (bromotrifluoromethane) evaporates completely without leaving any deposit, and it is not necessary to perform any cleaning operation in the nacelle sprayed by the agent. However, the agent might puddle, particularly on cold days, and cause corrosion. Therefore, the nacelle should be inspected and, if any puddles of agent are found, they should be blown out with compressed air.

(1) The fire-extinguishing system lines and valves should be serviced (blown out with compressed air). (Refer to Chapter 26.)

(2) The engine, nacelle, thrust reverser and airframe should be inspected for fire damage.

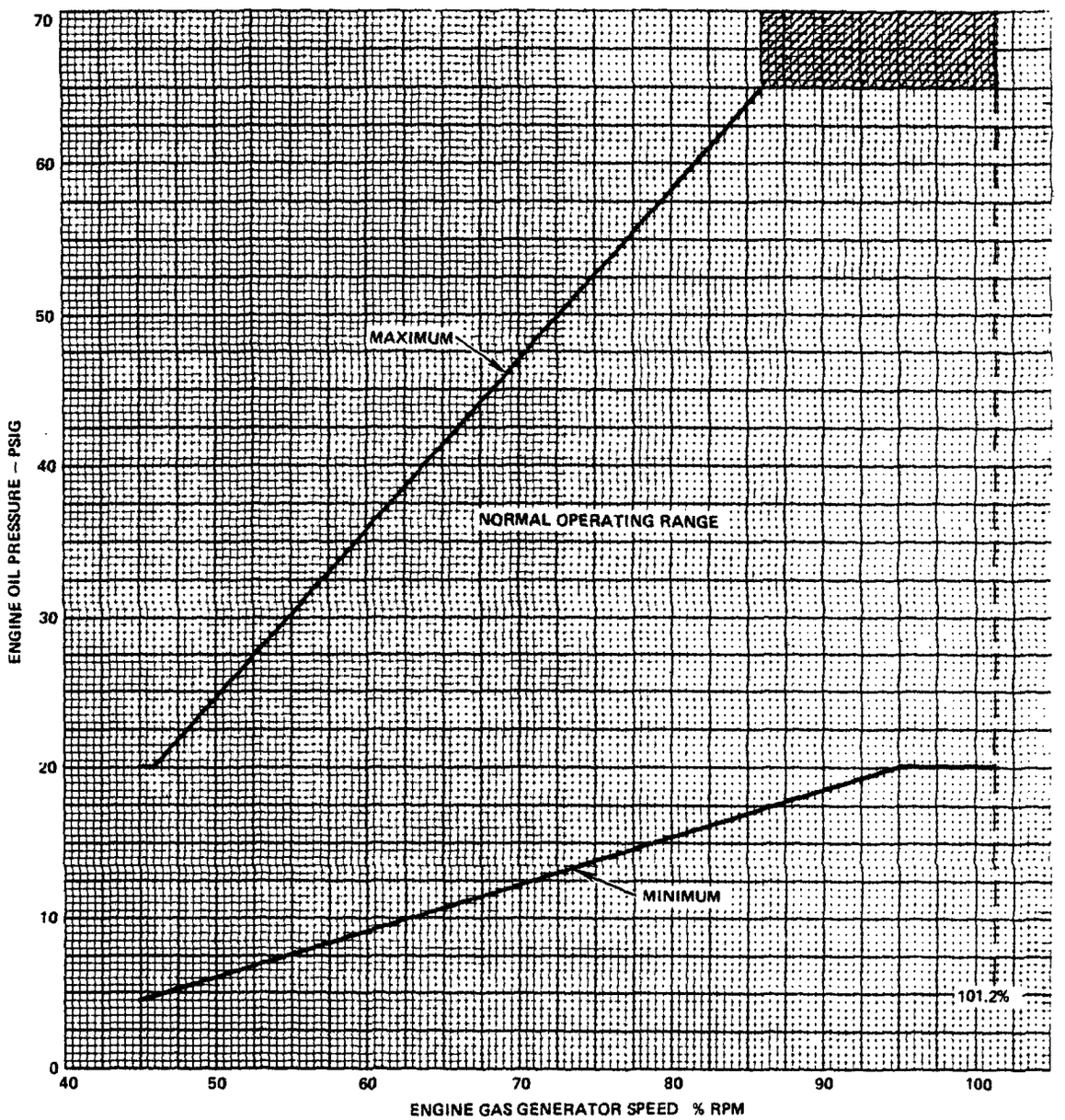
(3) The fire-extinguishing and detection system should be inspected as directed in Chapter 26.



NOTE: IF LIMITS ARE EXCEEDED, REFER TO G.E. CF700 MAINTENANCE MANUAL, SEI-187

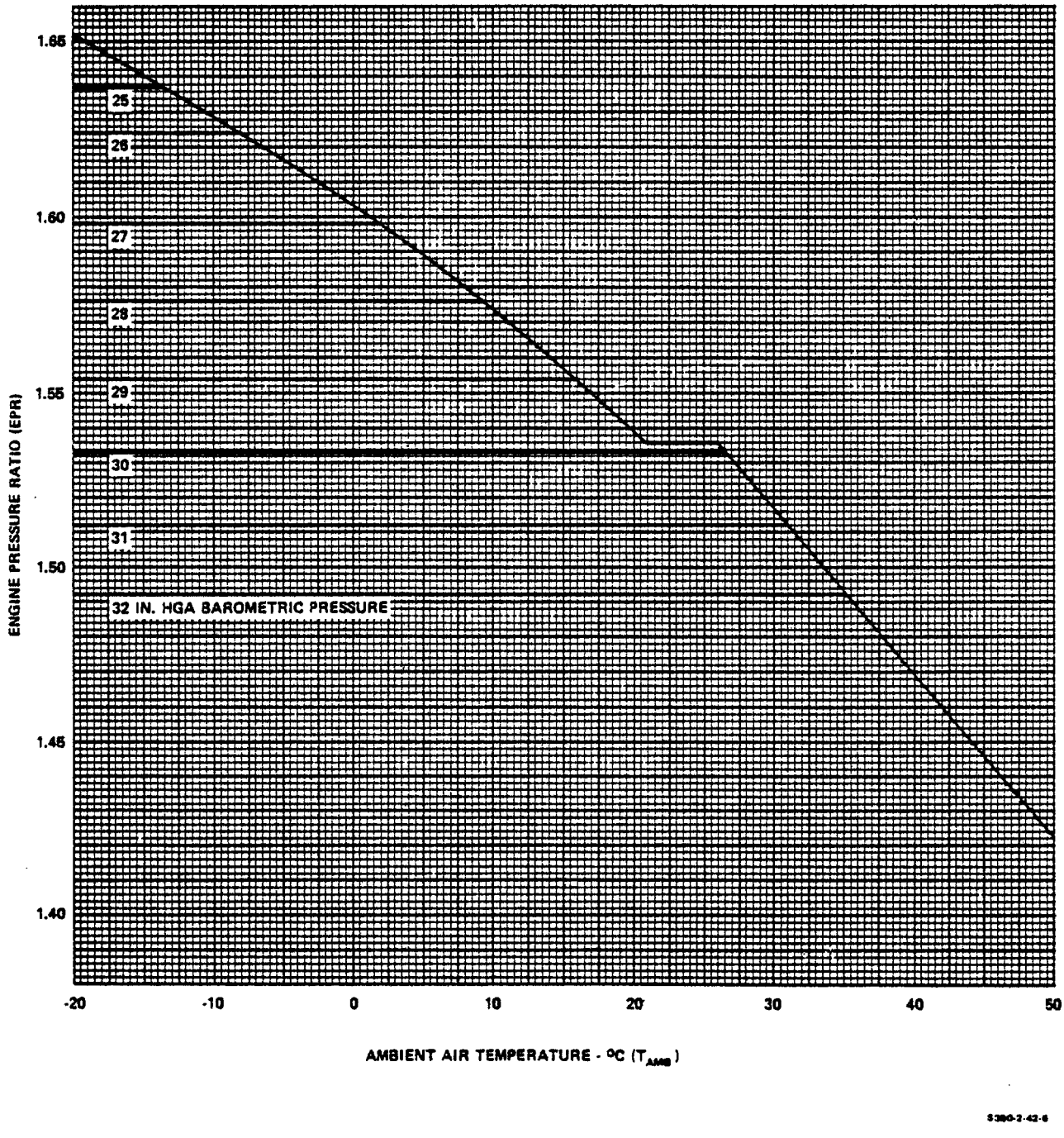
9380-2-43-3

Exhaust Gas Temperature Limits
Figure 71-00-501



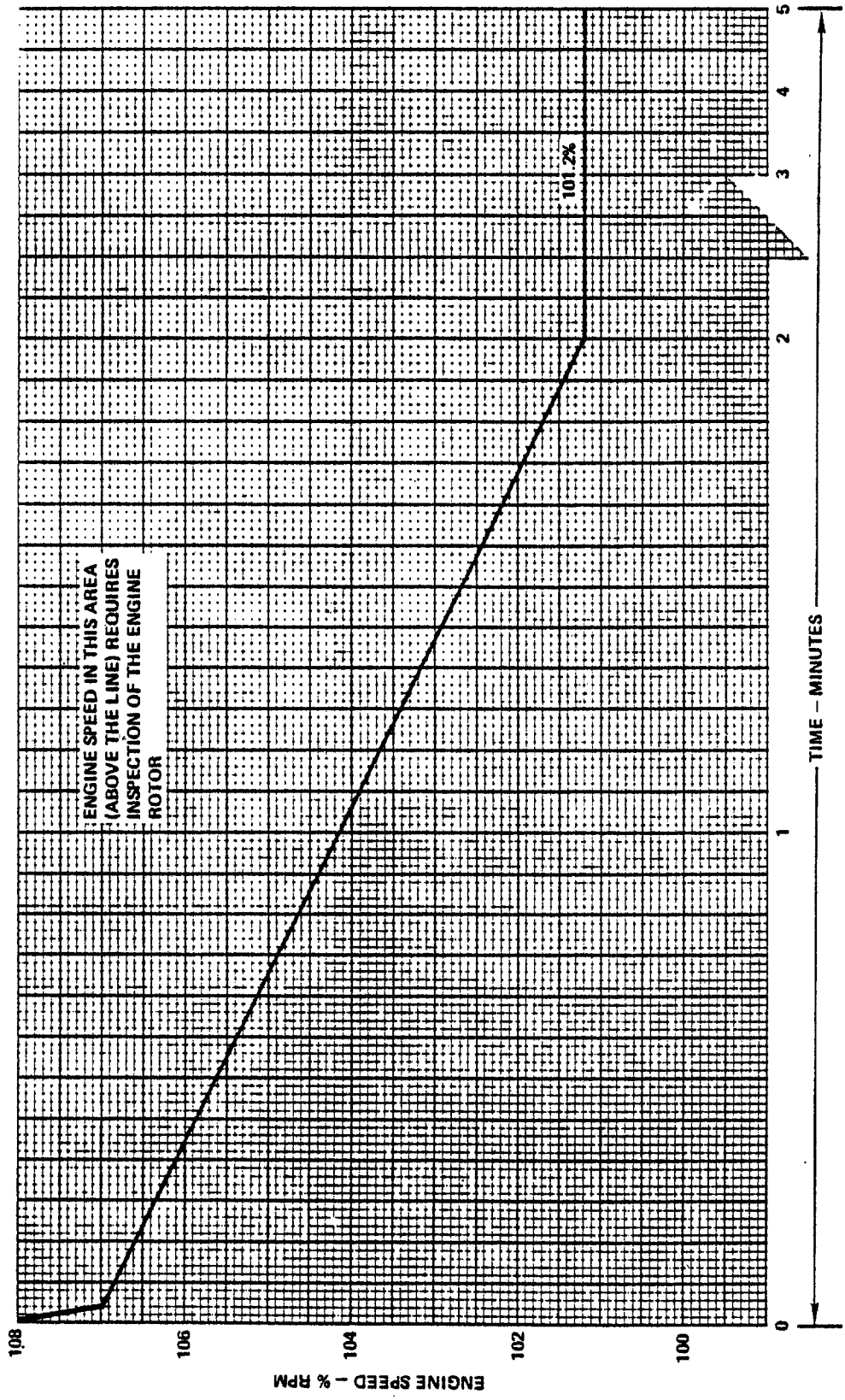
Engine Oil Pressure Limits
Figure 71-00-502
(71-00-4, Page 522 Deleted)

Engine operation with engine air inlet duct, tailpipe and all nacelle doors installed. No inlet screen. Normal power extraction, cabin pressurization and anti-icing systems off. No fuel heating (above 40° F fuel temp). Thrust reversers installed and stowed.



S380-2-42-6

Take-off Thrust EPR Curve
Figure 71-00-504



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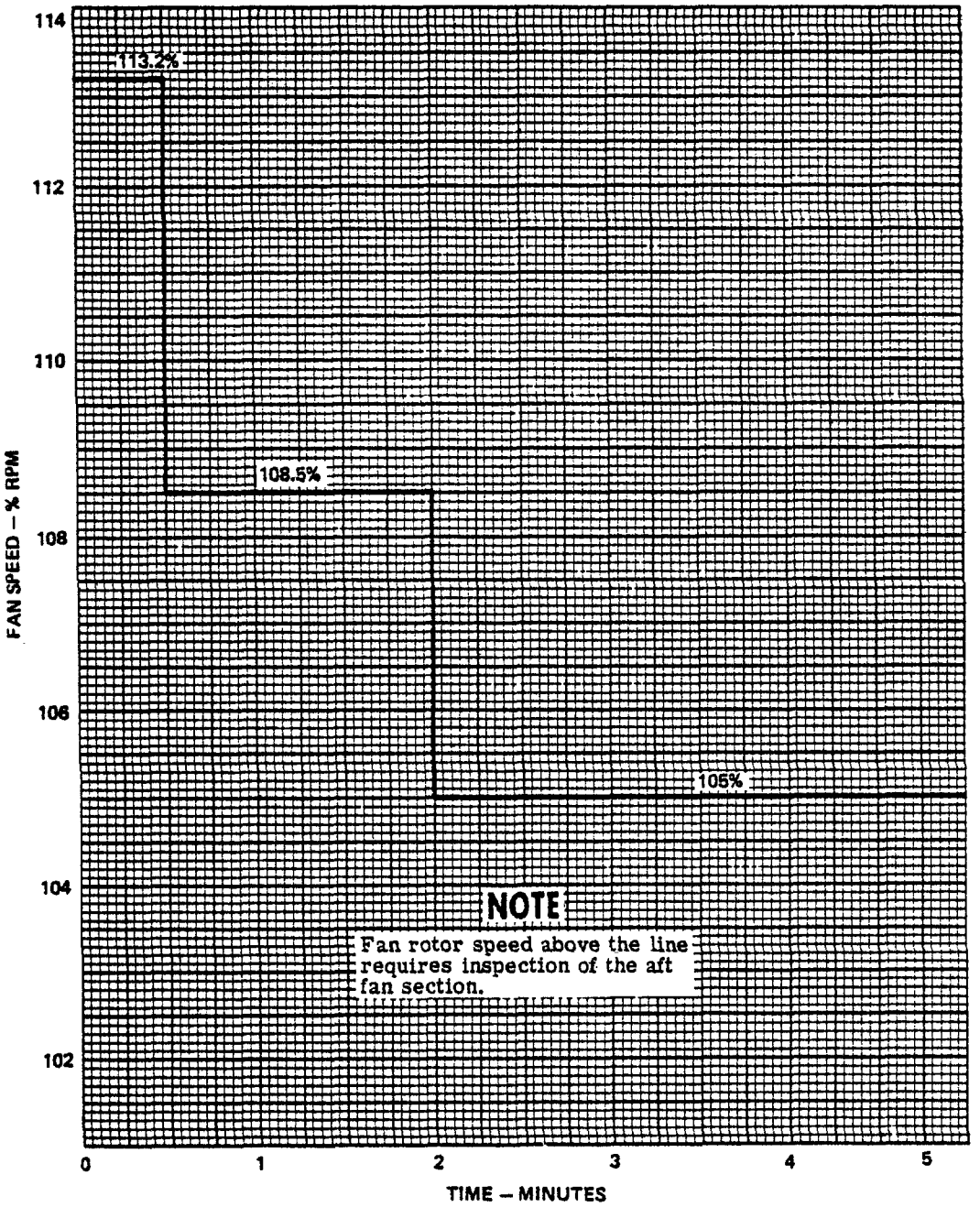
Engine Overspeed Limits
Figure 71-00-505



S380-2-42-8

Aft Fan Overspeed Limits
Figure 71-00-506

(71-00-4, Page 526 Deleted)

Aircraft S/N_____ Engine S/N_____ Date_____

Engine Position LH_____ RH_____

Engine Fuel Control

Manufacturer_____ Serial No._____

Part No._____ Parts List No._____

1. Ambient Temperature_____ °F _____ °C

2. True Barometer (Barometric Pressure) _____ in. HgA.

3. Observed Initial Engine Idle Speed (N_g)_____ percent rpm
(Before Adjustment) (Ref. paragraph 7.3)

Engine Idle Speed (N_g) Limits 45 percent to 47 percent rpm

4. Observed Engine Idle Speed (N_g) _____ percent rpm
(After Maximum Speed Adjustment if required.

5. Observed Fan Speed (N_f) at Idle _____ percent rpm

6. Engine Variable Geometry System

Interstage Bleed Valve Initial Closing _____ percent rpm

Interstage Bleed Valve Fully Closed _____ percent rpm

Interstage Bleed Valve Initial Opening _____ percent rpm

Interstage Bleed Valve Fully Open _____ percent rpm

7. Observed Take-off Thrust Engine Pressure Ratio (EPR) _____ EPR

8. Observed Engine Speed (N_g) _____ percent rpm

9. Observed Fan Speed (N_f) _____ percent rpm

10. Exhaust Gas Temperature (EGT) _____ °C

11. Engine Fuel Flow (PPH) _____ pound/hour

12. Engine Oil Pressure _____ psig

13. Engine Oil Temperature _____ °C

14. Maximum Fuel Control Lever Position (Stop)

(a) Engine Speed (N_g) _____ percent rpm

(b) Fan Speed (N_f) _____ percent rpm

(c) Exhaust Gas Temperature (EFT) _____ °C

(d) Engine Pressure Ratio (EPR) _____ EPR

(e) Engine Fuel Flow (PPH) _____ pound/hour

Sample Engine Trim Record
Figure 71-00-508 (Sheet 1 of 2)

(f) Engine Oil Pressure _____ psig

(g) Engine Oil Temperature _____ ^{o}C

15. Overspeed Governor

(a) Engine Speed (N_g) _____ percent rpm

(b) Fan Speed (N_f) _____ percent rpm

16. Engine Acceleration and Deceleration

(a) Acceleration Time _____ seconds

(b) Deceleration Time _____ seconds

17. Amount of Oil Added

(a) Type and Brand of lubricating oil _____

(b) Oil added after initial engine runs

(1) LH _____

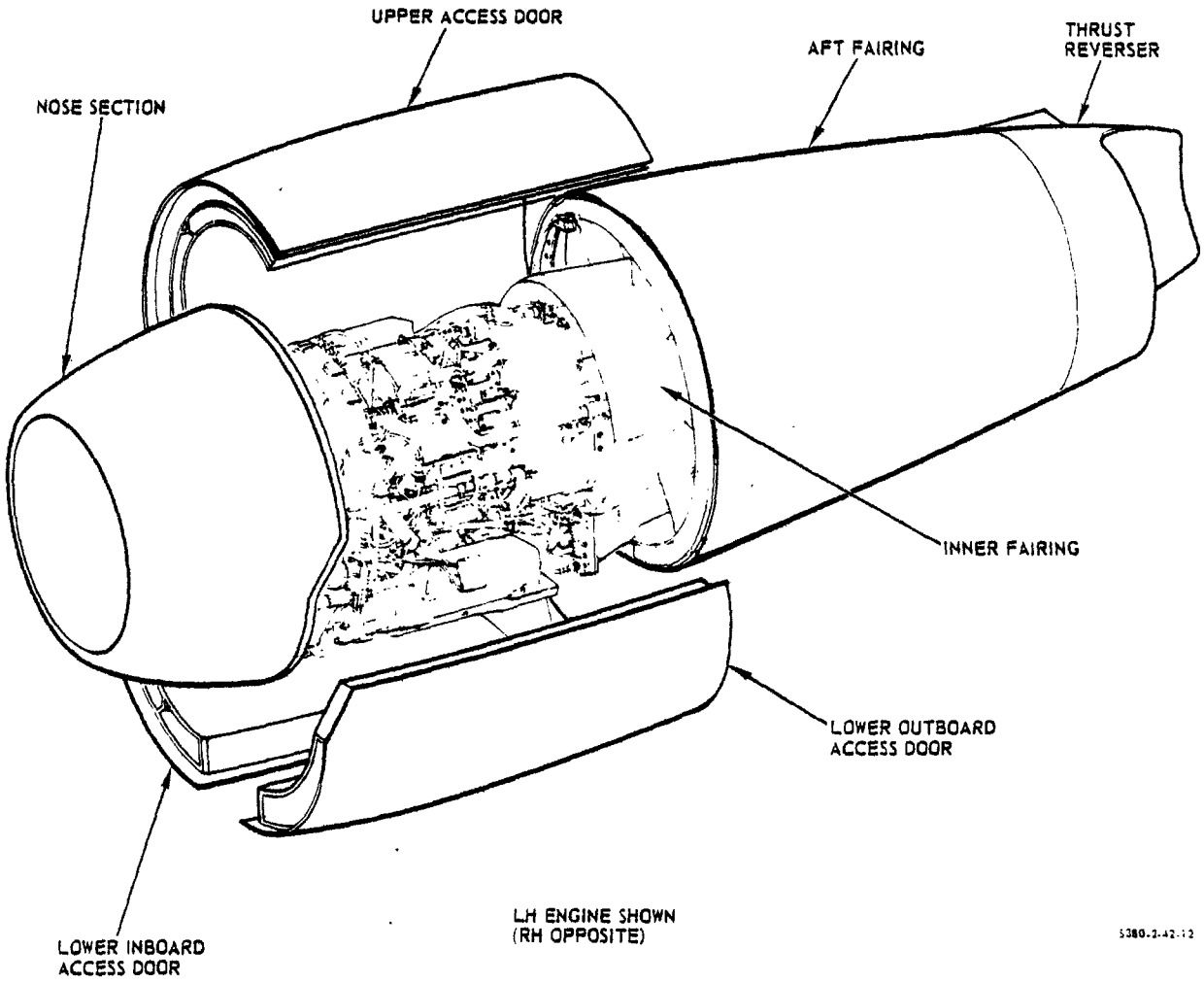
(2) RH _____

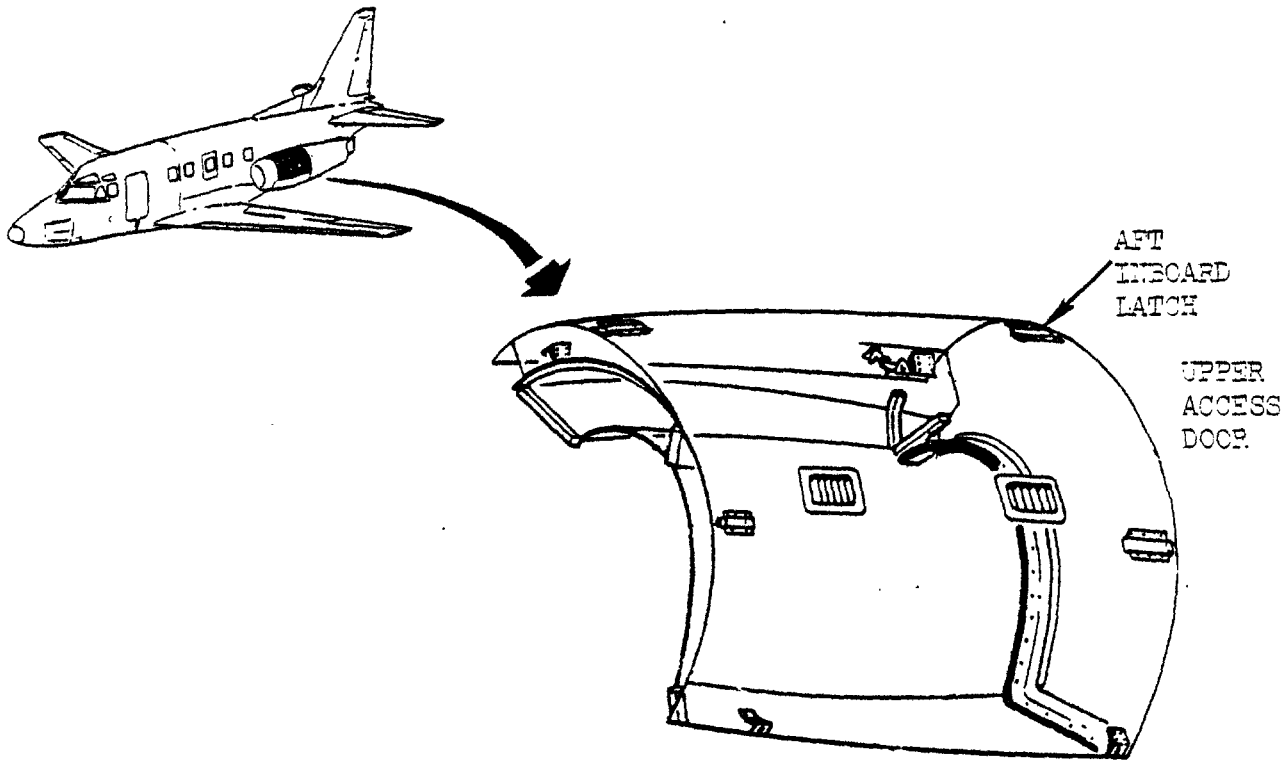
Sample Engine Trim Record
Figure 71-00-508 (Sheet 2 of 2)

ENGINE NACELLES - DESCRIPTION

1. General

For a description of engine nacelles, refer to Chapter 54.





Engine Nacelle Access Doors and Fairings
Figure 71-10-1

1. Removal of Engine Access Doors

A. To remove the engine access doors, proceed as follows:

NOTE: On airplanes equipped with the engine synchronizer system (Chapter 76), disconnect flex cable to actuation prior to removing access doors. When removing the flex shaft, do not turn the inner shaft. This will prevent rigging of the synchronizer system.

(1) Release the two forward and aft latches on lower inboard access door.

(2) Release the two forward and aft latches on lower outboard access door.

(3) Release the four forward and aft latches on upper access door.

(4) Release pin type latch and two hook type latches on lower inboard access door.

(5) Open lower outboard access door far enough to disconnect bonding braids.

(6) Remove "pip" type hinge pins from lower outboard access door hinges and remove door.

(6A) Disconnect access door stop cables.

(7) Rotate lower inboard access door inboard far enough to disconnect bonding braids.

(8) Remove "pip" type hinge pins from lower inboard access door hinges and remove door.

CAUTION: TO PREVENT POSSIBLE DAMAGE TO FUSELAGE WHEN OPENING THE ENGINE UPPER ACCESS DOOR, ENSURE THAT AFT INBOARD LATCH IS CLOSED (LOCKED POSITION) PRIOR TO OPENING ACCESS DOOR. SEE FIGURE 71-10-1.

(9) Raise upper access door far enough to disconnect bonding braids.

(10) Remove "pip" type hinge pins from upper access door hinges and remove door.

2. Installing Engine Access Doors

A. To install engine access doors, proceed as follows:

NOTE: On airplanes equipped with the engine synchronizer system (Chapter 76) connect flex cable to actuator subsequent to installation of the access doors. If the inner shaft was rotated, refer to Chapter 76 for rigging procedures of the engine synchronizer system.

(1) Position upper access door so that hinges on upper access door mate with upper hinges on pylon.

(2) Install pip pin for forward hinge from aft side of hinge. Install pip pin for aft hinge from forward side of hinge.

(3) Connect bonding braids. Open and close door several times to ensure that bonding braids do not interfere with door operation.

(4) Position lower inboard access door so that hinges on access door mate with lower hinges on pylon.

(5) Install pip pin for forward hinge from forward side of hinge (through engine air intake).

(6) Install pip pin for aft hinge from aft side of hinge.

(7) Connect bonding braids. Open and close door several times to ensure that bonding braids do not interfere with door operation.

(8) Position upper access door in the open position and mate the lower outboard access door hinges with the hinges on the outboard side of the upper access door.

(9) Install pip pins through hinges from the aft side of hinges.

(10) Connect bonding braids. Open and close both upper and lower outboard access doors several times to ensure that bonding braids to not interfere with door operation.

(10A) Connect access door stop cables.

CAUTION: TO PREVENT POSSIBLE DAMAGE TO ENGINE UPPER ACCESS DOOR AFT INBOARD LATCH, ENSURE THAT LATCH IS FULLY OPEN PRIOR TO CLOSING DOOR. SEE FIGURE 71-10-1.

(11) Engage and lock the four latches on forward and aft sides of upper access door.

(12) Engage and lock the two latches on the forward and aft sides of lower outboard access door.

(13) Fasten lower inboard and lower outboard access doors together by engaging and locking the center pin type latch and then the two hook type latches on nacelle lower centerline.

(14) Engage and lock the two latches on the forward and aft sides of lower inboard access door.

3. Removing Engine Air Intake Duct

A. To remove the engine air intake duct, proceed as follows:

(1) Remove three engine access doors. (refer to 71-10-1)

(2) Remove bolt through engine forward support ring connecting forward engine mount shock absorber and connecting link to ring.

(3) Disconnect forward engine mount bonding braid (engine side).

(4) Disconnect power lever control arm and fuel control arm Teleflex cables.

(5) On the pylon side of the engine, disconnect fuel supply and pylon de-icing lines.

(6) Disconnect nacelle de-icing thermostat connector.

(7) Disconnect fuel filter blocked warning switch.

(8) Remove fixed connector from de-icing valve support.

(9) On LH engine, disconnect two couplings and two clamps from access door de-icing line; then remove line.

(10) On RH engine, disconnect coupling on access door de-icing line.

(11) Disconnect de-icing valve inlet and outlet couplings.

(12) Disconnect fuel filter inlet and outlet lines.

(13) Remove fuel filter.

(14) Remove de-icing valve and support.

(15) Disconnect two compressor inlet temperature control (T-2) lines at fuel control.

(16) Remove nuts and washers (19) from upper and lower portions of air inlet attaching flange.

(17) Remove screws and nuts (16) from side portions of air inlet attaching flange.

NOTE: Do not remove two lower hexagonal-head screws; they are used to index air inlet to front face of engine.

4. Installing Engine Air Intake Duct

A. To install engine air intake duct, proceed as follows:

(1) Position air intake duct over attaching studs on forward face of engine.

(2) Install nuts and washers (19) on upper and lower attaching studs. Torque to 35-40 inch pounds.

(3) Install screws and nuts (16) through holes in sides of air intake duct attaching flange and forward face of engine. Torque to 35-40 inch pounds.

(4) Install de-icing valve support and de-icing valve. The electric actuator of the de-icing valve will be toward the front on the LH engine and toward the rear on RH engines.

(5) Install fuel filter.

NOTE: Attaching parts will be finally tightened only after alignment of fuel filter and de-icing valve support with rigid fuel and de-icing lines.

(6) Connect fuel filter inlet and outlet lines to fuel filter. Torque to 40 inch-pounds.

(7) Connect de-icing valve inlet and outlet couplings and safety with lockwire.

(8) On RH engine connect access door de-icing line coupling and safety with lockwire.

(9) On LH engine fasten access door de-icing line to air intake and engine with two clamps. Connect two couplings on de-icing line and safety with lockwire.

(10) Install fixed connector on de-icing valve support.

(11) Connect nacelle de-icing thermostat connector.

(12) Connect de-icing valve connector.

(13) Connect two compressor inlet temperature (T-2) lines to fuel control.

(14) Connect fuel supply line on pylon side of engine. Torque to 40 inch-pounds.

(15) Connect de-icing line on pylon side of engine.

(16) Connect power lever control arm and fuel control arm Teleflex cables.

(17) Install bolt securing forward engine mount shock absorber and connecting link to engine forward support ring. Head of bolt is forward. Tighten and safety castellated nut.

(18) Install bonding braid on engine side of forward mount.

(19) Install engine access doors (refer to 71-10-1).

5. Removing Engine Aft Nacelle Fairing

A. Prior to removal of the engine aft nacelle fairing, the thrust reversers must be actuated toward the deployed position and stopped prior to movement of the blocker doors. Remove engine aft fairing as follows:

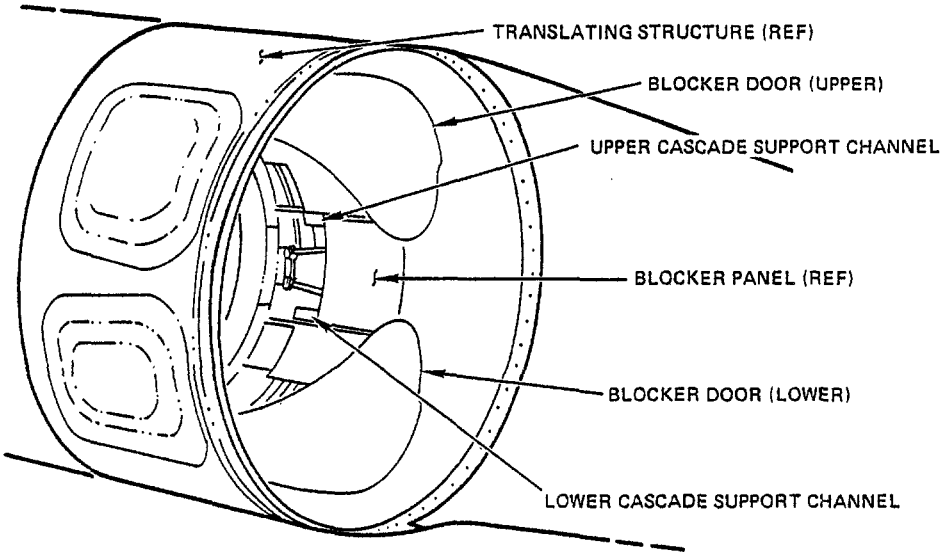
(1) Open engine access doors.

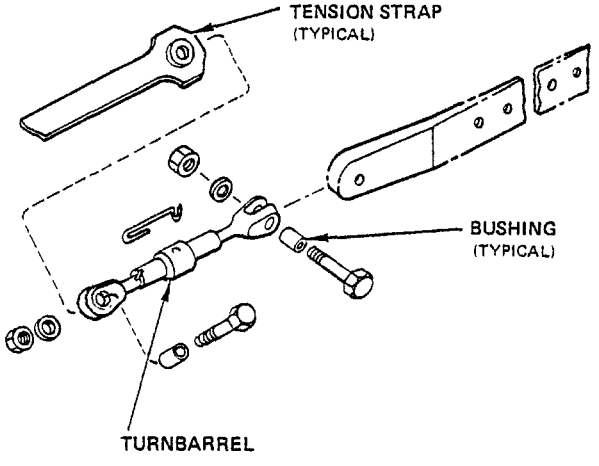
(2) Open four small doors around periphery of aft nacelle fairing.

(3) Through the four small access doors, remove tension on the four turnbuckles securing the aft fairing to the engine. Remove bolts and bushings (outboard rods) from turnbuckles.

(4) Disconnect electrical and pneumatic connections to thrust reverser components in lower portion of aft fairing.

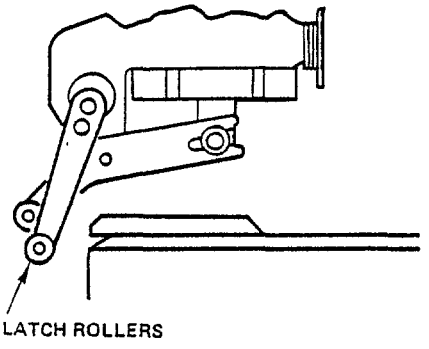
(5) Back-out (turn clockwise) four aft fairing positioning screws. These screws are left hand threads.





OUTBOARD TURNBARREL
ASSY (TYPICAL)

SHOWN IN UNLATCHED POSITION



S380-2-42-13

Installing Engine Aft Nacelle Fairing
Figure 71-10-2

(6) Remove two screws securing the forward inboard corners of the aft fairing to the pylon structure.

NOTE: Support aft fairing before removing last screw (top center screw).

(7) Remove nine screws securing the aft fairing forward frame to the aft fairing attachment ring bolted to the forward flange of the aft fan section of the engine.

(8) Support and move aft nacelle fairing straight aft until clear of the tailpipe, then place in a suitable support fixture or dolly

6. Installing Engine Aft Nacelle Fairing

NOTE: This procedure is written on the assumption that the electrical control test set P/N 234-23210 is available. If the electrical control test set is not available, the cockpit controls may be used provided the thrust reverser electrical plug remains connected to receptacle on engine mount ring. On NA380-5, -10, -12 and later airplanes, if the cockpit controls are to be used and the airplane is on jacks, the "EMER STOW" circuit breaker must be pulled to prevent activation of the auto stow system.

A. To install engine aft nacelle fairing, see figure 71-10-2 and proceed as follows:

(1) Before installing aft nacelle fairing on engine, the following tasks must be accomplished.

(a) Make certain the four aft nacelle fairing positioning jacks are fully retracted (turn clockwise to retract).

(b) Secure the two inboard and two outboard tension straps over their respective teflon rub pads on aft engine mount ring.

(c) Check that thrust reverser is partially deployed (12-14 inches).

(d) Remove all access doors on aft nacelle fairing.

(e) Check that latch rollers are retained in the unlatched position. (See figure 71-10-2).

(f) Place thrust reverser electrical lead out of adjacent access opening.

(g) Check that turnbuckle bushings in outboard tension straps may be installed without using force.

(2) With two or more persons supporting the aft nacelle fairing, move fairing forward over engine tailpipe until forward frame of aft fairing contacts aft fairing attachment ring secured to engine.

(3) Rotate aft nacelle fairing as necessary to align attachment holes in aft nacelle fairing forward frame with attachment holes in attachment ring on engine.

(4) Install attachment screws through attachment ring on engine into nut plates on aft side of forward frame of aft fairing.

(5) Install screws, washers, and nuts attaching forward inboard corners of aft fairing to pylon.

(6) Connect test panel (P/N M304522) and a regulated source of pneumatic power to thrust reverser pneumatic actuator (refer to Chapter 78).

(7) Connect electrical control test set (P/N 234-23210) to thrust reverser electrical harness (refer to Chapter 78). Make certain toggle switch on control test set is at neutral.

(8) Apply a pressure of 7 ± 1 psig to the pneumatic actuator and place the control test set toggle switch at deploy. After thrust reverser is deployed, reduce pneumatic pressure to 4 ± 1 psig.

CAUTION: STAY CLEAR OF BLOCKER DOORS AS THEY CLOSE.

(9) Momentarily hold the toggle switch on the control test set at STOW and move the thrust reverser towards the stow position about four inches. At this point the forward edges of the upper and lower cascade suuport channels, both inboard and outboard, should be adjacent to the aft edge of the tailpipe. (See figure 71-10-2.)

CAUTION: STOWING THE THRUST REVERSER TRANSLATING STRUCTURE BEFORE ADJUSTMENTS ARE COMPLETED COULD RESULT IN DAMAGE TO THE TAILPIPE SEAL.

(10) Adjust the aft nacelle fairing positioning jacks to obtain a dimension of 0.150 (minimum) between the leading edge of the cascade support channels and the tailpipe.

NOTE: Counterclockwise movement of the four aft nacelle fairing positioning jacks (when jacks are in contact with engine ring) will cause the aft nacelle fairing to move in the following directions:

(a) Upper jack . . . . . Up

(b) Outboard jack. . . . . Outboard

(c) Lower inboard jack . . . . . Inboard and down

(c) Upper inboard jack . . . . . Inboard and up

- Adjustment of jacks (a) and (b) shall not exceed 3 full turns.

(11) Continue translating the thrust reverser toward the stowed position until the leading edges of the blocker doors approach but do not contact the tailpipe. Adjust the aft fairing positioning jacks to obtain 0.110 inch (min) clearance between the blocker doors and the top and bottom of the tailpipe assembly.

(12) Slowly translate the thrust reverser toward the stowed position until the blocker doors and the tailpipe are opposite each other (parallel). Stop at this point and check that the 0.110 dimension is maintained. Continue translating thrust reverser and recheck the 0.110 dimension at 3 inches before full stow.

NOTE: There are no dimensional checks of the thrust reverser relative to the tailpipe after the unit has been fully stowed.

(13) Check all aft nacelle fairing positioning jacks and make certain that they are at least touching the tailpipe (finger tight) then tighten each jack an additional $\frac{1}{4}$ turn counterclockwise.

(14) Deploy thrust reverser and repeat steps (9) through (12) to verify measurements.

(15) Stow thrust reverser and connect outboard straps to clevis. Tighten turnbuckles until there is no slack remaining in strap assembly, but a slight movement of the turnbuckle bolts is noticeable when light hand pressure is applied.

CAUTION: BE CERTAIN WHEN INBOARD AND OUTBOARD STRAPS ARE INSTALLED, THEY ARE IN TENSION AND NOT IN COMPRESSION DUE TO BUCKLING OF STRAPS AT ATTACH POINTS.

(16) Connect inboard straps to turnbuckles. Tighten turnbuckles until there is no slack remaining in strap assembly, but a slight movement of the turnbuckle bolt is noticeable when light hand pressure is applied.

(17) Apply a pressure of 7 ± 1 psig to the pneumatic actuator and place control test set toggle switch to deploy. Operate thrust reverser unit towards deploy just enough to open the pneumatic latches, then reduce pressure to 4 ± 1 psig (or less if practicable) and slowly cycle thrust reverser and recheck all dimensions obtained in steps 9, 10 and 11.

(18) Cycle thrust reverser unit 5 complete cycles. Insure that proper clearances have been maintained. If clearances are not held, disconnect turnbuckles and repeat steps (9) through (17).

(19) Install locks on turnbuckles.

(20) Shut off air supply and disconnect air supply line to pneumatic actuator.

(21) Reconnect bleed air line from engine to pneumatic actuator.

(22) Turn off electrical power to control test set if used, and disconnect test set from plug.

(23) Reconnect electrical plug to receptacle on engine ring. Safety plug with lockwire.

(24) Install all access doors on aft nacelle fairing.

(25) Install engine access doors. (Refer to 71-10-1.)

(26) Perform an operational checkout of thrust reverser system (engines operating).

7. Removing Engine Inner Fairing

A. To remove inner fairing from engine, proceed as follows:

(1) Open engine access doors. (Refer to Figure 71-10-1.)

(2) Remove two screws securing hydraulic line firewall to inner fairing.

(3) Cut lockwire and remove hexagonal-headed bolts (11) securing lateral fairing to upper and lower inboard segments of inner fairing. Remove lateral fairing.

(4) Cut lockwire and remove hexagonal-headed bolts (11) securing diffuser support fairing to lower inboard and outboard segments of inner fairing. Remove diffuser support fairing.

(5) Remove five screws securing upper and lower outboard segments of inner fairing together.

(6) Cut lockwire and loosen hexagonal-head bolts (6) securing aft end of lower outboard segment of inner fairing to engine.

(7) Remove bolts and washers (6) securing forward end of lower outboard segment of inner fairing to engine fire wall. Slide segment forward and remove.

(8) Cut lockwire and loosen hexagonal-head bolts (6) securing aft end of lower inboard segment of inner fairing to engine.

(9) Remove bolts and washers (6) securing forward end of lower inboard segment of inner fairing to engine firewall. Slide segment forward and remove.

(10) Cut lockwire and loosen hexagonal-head bolts (12) securing aft end of upper segment of inner fairing to engine.

(11) Remove bolts and washers (10) securing forward end of upper segment of inner fairing to engine firewall. Slide segment forward and remove.

8. Installing Engine Inner Fairing

A. To install inner fairing on engine, proceed as follows:

(1) Slide aft end of upper segment of inner fairing under previously loosened bolts (12) and locking plates. Tighten bolts and safety with lockwire.

(2) Install asbestos seal between inner fairing and firewall. Install bolts and washers (10) securing forward end of upper segment of inner fairing to engine firewall.

(3) Slide aft end of lower outboard segment of inner fairing under previously loosened bolts (6) and locking plates. Tighten bolts and safety with lockwire.

(4) Install asbestos seal between inner fairing and firewall. Install bolts and washers (6) securing forward end of lower outboard segment of inner fairing to engine firewall.

(5) Slide aft end of lower inboard segment of inner fairing under previously loosened bolts and locking plates. Tighten bolts and safety with lockwire.

(6) Install asbestos seal between inner fairing and firewall. Install bolts and washers (6) securing forward end of lower inboard segment of inner fairing to engine firewall.

(7) Install diffuser support fairing to lower inboard and outboard segments of inner fairing with hexagonal-head bolts (11). Safety bolts with lockwire.

(8) Install lateral fairing on upper and lower inboard segments of inner fairing with hexagonal-head bolts (11). Safety bolts with lockwire.

(9) Install hydraulic line firewall to inner fairing with screws (2).

(10) Install engine access doors. (refer to 71-10-1).

9. Engine and Pylon Rubber Seal Repairs

A. The rubber seals, plugs and strips may require repair as they become damaged or worn. To repair these items, proceed as follows:

(1) Clean the area to be repaired with MEK.

(2) Bond repair patch or new seal using MMM-A-1617, type III (or equivalent) general purpose adhesive. Silicon foam rubber seals and strips use MIL-A-4606 (or equivalent) or Dow Corning 732 RTV elastic adhesive.

10. Servicing Engine Damper

A. To service engine damper, proceed as follows:

NOTE: When removing the damper, use a bolt of the same length to push out the installed bolt. This method will allow the spacer washers to remain in their original position and prevent an alignment procedure.

(1) Remove bolts securing damper to engine and pylon mounts. Note direction of bolt heads.

(2) Hold bungee rod end down and remove plug from end of bungee.

(3) Completely fill bungee with MIL-H-5606( ) hydraulic fluid.

(4) Install and tighten plug in bungee.

(5) Reinstall bungee on engine and pylon mounts. Torque engine mount bolt (bungee mount bolt) to 8-11 foot-pounds. Install cotter key.

(6) Tighten bungee to pylon mount bolt. Install cotter key.

CAUTION: WHEN TORQUEING THE BUNGEE TO PYLON MOUNT BOLT, TIGHTEN NUT SO THAT BOLT DOES NOT ROTATE. TO HIGH TORQUE ON THIS BOLT MAY CAUSE DAMAGE TO PYLON MOUNT.

ENGINE MOUNT SYSTEM - DESCRIPTION

1. General

Each engine is supported within the nacelle at four attach points; one at the front face of the engine, two on the forward frame of the aft fan housing and one at the aft frame of the aft fan housing. The forward engine mount is non-adjustable and consists of the forward support fitting attached to the pylon, and the engine forward support ring bolted to the forward face of the engine. The forward support fitting is attached to the forward support ring by a connecting link and a shock absorber. The engine center mount is the primary engine support and consists of a vertical "cradle" type fitting bolted to an engine support forging attached to the pylon and fuselage structure. The vertical cradle fitting attaches to the front face of the aft fan housing at two points. The upper point is a fixed point while the lower point has an eccentric adjustment bushing which controls the vertical alignment of the engine. The aft attach fitting consists of an adjustable rod attached to the pylon structure at one end and to the engine aft support ring at the other. The aft support ring is attached to the aft frame of the aft fan housing. Changing the length of the adjustable rod controls the lateral alignment of the engine.

ENGINE MOUNT SYSTEM - MAINTENANCE PRACTICES

1. Removal of Engine Forward Support Fitting

A. To remove engine forward support fitting, proceed as follows:

(1) Remove engine from pylon (refer to 71-00-3).

(2) Remove bolt securing shock absorber to forward fitting.

(3) Remove bolt securing connecting link to forward fitting.

(4) Remove bolts (4) securing forward fitting to pylon. Remove fitting and shim (if installed) from pylon. Retain shim for reinstallation.

2. Installation of Engine Forward Support Fitting

A. The installation procedure for engine forward support fitting is essentially the reverse of the removal procedure. Torque bolts securing forward support fitting to pylon structure to 175-270 inch-pounds.

3. Removal of Engine Center Support Fitting

A. To remove engine center support fitting, proceed as follows:

(1) Remove engine from pylon (refer to 71-00-3).

(2) Open access doors on pylon upper and lower skins adjacent to center support fitting.

(3) Remove bolts (4) securing vertical center support fitting to structural support forging in pylon.

(4) Pull vertical cradle fitting straight out from support forging until clear of guide pin.

4. Installation of Engine Center Support Fitting

A. The installation procedure for engine center support fitting is essentially the reverse of the removal procedure. Torque bolts securing vertical cradle fitting to pylon structural support forging to 900-1220 inch-pounds.

5. Removal of Engine Aft Support Fitting

A. To remove engine aft support fitting, proceed as follows:

(1) Remove engine from pylon (refer to 71-00-3).

(2) Open two access doors on pylon lower skin adjacent to aft support fitting.

(3) Through access doors in pylon lower skin, remove cotter pin, nut and bolt from inboard end of aft support fitting.

(4) Remove small bolts (10) securing cup shaped bellows to circular fitting on outboard rib of pylon.

(5) Remove aft fitting from pylon.

6. Installing Engine Aft Support Fitting

A. The installation procedure for the engine aft support fitting is essentially the reverse of the removal procedure. Tighten nut securing inboard end of aft support fitting to pylon structure. Safety nut.

CHAPTER

72

ENGINES

CHAPTER 72

ENGINE

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ENGINE FUEL AND CONTROL

CHAPTER 73

ENGINE FUEL AND CONTROL

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ENGINE FUEL AND CONTROL - DESCRIPTION AND OPERATION

1. General

For Description and Operation and Maintenance Practices of the Fuel System and Controls for the JT-12A engine (airplanes NA-370-2 through -6, -8 and -9), refer to Pratt & Whitney Engine Maintenance Manual P&WA 435107.

For Description and Operation of the Fuel System and Controls of the CF-700-2D-2 engine (airplanes NA-370-1 and -7, and airplanes NA-380-1 and later), refer to General Electric CF-700 Maintenance Manual SEI-187.

Description and Operation of the Main Fuel Filter Differential Pressure switch and the Engine Fuel Recovery System are in the following paragraphs.

2. Main Fuel Filter Differential Pressure Switch - Airplanes NA-370-2 Through -6, -8, and -9

The main fuel supply filter incorporates a differential pressure switch which illuminates the respective engine fuel filter block caution light in the cockpit (see figure 73-00-2). The differential pressure switches sense fuel pressure drop across the fuel filters. In case either the left or right engine fuel filter becomes contaminated with ice or foreign matter and the pressure drops across the filter increases to 7 to 8 psi, the differential pressure switch closes and the respective caution light will come on. When either engine filter block caution light comes on during flight, the FUEL HEATER switch is actuated to ON to heat fuel before entering the filter to melt the ice. When the fuel filter block is eliminated, the engine filter block caution light will go out.

3. Main Fuel Filter Differential Pressure Switch - Airplanes NA-370-1 and -7, and Airplanes NA-380-1 and Later

The main fuel supply filter incorporates a differential pressure switch which illuminates the respective engine fuel filter block light in the cockpit (see figure 73-00-3). The pressure switches sense fuel pressure drop across the fuel filters. In case either the left or right engine fuel filter becomes contaminated with ice or foreign matter and the pressure drop across the filter increases to approximately 1.5 psi, the differential-pressure switch closes and the respective caution light will come on. In case the fuel pressure drop increases to approximately 1.7 psi, the filter bypass valve automatically opens, allowing full fuel flow to the engine. However, the caution light will remain on until the differential pressure switch is mechanically reset. The filter element must be removed in order to reset the differential pressure switch.

4. Engine Fuel Recovery System - Airplanes NA-370-1 and -7, and Airplanes NA-380-1 Through -33, -35, -37 and -38 Changed by Service Bulletin 75-2, and Airplanes NA-380-34, -36, -39 and Later

The engine fuel recovery system major components are the fuel collector tank, eductor, check valve, fuel recovery solenoid valve and associated fuel lines and wiring. (See figure 73-00-4). The purpose of the fuel recovery system is to use fuel that has been collected from the engine manifold drain after engine shut-down.

The operational sequence of the engine fuel recovery system is similar for both engines. Only the left engine fuel recovery system is described (see figure 73-00-4).

The fuel recovery system operates during take-off to 14,300 feet altitude or during engine ground run-up when engine thrust is 80% or above.

To operate the fuel recovery system, the engines must be running at 80% thrust or above, so that 28 vdc essential bus voltage can be applied to the fuel recovery solenoid valve (valve open). When the fuel recovery solenoid valve opens, engine fuel pump pressure is applied to the eductor which draws out residual fuel from the collector tank and returns it to the fuselage fuel tank for re-use. When the airplane attains 14,300 feet altitude, an altitude pressure switch will close and close the fuel recovery solenoid valve, which shuts off the applied fuel pump pressure to the eductor and de-energize the fuel recovery system. When the airplane descends to 14,300 feet altitude, the altitude pressure switch will open and energize the fuel recovery solenoid valve to open, and put the recovery system back into operation until landing.

The function of the eductor is when engine fuel pump pressure going through the nozzle and into the diffuser inlet, causes a low pressure around the nozzle, and draws out fuel from the collector tank and returns it to the fuselage fuel tank (see figure 73-00-4). The check valve between eductor and tank is to prevent fuel pump interstage pressure to enter into collector tank.

ENGINE FUEL AND CONTROL - MAINTENANCE PRACTICES

1. Removing and Inspecting Engine Main Fuel Supply Filter - Aircraft NA-370-1 and -7, NA-380-1 and Later

A. To remove and inspect the main fuel supply filter for the engine, proceed as follows:

NOTE: The engine main fuel filter is located on the forward pad of the engine accessory gearbox. To gain access to the fuel filter it will be necessary to open the engine lower, inboard and outboard access doors on the right engine. On the left engine, only the lower outboard access door need be opened.

(1) Remove lockwire from plug in center of filter cover (ref. Fig. 73-00-1).

(2) Remove plug and "o" ring from filter cover and drain residual fuel from filter into a container capable of holding at least one pint.

(3) Remove the four nuts and washers securing the filter cover to the filter body. Loosen all four nuts evenly until the compression on the filter cover is removed. This will prevent damage to the filter body.

(4) Remove the filter cover, cover "o" ring, spring, element insert, and plate from the filter body. Remove the outer filter element, inner element assembly, and "o" ring. Discard all "o" rings.

(5) Using a high intensity light source and a magnifier of at least 10 power magnification, inspect the inner and outer filter elements for metal chips, synthetic rubber, lint and other foreign materials.

(6) If the inspection of the filter elements reveal no defects, reinstall elements as follows:

(a) Install a new "o" ring on the element insert. Insert the element insert into the outer filter element and inner filter assembly. Install a new "o" ring on the inner filter assembly.

(b) Place spring and plate on the assembled filter elements and install in the filter body. With a new "o" ring installed in the filter cover, position the filter cover over the filter elements and secure in place using four nuts and washers. Hold cover in place while installing the nuts. The nuts must be tightened evenly to prevent damage to the filter body. Torque nuts to 60-85 inch-pounds.

(c) Using a new "o" ring on drain plug in filter cover, install drain plug and torque to 40-60 inch-pounds and safety with lockwire.

(7) If the filter elements are found to be defective, replace the filter.

(8) If contamination is evident, clean the filter elements as follows:

(a) Wash in Stoddard Solvent (Federal Specification P-D-680) or equivalent. Use a soft bristle brush to remove foreign material. Vapor degrease with trichloroethylene.

(b) Ultrasonic clean the filter elements for 10 to 20 minutes using trichloroethylene as a solvent. Suspend the filter elements vertically to permit the loosened contaminants to fall from the filter into the solvent.

(c) If ultrasonic cleaning equipment is not available, clean the filter elements by immersing in trichloroethylene and continually agitating the filter element. Lightly scrub the inner and outer surfaces of the stainless steel mesh filter elements with a stiff bristle brush.

WARNING: THE CLEANING OPERATION SHALL BE PERFORMED IN A WELL VENTILATED AREA. PRECAUTIONS SHALL BE TAKEN TO PREVENT INHALATION OF VAPORS EMITTED BY VOLATILE CLEANING SOLUTIONS AND TO MINIMIZE DANGER OF EXPLOSION AND/OR FIRE HAZARDS.

NOTE: A dental-type brush with the bristles cut to match the saw tooth flutes may be used.

- Only clean filtered solvents should be used. Make sure that containers used in the cleaning procedures are clean and free of contaminants.

(9) Blow off any visible moisture and/or contaminants, especially lint, with filtered, dry, compressed air or nitrogen. Visually inspect the filter elements as indicated in Step 5. If the elements are not completely clean, repeat the cleaning procedure.

(10) When the elements are completely clean, reinstall the filter elements as indicated in Step 6.

2. Removing and Inspecting Fuel Pump Filter - Aircraft NA-370-1 and -7, NA-380-1 and Later

A. To inspect the fuel pump filter, proceed as follows:

NOTE: The fuel pump filter is located on the lower side of the fuel pump. Access to the fuel pump filter will require that on the left engine the lower inboard and outboard access doors be opened. On the right engine only the lower outboard access need be opened.

(1) Remove lockwire and unscrew filter from the fuel pump (ref. Fig. 73-00-1).

(2) Remove "o" ring and filter element from the plug and spring assembly.

(3) Using a high intensity light and a magnifier of at least 10 power magnification, visually inspect the filter element for the presence of foreign material.

(4) If the inspection of the filter reveals a complete absence of foreign material, reinstall filter as follows:

(a) Install a new "o" ring on the spring and plug assembly. Apply petrolatum to the filter (to hold it in position until the spring is installed), and insert the filter, closed end out, into the fuel pump.

CAUTION: MAKE SURE THAT FILTER IS PROPERLY SEATED. THE END OF THE FILTER SHOULD BE ABOUT ONE-HALF INCH INSIDE OF THE CASTING WITH A MINIMUM OF FIVE THREADS SHOWING.

(b) Install the plug and spring assembly, maintaining the proper filter element position. Torque filter plug to 60-80 inch pounds and safety with lockwire.

CAUTION: IF IT IS NOT POSSIBLE TO TURN THE PLUG TWO COMPLETE TURNS WITH THE FINGERS, THE FILTER HAS DROPPED OUT OF POSITION AND FURTHER TIGHTENING WILL CAUSE IT TO COLLAPSE. REMOVE THE PLUG AND RESEAT THE FILTER.

(5) If the filter is found to be defective, replace the filter.

(6) If contamination is evident on the filter element clean the filter as follows:

(a) Immerse the filter element in trichloroethylene, Stoddard Solvent, kerosene or equivalent. Continually agitate the filter in the solvent while brushing the filter with a soft bristle brush to dislodge the contaminants.

WARNING: THE CLEANING OPERATION SHALL BE PERFORMED IN A WELL VENTILATED AREA. PRECAUTION SHALL BE TAKEN TO PREVENT INHALATION OF VAPORS EMITTED BY VOLATILE CLEANING SOLUTIONS AND TO MINIMIZE DANGER OF EXPLOSION AND/OR FIRE HAZARDS.

NOTE: Only clean, filtered solvents should be used. Make certain that all containers used in the cleaning procedures are clean and free of contaminants.

(b) Blow off any visible moisture and/or contaminants, especially lint, with filtered, dry compressed air or nitrogen. Visually inspect the filter element as indicated in Step 3 of this procedure. If filter element is not completely clean, repeat the cleaning procedure.

(c) When filter is completely clean, reinstall filter as indicated in Step 4 of this procedure.

2A. Adjustment of Fuel Filter Differential Pressure Switch - NA-370-1, -7, NA-380-1 and Later

A. Adjustment of the fuel filter differential pressure switch is required whenever the fuel filter blocked light illuminates when contamination or ice of the fuel filter elements are not a factor. To adjust switch, proceed as follows:

(1) Remove electrical power from airplane. Open engine cowling to gain access to fuel filter.

(2) Disconnect electrical plug from top of filter (see figure 73-00-5).

(3) Remove drain plug from filter cover and drain all fuel from filter assembly.

(4) Disconnect fuel lines attached to filter housing; then remove filter from airplane. Cap all open fuel lines.

(5) Remove filter cover and elements.

(6) Remove lockwire from both screws securing switch cover to filter housing. Place cover to one side to gain access to set screw.

(7) Reset switch by turning knurled knob clockwise.

(8) Insert .006 inch feeler gage in poppet assembly (see figure 73-00-5). Check that cam assembly is against the poppet assembly.

(9) Adjust set screw in magnet arm as required until switch closes (continuity between pins A and B of electrical plug).

(10) Bend leaf spring as required to obtain .065(±.005) inch between leaf spring and magnet. (See figure 73-00-5.)

(11) Remove .006 inch feeler gage.

(12) Manually operate cam assembly and check for proper switch operation and proper clearance between leaf spring and magnet assembly.

(13) Install switch cover on housing using new "O' ring (MS9021-035). Lockwire screws.

(14) Ensure that switch is reset; then install filter elements. Refer to Removing and Inspecting Engine Fuel Control Filter - Aircraft NA-370-1 and -7, NA-38--1 and Later.

(15) Remove caps from fuel lines; then install filter on engine using new "O" rings. Tighten bolts and lockwire. Connect, tighten and lockwire electrical connector.

(16) Apply fuel pressure to filter and check for fuel leaks.

(17) Close engine cowling.

2B. Replacement of Fuel Filter Differential Pressure Switch - NA-370-1, -7, NA-380-1 and Later

A. To remove fuel filter differential pressure switch, proceed as follows:

(1) Remove electrical power from airplane. Open engine cowling to gain access to fuel filter.

(2) Disconnect electrical plug from top of filter (see figure 73-00-5).

(3) Remove drain plug from filter cover and drain all fuel from filter assembly.

(4) Disconnect fuel lines attached to filter housing; then remove filter from airplane. Cap all open fuel lines.

(5) Remove filter cover and elements.

(6) Remove the four screws securing electrical connector to switch cover; then unsolder switch wires from connector. Identify wires for re-installation procedures.

(7) Remove lockwire from both screws securing switch cover to filter housing. Remove switch cover.

(8) Remove lockwire from the two screws securing switch to filter. Remove switch mounting screws and switch.

NOTE: Prior to removing mounting screws, secure switch actuating lever spring to prevent it from being lost.

(9) Unsolder electrical wires from switch. Identify wires for re-installation.

B. To install fuel filter differential pressure switch, proceed as follows:

(1) Obtain serviceable switch (MS25085-1) and solder wires (as identified in step A, -(9)) to switch. Install sleeving (two lengths on each wire) over soldered joints.

(2) Assemble switch, switch actuating lever and spring; then secure components to filter housing with the two mounting screws. Torque screws 4 to 5 in. lbs. and lockwire.

(3) Thread electrical wires through switch cover electrical connector opening. Position electrical connector so that keyway is up, then solder electrical wires (as identified in step A (6)) to connector pins. Place sleeving over soldered joints.

(4) Install electrical connector on housing. Install, tighten and lockwire connector mounting screws.

(5) Perform steps (7) through (17) of "Adjustment of Fuel Filter Differential Pressure Switch - NA-370-1, -7, NA-380-1 and Later" in this section.

3. Removing and Inspecting Engine Fuel Control Filter - Aircraft NA-370-1 and -7, NA-380-1 and Later

A. This inspection should be performed immediately after performing the engine fuel pump filter inspection if contamination is found on the fuel pump filter element. The fuel control filter is located on the front of the engine fuel control and is accessible through the lower outboard engine access door on the right engine. On the left engine, both the lower inboard and outboard access doors must be opened to perform this inspection. To inspect the engine fuel control filter, proceed as follows:

(1) Remove lockwire from plug securing the filter element in fuel control housing (ref. Fig. 73-00-1).

(2) Unscrew the plug and remove the "o" ring, spring, and filter element.

NOTE: Do not remove the second filter located immediately behind the retaining ring.

(3) Using a high intensity light source and a magnifier of a least 10 power magnification, visually inspect the filter element for defects and the presence of foreign material.

(4) If the inspection of the filter element reveals a complete absence of foreign material, reinstall filter element as follows:

(a) Install a new "o" ring on the plug. Install filter element, spring and plug in fuel control housing. Torque plug to 60-80 inch-pounds and safety with lockwire.

(5) If filter element is found to be defective, replace the filter element.

(6) If contamination is evident on the filter element, clean the element as follows:

(a) Immerse the filter element in trichloroethylene, Stoddard Solvent, kerosene, or equivalent. While agitating the filter element in the cleaning solution, brush the surfaces of the filter element with a soft bristle brush to remove the contaminants.

WARNING: CLEANING OPERATIONS SHALL BE PERFORMED IN A WELL VENTILATED AREA. PRECAUTIONS SHALL BE TAKEN TO PREVENT INHALATION OF VAPORS EMITTED BY VOLATILE CLEANING MATERIALS, AND TO MINIMIZE DANGER OF EXPLOSION AND/OR FIRE HAZARDS.

NOTE: Only clean filtered solvents should be used. Make certain that containers used during the cleaning procedures are clean and free of contaminants.

(b) Blow off any visible moisture and/or contaminants especially lint, with dry, filtered compressed air or nitrogen. Visually

inspect the filter as indicated in Step 3 of this procedure. If filter element is not completely clean, repeat cleaning procedure.

(c) When filter element is completely clean, reinstall using procedures indicated in Step 4 of this procedure.

4. Removing and Inspecting Overspeed Governor Servo Filter - Aircraft NA-370-1 and -7, NA-380-1 and Later

A. This inspection should be performed immediately after performing the engine fuel pump filter inspection if contamination is found on the fuel pump filter element. The overspeed governor servo filter is mounted on the overspeed governor and is accessible by opening the lower inboard and outboard engine access doors. To inspect the overspeed governor servo filter, proceed as follows:

(1) Disconnect and cap the fuel lines connecting to the filter housing (see Fig. 73-00-1).

(2) Loosen clamps securing filter assembly to the overspeed governor. Slide filter forward from clamps.

(3) Remove lockwire, plug, "o" rings and filter element from filter housing.

(4) Using a high intensity light source and a magnifier of at least 10 power magnification, visually inspect the filter element for defects and the presence of foreign materials.

(5) If inspection of the filter element reveals the absence of foreign materials, reinstall filter as follows:

(a) Install new "o" rings on filter element and filter plug.

(b) Insert filter element into filter housing. Install plug and torque to 90-110 inch-pounds and safety with lockwire.

(c) Install filter housing in clamps and tighten clamps. The largest O.D. of the filter housing should be forward of the clamp.

(d) Connect fuel lines to fittings at front and rear of filter housing.

(6) If filter element is found to be defective, replace filter element.

(7) If contamination is evident on the filter element, clean the element as follows:

(a) Immerse the filter element in trichloroethylene, Stoddard Solvent, kerosene or equivalent. While agitating the filter element in the cleaning solution, brush the surfaces of the element with a soft bristle brush to remove the contaminants from the filter element convolutions.

WARNING: CLEANING OPERATIONS SHALL BE PERFORMED IN A WELL VENTILATED AREA. PRECAUTIONS SHALL BE TAKEN TO PREVENT INHALATION OF VAPORS EMITTED BY VOLATILE CLEANING MATERIALS, AND TO MINIMIZE DANGER OF EXPLOSION AND/OR FIRE HAZARDS.

NOTE: Only clean filtered solvents should be used. Make certain that containers used during the cleaning procedures are clean and free of contaminants.

(b) Blow off any visible moisture and/or contaminants especially lint, with dry, filtered compressed air or nitrogen. Visually inspect the filter element as indicated in Step 4 of this procedure. If filter element is not completely clean, repeat the cleaning procedure.

CAUTION: DO NOT ATTEMPT TO PROBE THE FILTER CONVOLUTIONS WITH A SHARP TOOL AS THIS MAY DAMAGE THE FILTER ELEMENT.

NOTE: Some dark colored stains may remain in the bottom of the filter convolutions. These stains are not harmful and will not affect the operation of the filter.

(c) When the filter is completely clean, install the filter element in accordance with Step 5 of this procedure.

5. Resetting Fuel Filter Differential Pressure Switch - Aircraft NA-370-1, -7, NA-380-1 and Later

A. To reset fuel filter differential pressure switch, proceed as follows:

(1) Open engine access doors.

(2) Remove plug from fuel filter cover and drain residual fuel from filter housing.

(3) Remove nuts securing filter cap to housing; then remove filter cap.

(4) Remove elements from filter body (see figure 73-00-1).

(5) Turn reset indicator clockwise to reset pressure switch.

(6) Install elements in filter body; then install filter cap (use new "o" rings). Torque filter cap securing nuts to 60-85 inch-pounds.

(7) Install plug in fuel filter cap and torque to 40-60 inch-pounds. Safety plug with lockwire.

(8) Pressure check fuel filter assembly.

(9) Close engine access doors.

6. Removing Fuel Collector Tank and Cleaning Tank Screen - Airplanes NA-370-1 and -7 and Airplanes NA-380-1 Through -33, -35, -37 and -38 changed by Service Bulletin 75-2 and Airplanes NA-380-34, -36, -39 and Later

A. Remove and clean fuel tank screen as follows:

(1) Open both lower nacelle doors.

(2) Loosen strut duct clamp at the starter generator cooling air outlet.

(3) Remove six screws and washers from strut assembly and remove strut assembly from engine.

(4) Disconnect two lines (inlet and outlet) going to eductor and dust cap lines.

(5) Disconnect three lines going to fuel collector tank and dust cap lines.

(6) Remove five collector tank mounting bolts, washers and nuts and remove tank from engine.

(7) Remove the eductor and check valve together from collector tank fitting.

(8) Cap all fittings on collector tank except screen fitting. (See figure 73-00-4).

WARNING: THE CLEANING OPERATION SHALL BE PERFORMED IN A WELL VENTILATED AREA. PRECAUTION SHALL BE TAKEN TO PREVENT INHALATION OF VAPORS EMITTED BY VOLATILE CLEANING SOLUTIONS AND TO MINIMIZE DANGER OF EXPLOSION AND/OR FIRE HAZARDS.

NOTE: Only clean, filtered solvents should be used. Make certain that all containers used in the cleaning procedures are clean and free of contaminants.

(9) Half fill the collector tank with trichloro-ethylene, Stoddard Solvent, kerosene or equivalent through the screen fitting and cap fitting.

(10) Agitate collector tank and remove cap from engine manifold drain fitting (see figure 73-00-4) and drain tank into a clean container (loosen vent line fitting cap for easier draining). Inspect cleaning fluid in container for any contaminants. If contaminants appear, repeat cleaning procedure.

(11) When collector tank is completely clean remove all caps from tank fittings and purge tank with filtered dry compressed air or nitrogen until inside of tank is completely dry.

(12) Install check valve and eductor on collector tank fitting (eductor small fitting forward).

(13) Mount collector tank on engine with five bolts, washers and nuts and tighten.

(14) Connect all lines (3) to collector tank and tighten.

(15) Connect two lines (inlet and outlet) to the eductor and tighten. Be sure check valve is safetied to eductor.

(16) Install strut assembly with six screws and washers and tighten dust clamp at starter generator.

(17) Close both lower nacelle doors.

(18) Perform engine run-up and check for fuel leaks.

7. Removing and Installing Fuel Recovery Solenoid Valve - Airplanes NA-370-1 and -7, and Airplanes NA-380-1 Through -33, -35, -37 and -38 Changed by Service Bulletin 75-2, and Airplanes NA-380-34, -36 and -39 and Later

A. Remove and install fuel recovery solenoid valve as follows:

(1) Open both lower nacelle doors.

(2) Disconnect both fuel lines from fuel recovery solenoid valve and dust cap lines.

(3) Disconnect electrical connector from fuel recovery solenoid valve.

(4) Remove two mounting bolts and washers from solenoid valve and remove valve from engine.

(5) Place new valve on mount and install two mounting bolts and washers and tighten.

(6) Connect both fuel lines to solenoid valve and tighten.

(7) Connect electrical connector to solenoid valve and lockwire.

(8) Close both lower nacelle doors.

(9) Perform engine run-up and check for fuel leaks.

8. Removing and Installing Check Valve - Airplanes NA-370-1 and-7, and Airplanes NA-380-1 Through -33, -35, -37 and -38 Changed by Service Bulletin 75-2, and Airplanes NA-380-34, -36, and -39 and Later

A. Remove and install check valve as follows: (See figure 73-00-4).

(1) Perform steps (1) through (4) and (7) in "Removing Fuel Collector Tank and Cleaning Tank Screen".

(2) Remove check valve from eductor and remove "O" ring (see figure 73-00-4).

(3) Install new check valve and new "O" ring (P/N MS 28778-4) on eductor and lockwire.

(4) Perform steps (12) and (15) through (18) in "Removing Fuel Collector Tank and Cleaning Tank Screen".

9. Removing Engine Fuel Pump Screen - Airplanes NA-370-2 Through -6, and -8 and -9

A. To remove engine fuel pump screen, proceed as follows:

(1) Drain fuel from filter housing using drain valve.

CAUTION: WHEN REMOVING BOLTS SECURING COVER, DO NOT REMOVE ONE BOLT AT A TIME; BOLTS MUST BE LOOSENED AND REMOVED EVENLY TO PREVENT DAMAGE TO HOUSING.

(2) Remove lockwire and remove three bolts securing filter cover to fuel pump.

(3) Rotate cover one sixth of turn in clockwise direction and remove filter element.

(4) Inspect filter element for contamination which might indicate system failure; then discard filter element.

10. Installing Engine Fuel Pump Screen - Airplanes NA-370-2 Through -6, and -8 and -9

A. To install engine fuel pump screen, proceed as follows:

(1) Install new filter element (CE49881) and packing (MS9021-027 and MS9021-110) in housing.

(2) Position new packing (MS9021-154) on cover and place cover on filter housing.

CAUTION: HOLD COVER IN POSITION WHILE INSTALLING BOLTS. BOLTS MUST BE TIGHTENED EVENLY AROUND TO AVOID DAMAGE TO HOUSING.

(3) Install cover and rotate 1/6 turn counter-clockwise and install three bolts; then tighten bolts evenly to a torque value of 50 to 70 inch-pounds.

(4) Safety all three bolts with lockwire.

(5) Check fuel pump for fuel leaks after engine run.

11. Removing Fuel Control Unit Screen - Airplanes NA-370-2 Through -6, and -8 and -9

A. To remove fuel control unit screen, proceed as follows:

(1) Remove lockwire and screw holding lockpin retaining bracket; then remove pin retaining bracket.

(2) Remove screen assembly lockpin.

NOTE: It may be necessary to push in slightly on plug to release pin.

(3) Screw a No. 10 or AN3 bolt (used for a pulling handle) into screen assembly plug finger tight and withdraw screen assembly and spring. After removing screen, remove screw or bolt (pulling handle) from plug.

(4) Wash screen in solvent (P-D-680 or equivalent), using brush or air jet as required.

(5) Hold screen to light to test for cleanliness. Replace screen (P/N 561562) if required.

12. Installing Fuel Control Unit Screen - Airplanes NA-370-2 Through -6, and -8 and-9

A. To install fuel control unit screen, proceed as follows:

(1) Replace three "O" rings. (P/N 69890-136, P/N 69494S-18 and P/N 69494S-20.)

(2) Install ball, spring and screen assembly into fuel control unit.

(3) Turn plug end of screen assembly to align lockpin hole in plug with holes in fuel control body.

(4) Install lockpin, place pin retainer bracket in position and install pin bracket retaining screw.

(5) Safety retaining screw with lockwire.

(6) Check fuel control unit for fuel leaks after engine run.

13. Removing and Cleaning Fuel Pressurizing and Dump Valve Inlet Screen - Airplanes NA-370-2 Through -6, and-8 and-9

A. To remove and clean fuel pressurizing and dump (P & D) valve inlet screen, proceed as follows:

(1) Remove lockwire from P & D valve inlet screen plug, and remove plug.

(2) Remove P & D valve inlet screen.

(3) Wash screen in cleaning solvent (P-D-680, or equivalent), using brush or air jet as required.

(4) Hold screen to light to test for cleanliness.

14. Installing Fuel Pressurizing and Dump Valve Inlet Screen - Airplanes NA-370-2 Through -6, and-8 and-9

A. To install fuel pressurizing and dump (P & D) valve inlet screen, proceed as follows:

(1) Reinstall a clean screen into P & D valve housing.

(2) Install plug with a new preformed packing (P/N MS9020-12), and torque plug to 125-175 inch-pounds.

(3) Safety plug with lockwire.

(4) Check P & D valve for fuel leaks after engine run.

MAINTENANCE PRACTICES (Continued)

OPERATIONAL CHECK-OUT OF ENGINE FUEL RECOVERY SYSTEM — S/N 370-1 AND -7, AND 380-1 THRU -33, -35, -37 and -38 CHANGED BY S/B 75-2, AND 380-34, -36, -39 and SUBS

This check is performed with engine not running. Proceed as follows:

1. Pull LH & RH IGN & BOOST CONT circuit breakers.
2. Open both lower nacelle doors.
3. Disconnect lines at fuel recovery solenoid valve and dust cap lines.
4. Place ENGINE MASTER switch ON.
5. Position throttle past 80% thrust.
6. Fuel recovery solenoid valve should open. Blow through valve to ensure valve is open.
7. Retard throttle below 80% thrust.
8. Fuel recovery solenoid valve should close. Blow through valve to ensure valve is closed.
9. Position throttle to engine cut-off.
10. Place ENGINE MASTER switch OFF.
11. Connect fuel lines to solenoid valve and tighten.
12. Close both lower nacelle doors.
13. Close LH & RH IGN BOOST CONT circuit breakers.
14. Perform engine run-up and check for fuel leaks.

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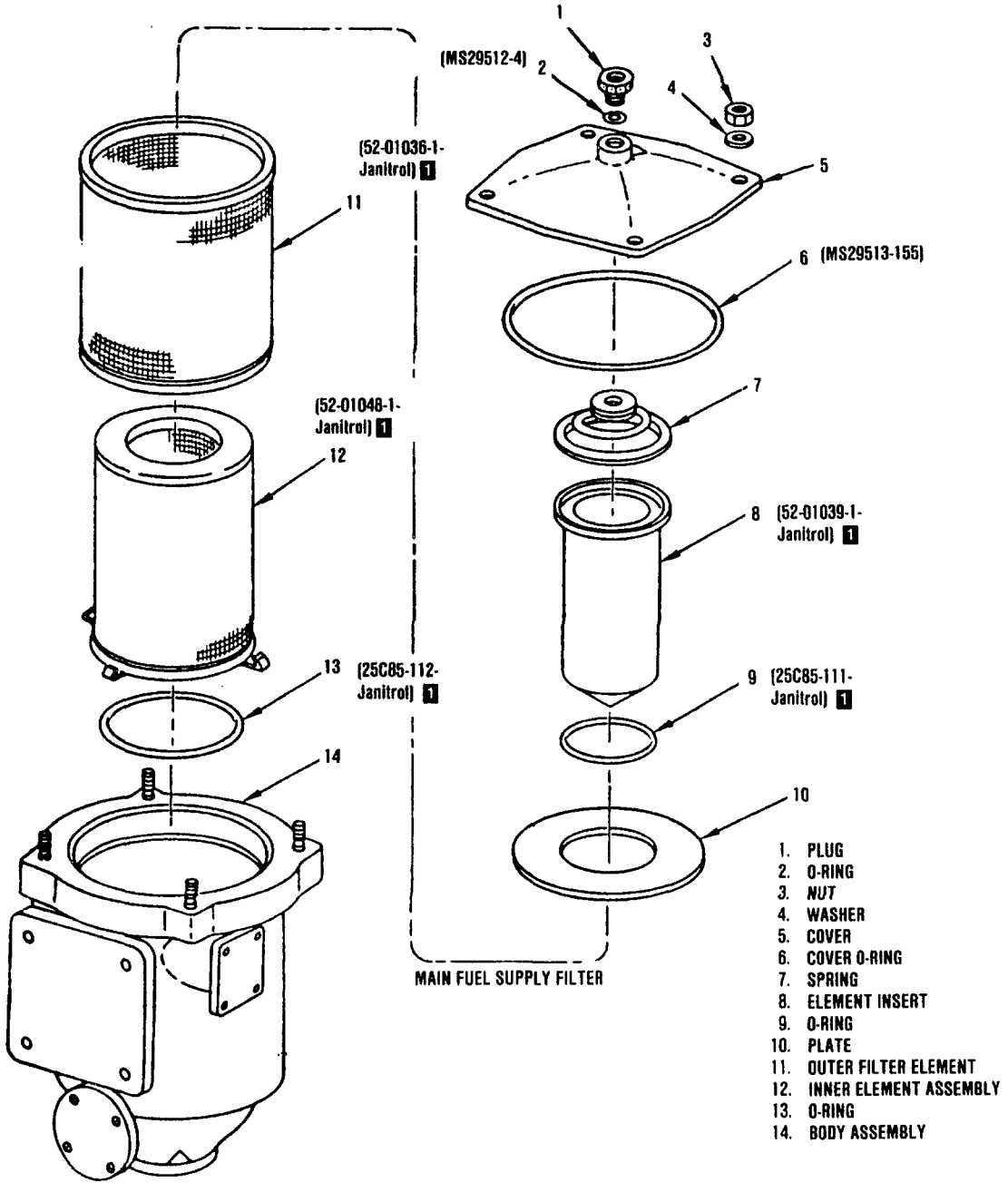

CAUTION

To prevent crushing of the filter elements, ensure that filter assemblies are stacked properly and held firmly in filter body prior to tightening the filter cap. The filter cover plug may be removed and a suitable rod inserted through the hole to securely hold the elements in the filter body.

1 Janitrol Aero Div. of Midland-Ross Corp.
4200 Surface Rd.
Columbus, Ohio 43228

Figure 73-00-1 — Engine Fuel and Control Filters (Sheet 1 of 2)

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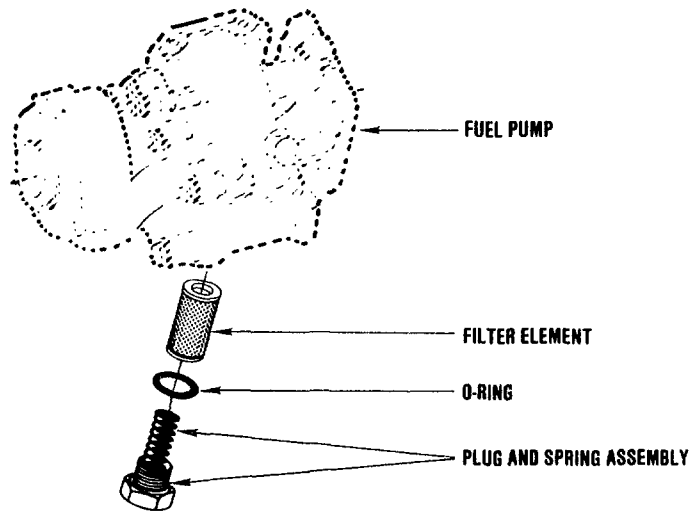
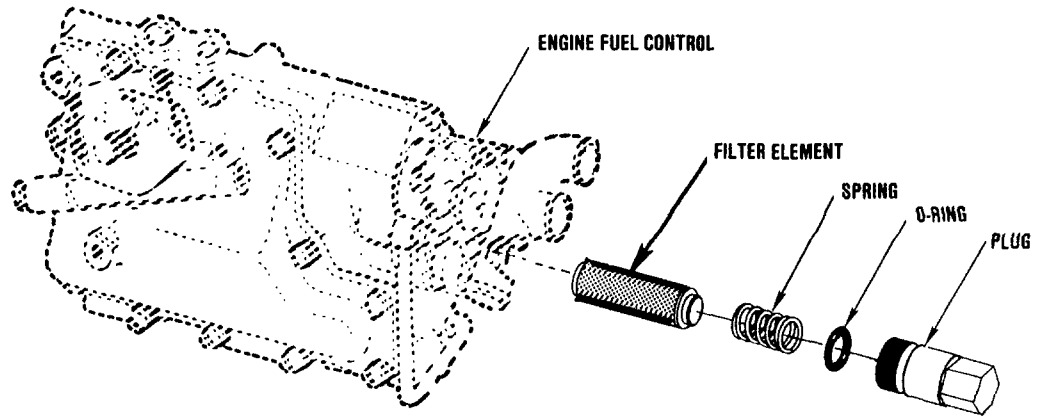
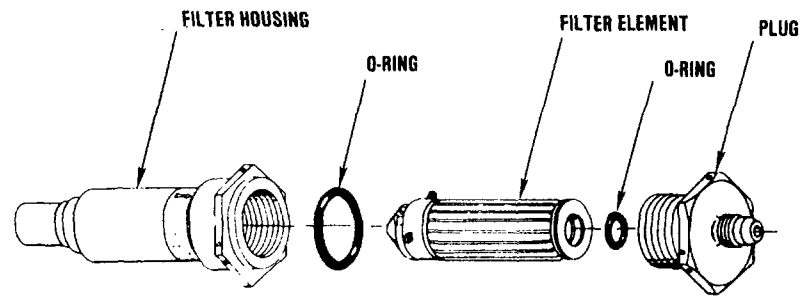


FUEL PUMP FILTER



ENGINE FUEL CONTROL FILTER



OVERSPEED GOVERNOR SERVO FILTER

5380 2-41.7

Figure 73-00-1 — Engine Fuel and Control Filters (Sheet 2 of 2)

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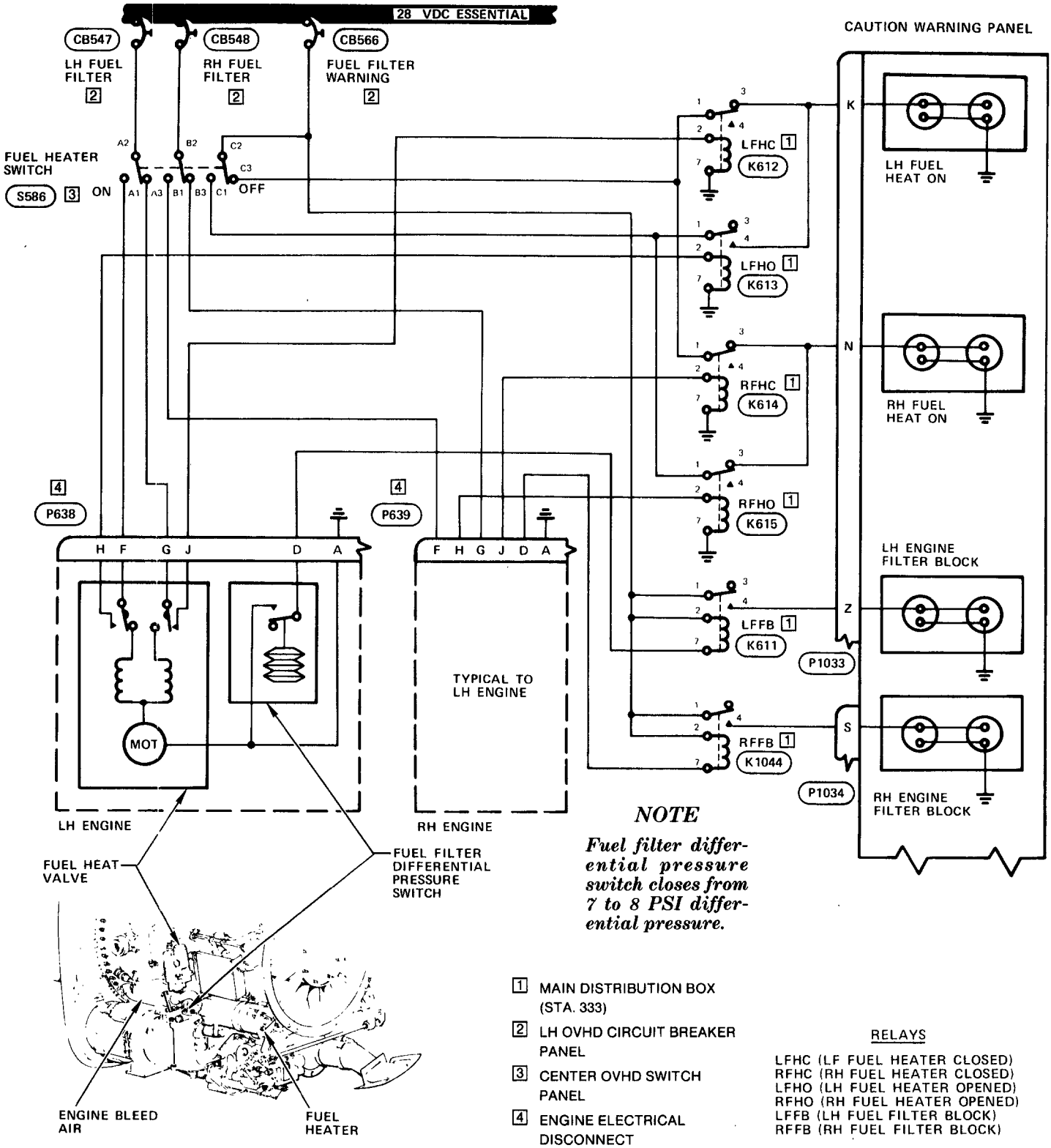

NOTE

Fuel filter differential pressure switch closes from 7 to 8 PSI differential pressure.

1 MAIN DISTRIBUTION BOX (STA. 333)

2 LH OVHD CIRCUIT BREAKER PANEL

3 CENTER OVHD SWITCH PANEL

4 ENGINE ELECTRICAL DISCONNECT

RELAYS

LFHC (LF FUEL HEATER CLOSED)
RFHC (RH FUEL HEATER CLOSED)
LFHO (LH FUEL HEATER OPENED)
RFHO (RH FUEL HEATER OPENED)
LFFB (LH FUEL FILTER BLOCK)
RFFB (RH FUEL FILTER BLOCK)

S380-2-41-18

Figure 73-00-2 — Fuel Filter Warning System — S/N 370-2 Thru -6 and -8 and -9

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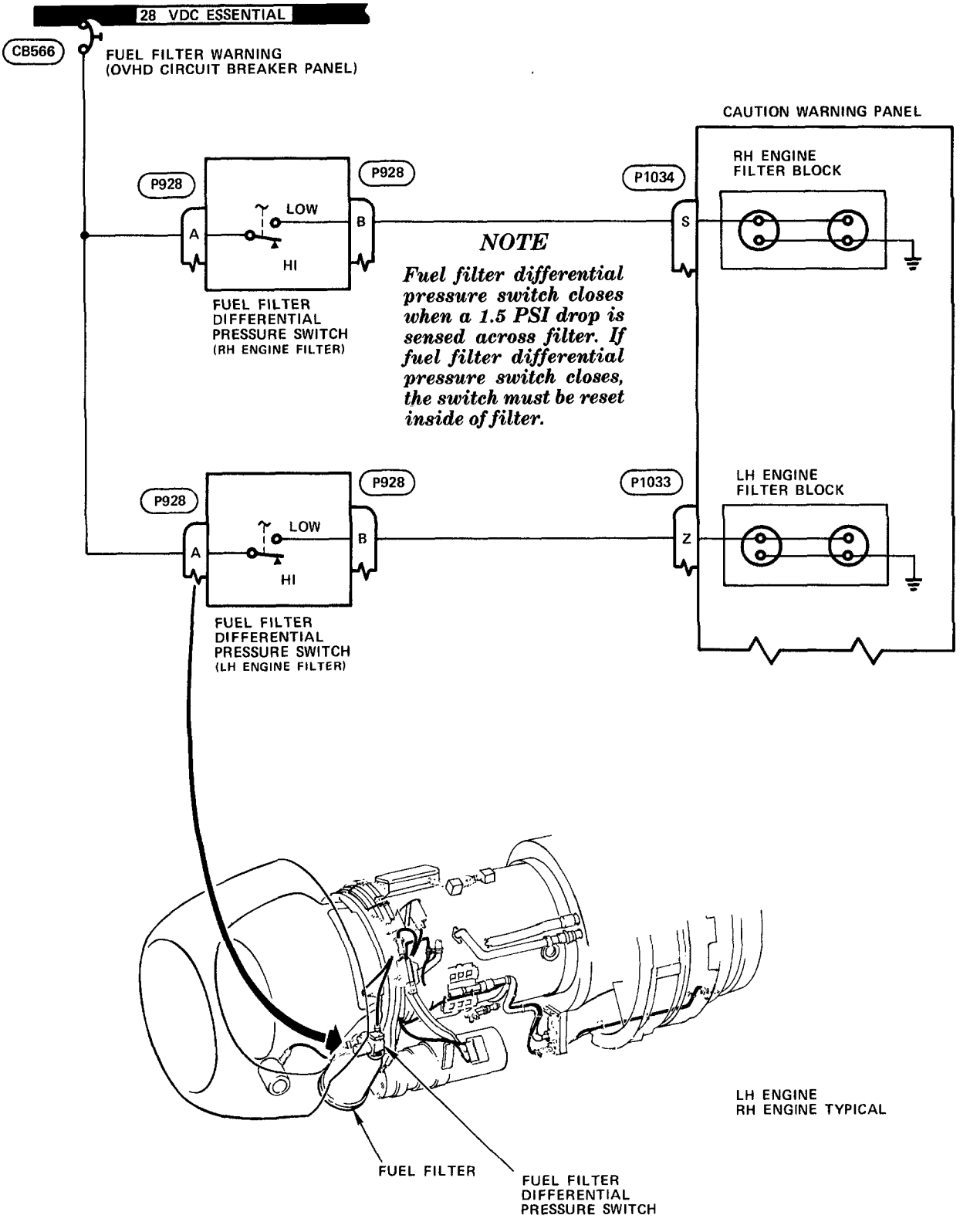

S 380-2-41-19

Figure 73-00-3 — Fuel Filter Warning System — S/N 370-1 -7, and 380-1 and Subs

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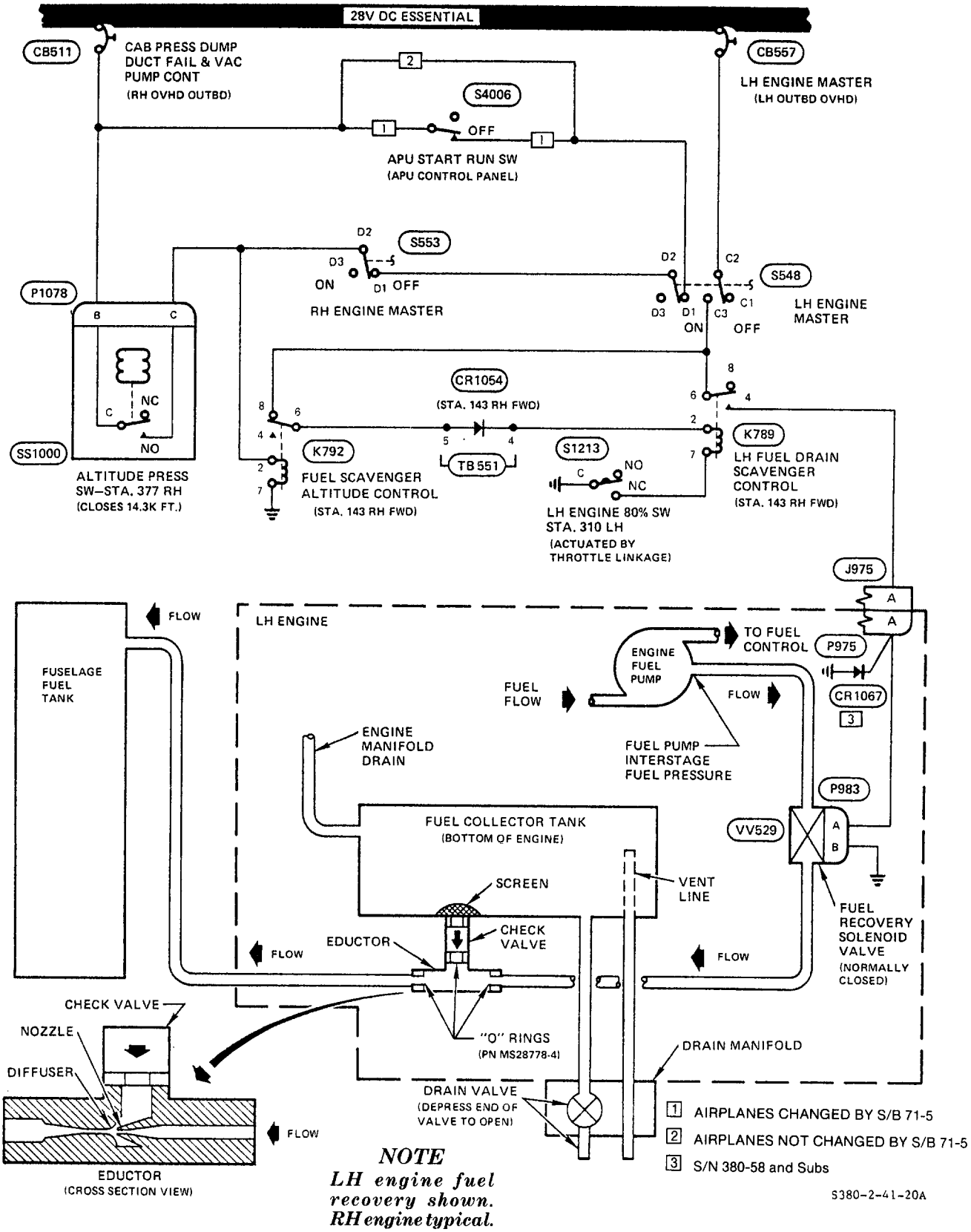

NOTE
LH engine fuel recovery shown. RH engine typical.

Figure 73-00-4 — Engine Fuel Recovery System — S/N 370-1 and -7, and 380-1 Thru -33, -35, -37 and -38 Changed By S/B 75-2, and 380-34, -36, -39 and Subs

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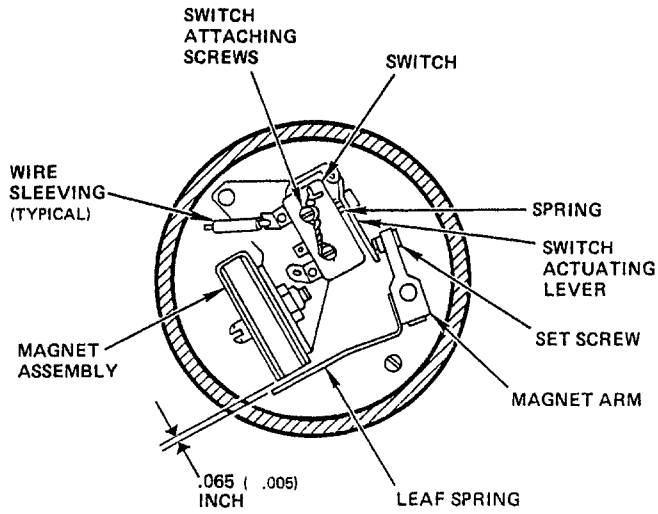

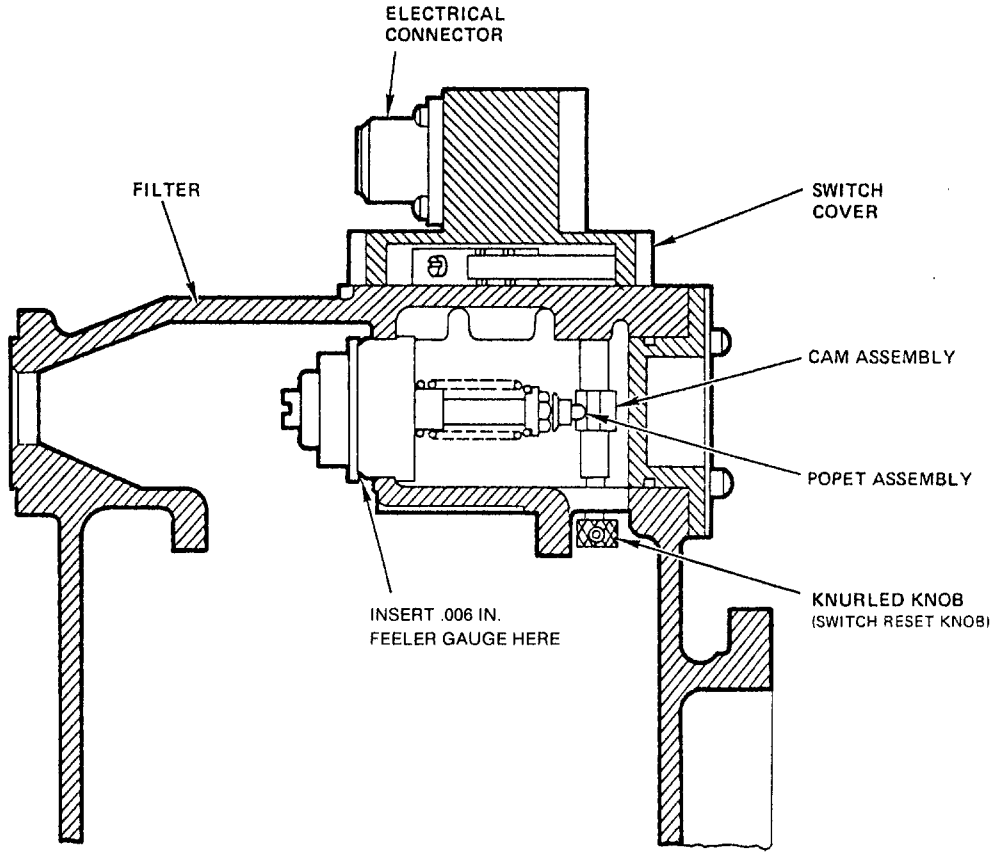

Figure 73-00-5 — Engine Fuel Filter Differential Pressure Switch — S/N 370-1, -7, and 380-1 and Subs

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MAINTENANCE PRACTICES (Continued)

REPLACE — FUEL HOSE, FUEL PUMP TO FILTER — S/N 370-1 and -7, 380-1 and Subs

1. Remove engine cowling to access fuel pump hose. See Figure 73-00-6 for hose location.
2. Remove and discard hose.
3. Install new hose between fuel pump and filter.
4. Run fuel boost pump and check for leaks.
5. Install engine cowling.

REPLACE — FUEL RECOVERY HOSE — S/N 370-1 AND -7 AND 380-1 THRU -33, -35, -37, AND -38 CHANGED BY S/B 75-2 AND S/N 380-34, -36, -39 AND SUBS

1. Open engine cowling to gain access to the flexible fuel recovery hose(s). Refer to Figure 73-00-7 for hose location and system layout.
2. Disconnect and discard flexible fuel recovery hose(s).
3. Install new flexible fuel recovery hose(s).
4. Start both engines, then shut down.
5. Inspect fuel recovery hose(s) for leaks.
6. Close cowling and secure airplane as required.

REPLACE — ENGINE FUEL HOSES — PRESSURE TAP AND LOW PRESSURE HOSES — S/N 370-1 AND -7, 380-1 AND SUBS

1. Remove cowling to access fuel pump pressure and tap hoses. Refer to Figure 73-00-8 for hose location.
2. Remove and discard hoses.
3. Install new fuel pump pressure and tap hoses.
4. Run fuel boost pump to lead check hoses.
5. Install engine cowling.
6. Secure airplane as necessary.

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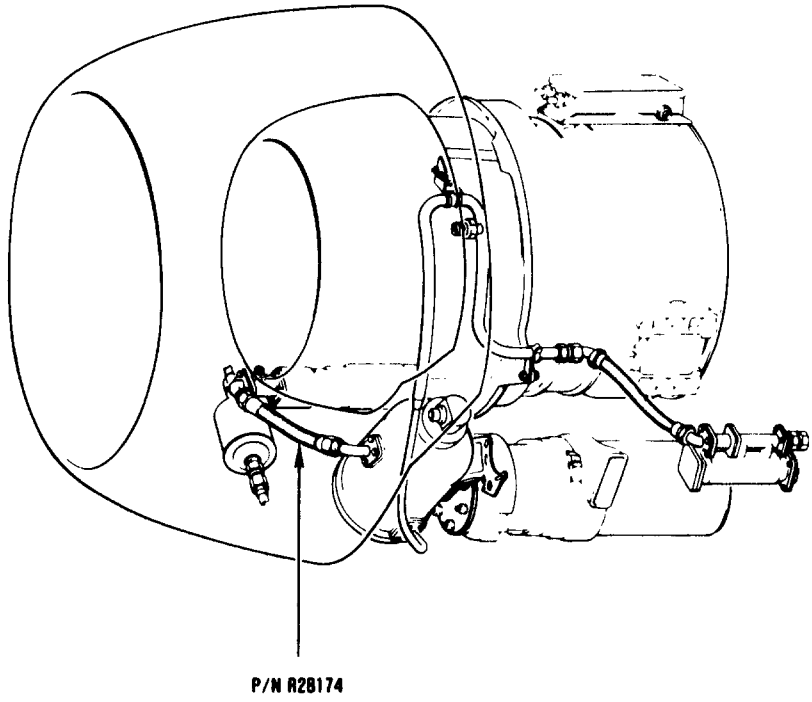

Figure 73-00-6 — Fuel Hose, Fuel Pump to Filter

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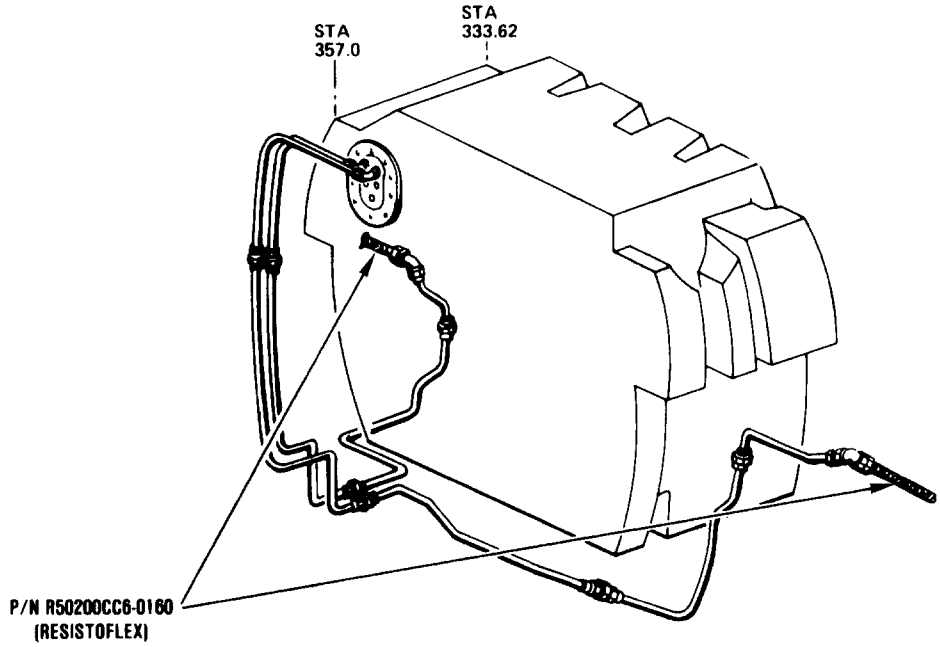

Figure 73-00-7 — Fuel Recovery Hose

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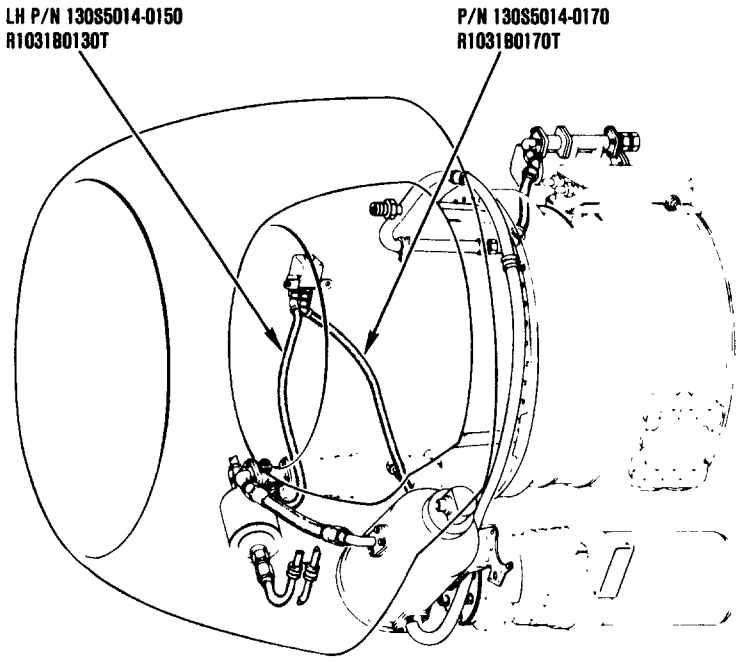

LH ENGINE SHOWN, RH TYPICAL

Figure 73-00-8 — Engine Fuel Hoses, Pressure Tap and Low Pressure Hoses

NEW

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ENGINE FUEL RECOVERY SYSTEM - TROUBLE SHOOTING

1. Trouble Shooting Engine Fuel Recovery System - Airplanes NA-370-1 and -7, and Airplanes NA-380-1 through -33, -35, -37 and -38 Changed by Service Bulletin 75-2, and Airplanes NA-380-34, -36, -39 and Later

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. FUEL DRAINING FROM FUEL COLLECTOR TANK VENT LINE.		
(1) Fuel collector tank is full by too many consecutive engine starts and stops under 80% thrust.		Drain tank with manual drain valve at engine drain manifold.
(2) Faulty fuel recovery solenoid valve.	Pull "LH & RH IGN & BOOST CONT" circuit breakers. "ENGINE MASTER" switch ON. Position throttle past 80% thrust (engine not running). Check for 28-volt dc between pins "A" and "B" at fuel recovery solenoid valve electrical connector.	If correct voltage is present, replace valve.
(3) Faulty fuel recovery solenoid valve circuitry.	Check continuity between solenoid valve electrical plug and essential bus. (See figure 73-00-4).	Replace defective relay, switch or wiring as necessary
(4) Faulty check valve between fuel collector tank and eductor.	Pressure check, check valve.	If check valve is faulty, replace check valve.
(5) Plugged fuel collector tank screen.		Clean screen.

CHAPTER 74

IGNITION

NOTICE

The following information applies to NA 265-70 airplanes only.

CHAPTER 74

IGNITION

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ORIGINAL
As Received By
ATP

CHAPTER 74

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74-00-1		101	Mar 1/70
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		402	Mar 1/70
74-00-2		501	Mar 1/70
		502	Jul 6/73

*The asterisk indicates pages revised, added, or deleted by the current revision.

1. General

The engine ignition system for each engine consist of two igniter units mounted on the engine, two spark-igniter plugs located in the No. 3 and No. 6 combustion chamber, an ignition control relay and an ignition stall signal warning relay. An engine start switch for each engine, an air-start switch, and a continuous ignition switch are located on the center pedestal. A throttle-actuated fuel boost and ignition switch is located inside the center pedestal.

When a normal or airstart is made. a pulsating high-tension voltage is supplied to both spark-igniters to ignite the fuel-air mixture. After "light-off" occurs in the No. 3 and No. 6 combustion chambers, the remaining burners are ignited through the interconnecting flame tubes which also sustain continuous combustion. When a start is accomplished, releasing the engine start switch deactivates the spark igniters. For an airstart, positioning the airstart switch to start the desired engine, and advancing the engine's throttle to idle, activates both spark igniters on the selected engine. The spark igniters will remain operative until the airstart switch is moved to OFF. A two position continuous ignition switch is to be used during flight in turbulent air. With the switch in the continuous position, the spark-igniter plug in the No. 3 combustion chamber of each engine will operate continuously.

2.. Operation of Engine Ignition System - Normal

Operational sequence of the engine ignition system is the same on the left and right engine. Only the right engine ignition system is described. During a normal engine ground start, power from the essential bus is applied through the ON position of the right "ENGINE MASTER" switch, and to the right engine start switch. When the start switch is pushed in, a circuit is completed between the essential bus and the right start power monitor relay. When the engine rpm reaches 10% during the starting cycle, advancing the right engine throttle from CLOSED position actuates the fuel boost and ignition switch. This, in turn, de-energizes the right fuel boost pump control relay and completes a circuit through the energized right engine start power monitor relay and the ignition control relay to the right engine ignition units and to the right engine ignition-on light (red) on the right engine start button. The ignition-on light comes on, indicating that power is being supplied to the right engine ignition units. The energized ignition units deliver high-tension voltage to the spark-igniter plugs in No. 3 and 6 combustion chambers. When the engine rpm reaches about 35%, the right engine start controller drops out, de-energizing the holding coil in the right engine start switch (start switch pops out). This, in turn, de-energized the right start power monitor relay. When the start power monitor relay is de-energized, the circuit to the spark-igniters and the ignition-on light is opened. (See figure 74-00-1).

3. Operation of Engine Ignition System - Continuous

The continuous ignition switch, when placed in the CONTINUOUS position energizes the continuous ignition on both engines. With the switch at continuous electrical power passes directly from the throttle-actuated fuel boost and ignition switch to the spark igniter on each engine. Continuous ignition occurs in the No. 3 combustion chamber until the continuous ignition switch is again placed at NORMAL. The red ignition on indicator light will remain on as long as the switch is at CONTINUOUS.

If the airplane approaches a stall condition, both the continuous and intermittent ignition units are energized automatically regardless of switch position, by the stall warning system, and remain energized for the duration of the stall condition. The stall warning computer energizes the stall warning signal relay at 15 percent above the engine stall, or flame-out airspeed. Ignition is continuous for the duration of the stall warning, and the ignition-on light is illuminated.

4. Operation of Engine Ignition System - Air Start.

The airstart system is composed of two independent systems: one for the left engine and one for the right engine. These systems are controlled by the "AIRSTART" switch and the corresponding "ENGINE MASTER" switch. During an airstart, battery power is used to supply the necessary voltage to energize both engine ignition units. To energize the essential bus with battery power, refer to Chapter 24.

All current to the ignition units must pass through the corresponding throttle-actuated fuel boost and ignition switch; therefore, the throttle must be advanced to IDLE for an airstart. Advancing the throttle closes the fuel boost and ignition switch (normally closed contacts). Power from the essential bus is then supplied to the engine ignition units and to the ignition-on light, in the engine start button. The ignition-on light will remain on and both spark-igniters will remain in operation as long as the airstart switch is at on. After the engine has started, moving the airstart switch to OFF will de-energize both ignition units.

5. Engine Start Switches

A solenoid-held, push-pull starter button is provided for each engine. These PUSH TO START pull to stop starter buttons are on the engine switch panels, one on each side of the throttle quadrant, and are used when normal battery or external power starts are made. When the starter button is actuated, a solenoid holds the button down during the cranking portion of the starting cycle. A relay in the starter controller automatically de-energizes the ignition and releases the starter button when the engine has accelerated to approximately 31 to 35 percent rpm and the starter load has decreased to approximately 200 amperes when using external electrical power for starting. During battery starts, the rpm may be somewhat higher. There is a red starting cycle light in each starter button which illuminates during ground or air engine starting and indicates when ignition i s on. This light goes out when the starting cycle and ignition ends and the starter-generator begins to function as a generator. The electrically powered starting cycle can be interrupted at any time by pulling the starter button up.

6. "AIRSTART" Switch

The "AIRSTART" switch is used to supply 28-volt dc power from the essential bus to the air start relay and the engine ignition units during an airstart. The "AIRSTART' switch is mounted on the center pedestal. A 55-ohm resistor is placed in the airstart circuit to protect the coils of the ignition control relay. (On airplanes changed by Service Letter No. 3 a 75 ohm resistor is used). This is to protect the relays should the airstart switch be left in the ON position for a prolonged period of time.

7. Pilot Selected Continuous Ignition Switch

A pilot selected, lever type, two position ignition switch is mounted on the center pedestal. The switch has two positions labeled, NORMAL and CONTINUOUS. With the switch at NORMAL, both the intermittent and continuous-ignition units are de-energized when the engine start switch is disengaged. At any time the ignition selector switch is placed at CONTINUOUS and the throttle is advanced, the continuous ignition unit for combustion chamber No. 3 is energized and the ignition on indicator red light in the engine start button is on. Placing the ignition selector switch at NORMAL de-energizer the ignition unit, and the ignition-on light goes out.

8. Engine Ignition Units

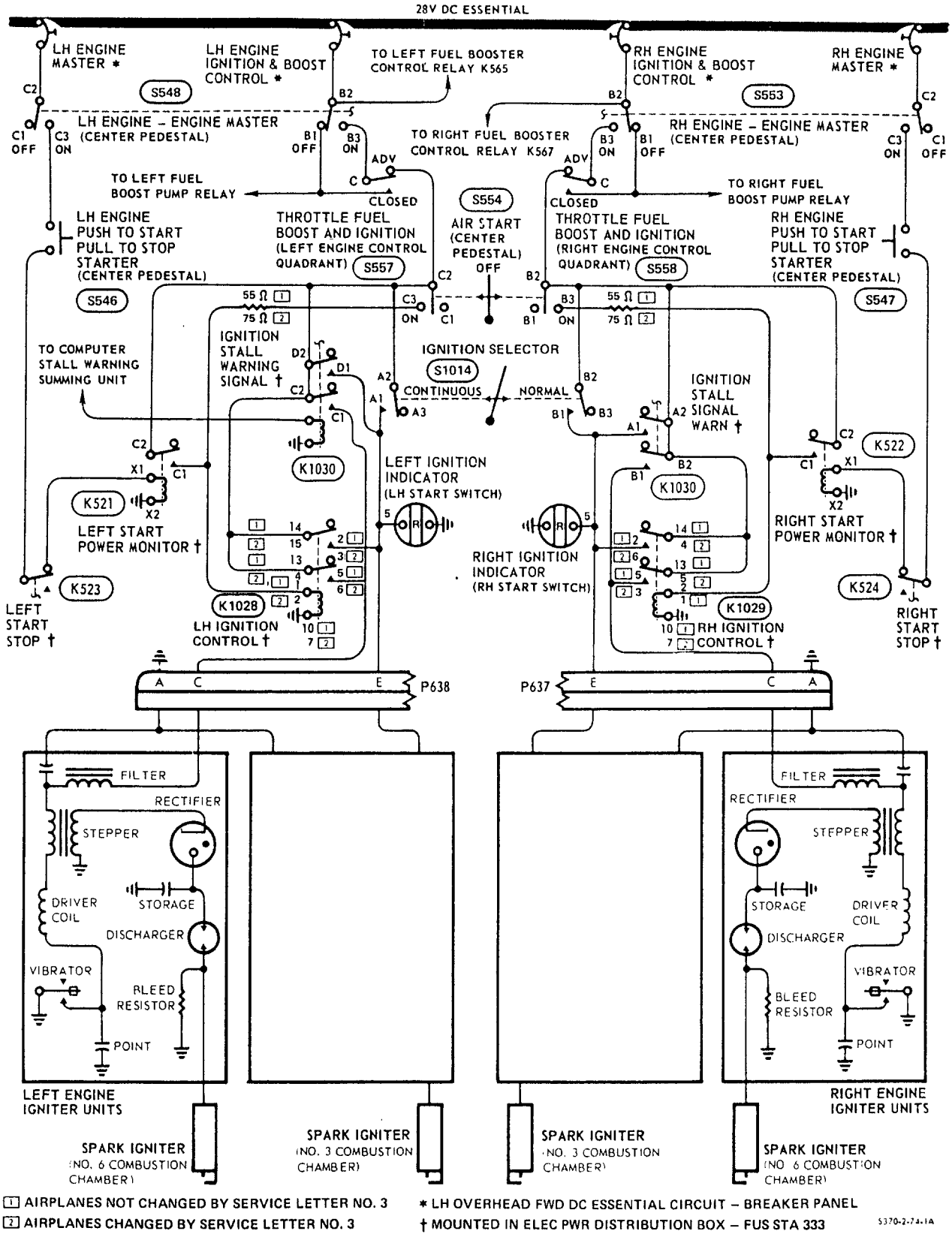
Two engine ignition units are mounted on the engine compressor case. one on each side. The ignition units supply rapidly pulsating high-tension voltage to the igniter plugs for engine ignition. Each unit operates independently and will operate satisfactorily from 10- to 29-volt dc input. The spark rate (which depends on the input voltage) is 1.6 to 3.3 per second for the continuous duty ignition exciter and 1 per second (minimum) for the intermittent duty ignition exciter. Each unit consists of an input filter, a stepper transformer, a rectifier tube, a vibrator driver coil, two capacitors, a discharge tube, and bleed resistors. (See figure 74-00-1.)

9. Engine Spark-igniter Plugs

Two engine spark-igniter plugs, one in combustion chamber No. 3 and one in No. 6, convert high-tension voltage from the ignition units into spark for normal engine starts and airstarts. Each igniter plug comprises a plug case, an electrical connector, a ceramic-insulated electrode, and an electrode seal. An electrical lead from each ignition unit is connected to its respective igniter plug, and, upon installation, each plug is grounded to the engine. When high-tension voltage is applied to the igniter plug, current flows through the center electrode and arcs, across the gap, to ground.

10. Ignition Control Relays.

Ignition control relays, one for each engine, are mounted in the electrical power distribution box at fuselage station 333. Power from the advanced side of the throttle-actuated fuel boost and ignition switch energizes the relays and furnishes ignition during a normal start or airstart. The relay is de-energized upon completion or termination of the start cycle.



[1] AIRPLANES NOT CHANGED BY SERVICE LETTER NO. 3
[2] AIRPLANES CHANGED BY SERVICE LETTER NO. 3

* LH OVERHEAD FWD DC ESSENTIAL CIRCUIT – BREAKER PANEL
† MOUNTED IN ELEC PWR DISTRIBUTION BOX – FUS STA 333

5370-2-74-1A

Engine Ignition System Electrical Schematic
Figure 74-00-1

11. Stall Warning Signal Relays

A stall warning signal relay, one for each engine, is mounted in the electrical power distribution box at fuselage station 333. If the airplane approaches a stall condition, the stall warning computer energizes the stall warning signal relays at 15 percent above the engine stall, or flame-out speed. Both the continuous and intermittent ignition units are energized automatically for the duration of the stall and the ignition on red indicator light in the start button is on.

12. Throttle-actuated Fuel Boost and Ignition Switch

The throttle-actuated fuel boost and ignition switch is mounted inside the center pedestal. This switch controls the engine ignition and fuel boost pump circuits during engine start. When the engine throttle is in closed position, power is supplied to the fuel boost pump control relay, opening the circuit to the fuel boost pump. When the engine throttle is advanced to IDLE, the circuit to the fuel boost pump control relay is opened and the circuit to the engine ignition units is closed.

ENGINE IGNITION SYSTEM - TROUBLE SHOOTING

1. Trouble Shooting Engine Ignition System

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. NO SPARK AT IGNITER PLUGS.		
(1) Engine ignition circuit breaker not closed.	Visually check.	Close the following circuit breakers: "LH ENGINE - IGN & BOOST," "RH ENGINE - IGN & BOOST," "LH ENGINE - MASTER," and "RH ENGINE - MASTER."
(2) Faulty spark-igniter plugs.	Test ignition system. (Refer to 74-00-2.)	Replace faulty plugs.
(3) Faulty throttle-actuated fuel-ignition switch.	Open either left or right fuel boost pump circuit breaker, depending upon engine ignition system being checked. Place corresponding "ENGINE MASTER" switch to ON and move throttle to IDLE. With multimeter, check for 28-volt dc at normally open contact.	Replace faulty switch.
(4) Faulty start power monitor relay.	Refer to Chapter 80.	Replace faulty relay.
(5) Faulty ignition unit (exciter).	Remove either engine access door on right engine nacelle or engine access door on left engine nacelle. Disconnect electrical disconnect plug P638 (left engine) or P637 (right engine). Test ignition system. (Refer to 74-00-2.) With multimeter, check for 28-volt dc at pins "C" and "E" of electrical plug P637 on right engine or P638 on left engine.	Replace faulty engine ignition unit (exciter).

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
B. INDICATOR LIGHT IN START SWITCH DOES NOT COME ON DURING ENGINE START.		
(1) Faulty light bulb.	Place "ENGINE MASTER" switch (left or right) to ON and move corresponding throttle to IDLE. Place "AIRSTART" switch to ON (left or right) corresponding to engine ignition system being checked. With multimeter, check for 28-volt dc at terminal "5" on corresponding start switch.	Replace faulty light bulb.

ENGINE IGNITION SYSTEM - MAINTENANCE PRACTICES

1. Removing Engine Spark-igniter Plugs

A. The engine spark-igniter plugs are threaded into combustion chamber No. 3 and 6. For access to these plugs, open lower engine access doors and proceed as follows:

WARNING: BECAUSE THE VOLTAGE TO THE SPARK IGNITERS IS DANGEROUSLY HIGH, THE "AIRSTART" AND APPLICABLE "ENGINE MASTER" SWITCHES MUST BE IN OFF POSITION BEFORE ANY OF THE IGNITION SYSTEM COMPONENTS ON THE ENGINE ARE REMOVED OR CHECKED. A SUFFICIENT PERIOD OF TIME MUST ELAPSE BETWEEN THE OPERATION OF THE IGNITION SYSTEM AND THE REMOVAL OR CHECKING OF COMPONENTS, TO ENSURE THE COMPLETE DISSIPATION OF ELECTRICAL ENERGY FROM THE IGNITION SYSTEM.

(1) Make sure that "AIRSTART" and applicable "ENGINE MASTER" switches are at OFF.

(2) Disconnect igniter lead from spark-igniter plug. Hold insulated part of lead, and ground terminal to engine. (This action discharges the capacitors in the ignition units.)

WARNING: DO NOT TOUCH IGNITION ELECTRICAL LEAD TERMINAL. IGNITION VOLTAGE IS DEADLY. FAILURE TO DISSIPATE THE ELECTRICAL CHARGE STORED IN THE IGNITION UNIT COULD CAUSE SERIOUS INJURY TO PERSONNEL.

(3) Screw igniter plugs out of engine combustion chamber.

(4) Cap ignition leads and boss in diffuser case. Wrap igniter plug in suitable material for protection.

2. Installing Engine Spark-igniter Plugs

A. Before installing spark-igniter plugs, check the following:

(1) Visually check barrel threads and shell threads. Chase threads if necessary, using a 0.625-24NS die for barrel thread and a 0.750-20NS die for shell threads.

(2) Visually check exposed ceramic section of spark-igniter plug. Any cracks are cause for rejection.

(3) Visually check spark-igniter for erosion. Erosion to a limit of 0.225 inch below the surface of the ceramic is acceptable.

B. To install spark-igniter plugs, proceed as follows:

(1) Coat threads of spark-igniter plug with BG Mica Lube A-768 antiseize compound (manufactured by BG Corp, Ridgefield, N.J.), and install gasket.

(2) With fingers, thread spark-igniter plug into boss in diffuser case until it bottoms on gasket.

(3) Torque spark-igniter plug to 300-360 inch-pounds.

(4) Carefully insert lead connector into igniter plug and tighten nut finger-tight.

(5) Torque nut to 140-160 inch-pounds. After torquing nut, check that nut skirt bottomed on mating part. Safety nuts with lockwire.

3. Removing Engine Ignition Units

A. The engine ignition units are mounted on brackets, one on each side of the engine compressor section. The ignition units installed on the inboard side of each engine (engine installed) are not accessible. To gain access to these units, remove engine. The ignition units installed on the outboard side of each engine are accessible when the lower engine access door is open. To remove ignition units, proceed as follows:

(1) Make sure that "AIRSTART" and applicable "ENGINE MASTER" switches are at OFF.

(2) Disconnect spark-igniter lead from spark-igniter plug. Hold insulated part of lead, and ground terminal to engine. (This action discharges the capacitors in the ignition units.)

WARNING: DO NOT TOUCH IGNITION ELECTRICAL LEAD TERMINAL. IGNITION VOLTAGE IS DEADLY. FAILURE TO DISSIPATE THE ELECTRICAL CHARGE STORED IN THE IGNITION UNIT COULD CAUSE SERIOUS INJURY TO PERSONNEL.

(3) Disconnect electrical harness and spark-igniter lead from ignition unit (exciter).

(4) Remove bolts securing ignition unit to mounting bracket. Remove ignition unit.

4. Installing Engine Ignition Units

A. To install engine ignition units, proceed as follows:

(1) Position ignition unit on mounting bracket. Secure ignition unit with bolts.

(2) Connect lead to ignition unit and torque to 140-160 inch-pounds. Safety nut with lockwire.

1. Testing Engine Ignition System

WARNING: FOR THIS TEST, DO NOT REMOVE SPARK-IGNITER PLUGS FROM ENGINE. PRESENCE OF COMBUSTIBLE VAPORS COULD RESULT IN AN EXPLOSION OR FIRE.

- DO NOT CHECK IGNITION IMMEDIATELY AFTER AN ATTEMPTED ENGINE START OR AT ANY TIME EXCESSIVE FUEL OR FUEL VAPOR IS IN ENGINE OR TAIL PIPE.

A. To test engine ignition system, proceed as follows:

(1) To prevent operation of fuel boost pumps during the following test, pull (open) "BOOST PUMP - LH FUEL" and "BOOST PUMP - RH FUEL" circuit breakers.

(2) Close "LH ENGINE - IGN & BOOST" and "RH ENGINE - IGN & BOOST" circuit breakers.

(3) If battery is not installed in airplane, connect a 28-volt dc power source to external utility power receptacle.

(4) If battery power is used, position "ELECTRIC MASTER" and battery switches at ON.

(5) Position "LH ENGINE - ENGINE MASTER" switch at ON and "AIRSTART" switch at LEFT HAND ON.

NOTE: Before placing "LH ENGINE - ENGINE MASTER" switch to ON, pull "LH ENGINE - MASTER" circuit breaker. This prevents the fuel shutoff valve from opening during the ignition check.

(6) Advance left engine throttle from CLOSED to IDLE. If the spark-igniter plugs are firing (arcing), the firing can be heard from each side of the engine nacelle.

(7) Move left engine throttle to CLOSED. Spark igniters should stop firing.

(8) Place "LH ENGINE - ENGINE MASTER" switch at OFF.

(9) Position "RH ENGINE - ENGINE MASTER" switch at ON and "AIRSTART" switch at RIGHT HAND ON.

NOTE: Before placing "RH ENGINE - ENGINE MASTER" switch to ON, pull "RH ENGINE - MASTER" circuit breaker. This prevents the fuel shutoff valve from opening during the ignition check.

(10) Advance right engine throttle from CLOSED to IDLE. If spark-igniter plugs are firing (arcing), the firing can be heard from each side of engine nacelle.

(11) Return right engine throttle to CLOSED.

(12) Place "AIRSTART," "RH ENGINE - ENGINE MASTER," "BATTERY," and "ELECTRIC MASTER" switches to OFF.

(13) If external power source was used, remove power source from airplane.

(14) Reset "BOOST PUMP - LH FUEL" and "BOOST PUMP - RH FUEL" circuit breakers.

2. Testing Engine Ignition Units

A. The service life of the continuous-ignition system is equivalent to engine time before overhaul when the following maintance is accomplished.

(1) Check operation of spark-igniter plugs as follows:

(a) Remove both spark-igniter plugs. With plugs removed, connect electrical leads and ground shell of both plugs securely to engine.

(b) Station a man in vicinity of engine to watch spark-igniter plugs.

WARNING: STAY CLEAR OF SPARK-IGNITER PLUGS DURING TESTS, AS THE DISCHARGE CAN BE LETHAL.

(c) Disconnect "BOOST PUMP - LH FUEL" and "BOOST PUMP - RH FUEL" circuit breakers. Position "ELECTRIC MASTER," "BATTERY" and ENGINE MASTER switches for engine being checked to ON.

(d) Advance throttle on engine being checked to idle.

(e) Depress engine start button. Both spark-igniter plugs should fire, as evidenced by the appearance of a spark.

(f) Disengage start button. Both spark-igniter plugs should stop discharging. The right spark-igniter plug should stop discharging when the start button is disengaged, unless the ignition selector switch on the center pedestal is at CONTINUOUS.

(g) Return engine control lever to cutoff.

(h) Advance engine throttle to idle and turn "AIRSTART" switch on. Both spark-igniter plugs should fire.

(i) Turn "AIRSTART" switch off. Both spark-igniter plugs should stop firing.

(j) Position "ELECTRIC MASTER," "BATTERY" and ENGINE MASTER switches at OFF and reset circuit breakers and re-install both spark-igniter plugs.

NOTICE

The following information applies to NA 265-80 airplanes only.

CHAPTER 74

IGNITION

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CHAPTER 74

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ENGINE IGNITION SYSTEM - DESCRIPTION AND OPERATION

1. General

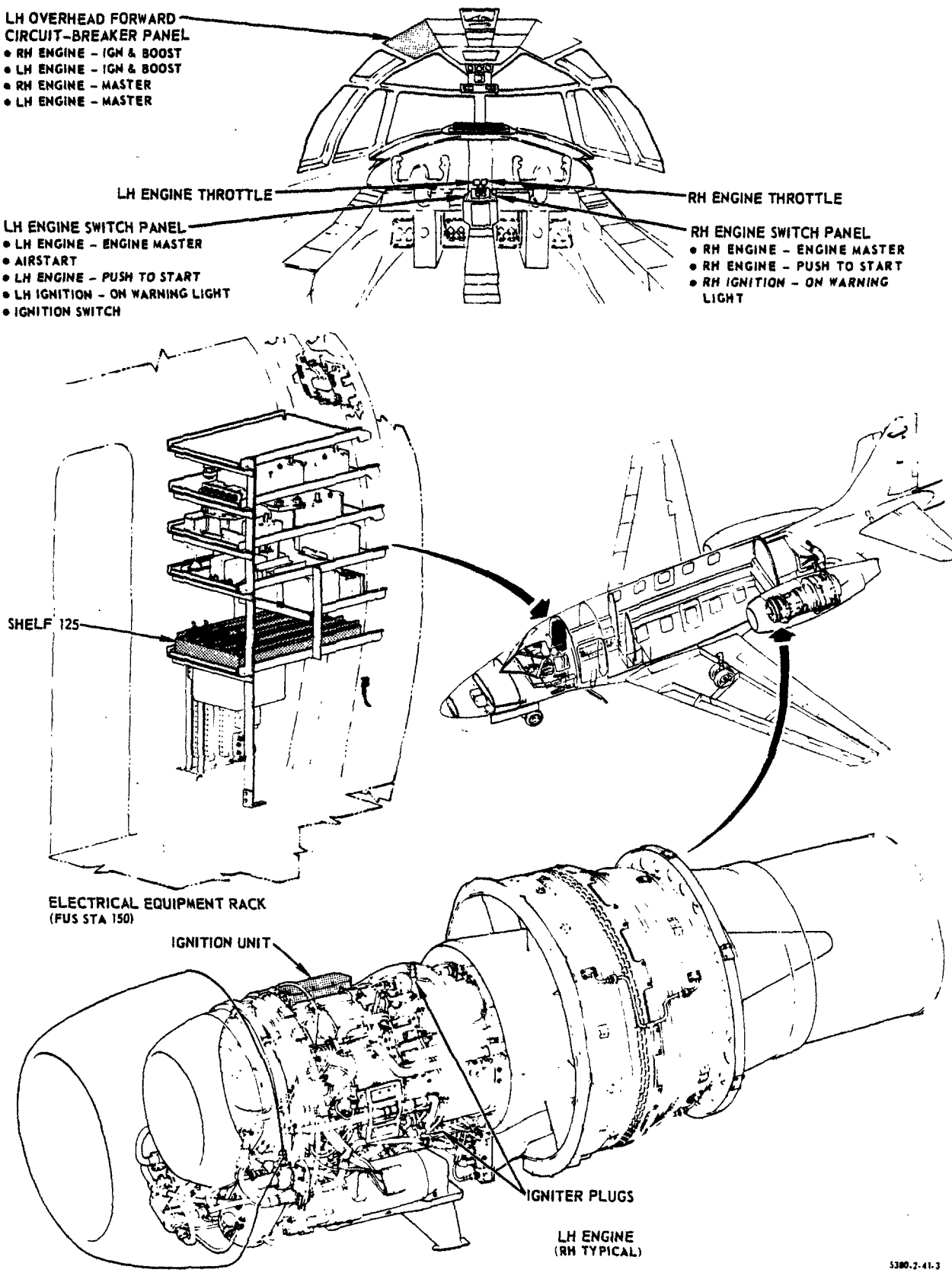
The ignition system for each engine consists of an igniter unit mounted on the engine, two igniter plugs located in the combustion chamber at 8 and 12 o'clock positions, an ignition control relay, and control circuitry. (See figures 74-00-1 and 74-00-2.) The control circuitry for each engine ignition system contains an airstart switch, an ignition selector switch, and the start switch and No. 1 start control relay which are primarily starter system components. The engine master switch and throttle fuel boost and ignition switches in the throttle quadrant are also part of the ignition control circuitry. When a normal or air start is made, a pulsating, high tension voltage is supplied to both spark-igniters to ignite the fuel-air mixture. At an engine speed of 42% rpm, the ignition system is de-energized and the combustion in the single annular combustion chamber is self-sustaining. A continuous position on the ignition selector switch allows the ignition to be operated continuously, by-passing the start control circuitry. This is used primarily for flight in turbulent air. For an airstart, positioning the airstart switch to the desired engine and advancing the engine's throttle to idle activates both spark igniters on the selected engine. The spark igniters will remain operative until the airstart switch is moved to OFF or the throttle closed. Both engine ignition systems are similar, therefore, only the right engine system will be described.

2. Operation of Engine Ignition System - Normal

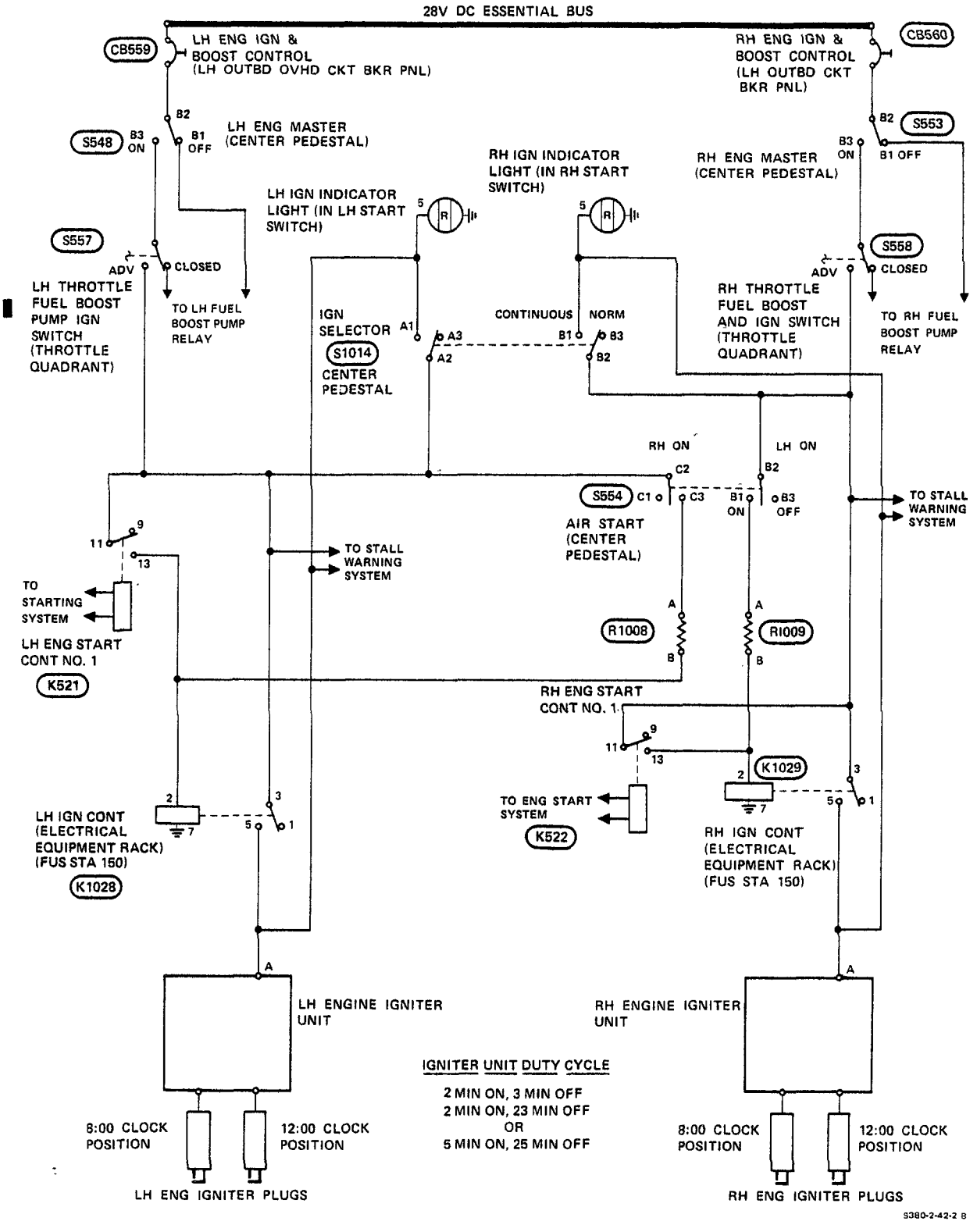
Normal operation of the ignition system occurs during the starting cycle. With the right engine master switch on and the starter switch depressed, 28-volt dc power from the essential bus is connected to normally open contacts in the throttle fuel boost and ignition switch. When the throttle is moved to idle, at about 10% rpm during the start cycle essential bus power is connected to the coil of the ignition control relay through the No. 1 engine start control relay. The ignition control relay energizes and connects essential bus power to the solid state ignition unit and to the ignition indicator light in the start button. The ignition unit generates the high power 3,000 volts to the igniter plugs. When engine rpm reaches 42%, the start system speed control (Chapter 80) causes the No. 1 start control relay to de-energize which in turn de-energizes the ignition control relay, disconnecting power to the ignition unit. The ignition system may also be energized automatically through actuation of the stall warning system.

3. Operation of Engine Ignition System - Continuous

With the ignition selector switch at continuous, the No. 1 engine start relay and ignition control relay are bypassed so that the start system operation has no affect on the ignition system. Any time the essential bus is energized, the engine master switch is on, and the throttle at idle the ignition system will be on.


LH OVERHEAD FORWARD CIRCUIT-BREAKER PANEL
• RH ENGINE – IGN & BOOST
• LH ENGINE – IGN & BOOST
• RH ENGINE – MASTER
• LH ENGINE – MASTER
LH ENGINE THROTTLE
RH ENGINE THROTTLE
LH ENGINE SWITCH PANEL
• LH ENGINE – ENGINE MASTER
• AIRSTART
• LH ENGINE – PUSH TO START
• LH IGNITION – ON WARNING LIGHT
• IGNITION SWITCH
RH ENGINE SWITCH PANEL
• RH ENGINE – ENGINE MASTER
• RH ENGINE – PUSH TO START
• RH IGNITION – ON WARNING LIGHT
SHELF 125
ELECTRICAL EQUIPMENT RACK
(FUS STA 150)
IGNITION UNIT
IGNITER PLUGS
LH ENGINE
(RH TYPICAL)
5380.2-41-3

Engine Ignition System Component Location
Figure 74-00-1



Engine Ignition System Electrical Schematic
Figure 74-00-2

4. Operation of Engine Ignition System - Airstart

During an airstart, the "AIRSTART" switch, when positioned at ON for the affected engine, performs essentially the same function as the No. 1 start control relay. With the essential bus energized, engine master switch on, and throttle at idle, the ignition control relay is energized through the ON position of the "AIRSTART" switch, connecting essential bus power to the ignition unit.

5. "AIRSTART" Switch

The "AIRSTART" switch is used to supply 28-volt dc power from the essential bus to energize the ignition control relay during an airstart. The switch is mounted on the center pedestal. A 55-ohm resistor is installed in the airstart circuit to protect the coil of the ignition control relay by dropping the 28-volt dc power down to 12 volts. This protects the relay coil in the event that the air start switch is left on for prolonged periods.

6. Ignition Selector Switch

The ignition selector switch, mounted on the center pedestal, has two positions labeled CONTINUOUS and NORMAL. With the switch at NORMAL, both continuous and normal ignition circuits are de-energized unless the "START" switch is depressed or the stall warning system is activated. With the switch at CONTINUOUS, the ignition system will operate and the red ignition indicator light will come on anytime power is available to the system and the throttle lever i. at idle or above.

7. Engine Ignition Unit

A dual output, high intensity capacitor discharge ignition unit is installed on the compressor case of each engine. The ignition unit operates on 14 to 30 volt dc input to provide a 3,000 volt output in pulses at a minimum rate of 2 per second to the two igniter plugs. The unit has a duty cycle of 2 minutes on, 3 minutes off, 2 minutes on, 23 minutes off, or 5 minutes on, 25 minutes off.

8. Engine Spark-igniter Plugs

Two engine spark-igniter plugs are installed in each engine at the 8 and 12 o'clock positions in the annular combustion chamber. The igniter plugs are self-ionizing, shunted gap type plugs designed for relatively low voltage. Ionization will occur in these plugs at about 800 volts, creating a conducting path for the ignition unit output, which arcs across the gap.

9. Ignition Control Relays

On airplanes NA-380-1 and subsequent, the ignition control relays, one for each engine, are mounted on shelf 125 of the electrical equipment rack at station 150. On airplanes NA-370-1 and -7, the ignition control relays are mounted in the electrical power distribution box at station 333. Each relay controls application of essential bus power to the respective ignition

control unit except when continuous ignition is selected. The relays are energized when the throttle is advanced and the "AIRSTART" is on or the engine start button is depressed.

10. Throttle-actuated Fuel Boost and Ignition Switches

The throttle-actuated fuel boost and ignition switches are mounted inside the center pedestal and actuated by the throttle levers. Each switch controls the fuel boost pump and the ignition unit for the respective engine.

ENGINE IGNITION SYSTEM - TROUBLE SHOOTING

1. Trouble Shooting Engine Ignition System

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. NO SPARK AT IGNITER PLUGS.		
(1) Engine ignition circuit breaker not closed.	Visually check the following circuit breakers: "RH ENGINE - IGN & BOOST" "LH ENGINE - IGN & BOOST"	Close circuit breakers.
(2) Faulty spark-igniter plug.	Replace suspected plug with a serviceable plug and check system.	Replace igniter plug.
(3) Faulty throttle actuated fuel-ignition switch.	Refer to Wiring Diagram Manual and perform a continuity check of the switch.	Replace switch.
(4) Faulty ignition control relay.	Check system with ignition selector switch at CONTINUOUS. If igniters fire, relay is faulty.	Replace relay.
(5) Ignition unit faulty.	Disconnect plug P913 from igniter on engine. Energize ignition system and check for minimum of 19-volts dc at plug.	Replace ignition unit.
B. INDICATOR LIGHT IN START SWITCH INOPERATIVE DURING START.		
(1) Faulty light bulb.		Replace bulb.

ENGINE IGNITION SYSTEM - MAINTENANCE PRACTICES

1. Removing Engine Spark-igniter Plugs

A. The engine spark-igniter plugs are threaded into the engine combustion chamber at the 8 and 12 o'clock positions.

(1) Make sure engine master switches and "AIRSTART" switch are off and ignition selector switch at NORMAL.

(2) Open engine access doors as required.

(3) Disconnect igniter lead from igniter.

(4) Remove igniter plug and washers (shims) from combustion chamber and retain shims for use on installation.

2. Installing Engine Spark-igniter Plugs

A. Before installing igniter plug, check the following:

(1) Check outer electrode for cracks, breakage or looseness. Check weld for cracks. Any cracks are cause for rejection.

(2) Check outer electrode for chafing exceeding 0.030 inch deep on diameter.

(3) Check that center electrode is not loose and that it is not eroded more than 0.040 inch below outer electrode.

(4) Check ceramic insulator for cracks, looseness, or missing material. Any cracks are cause for rejection. Up to 25 percent of original material may be missing without rejection.

B. Check spark-igniter immersion depth, using GE tool Part No. 2C5388 and install igniter as follows: (See figure 74-00-401.)

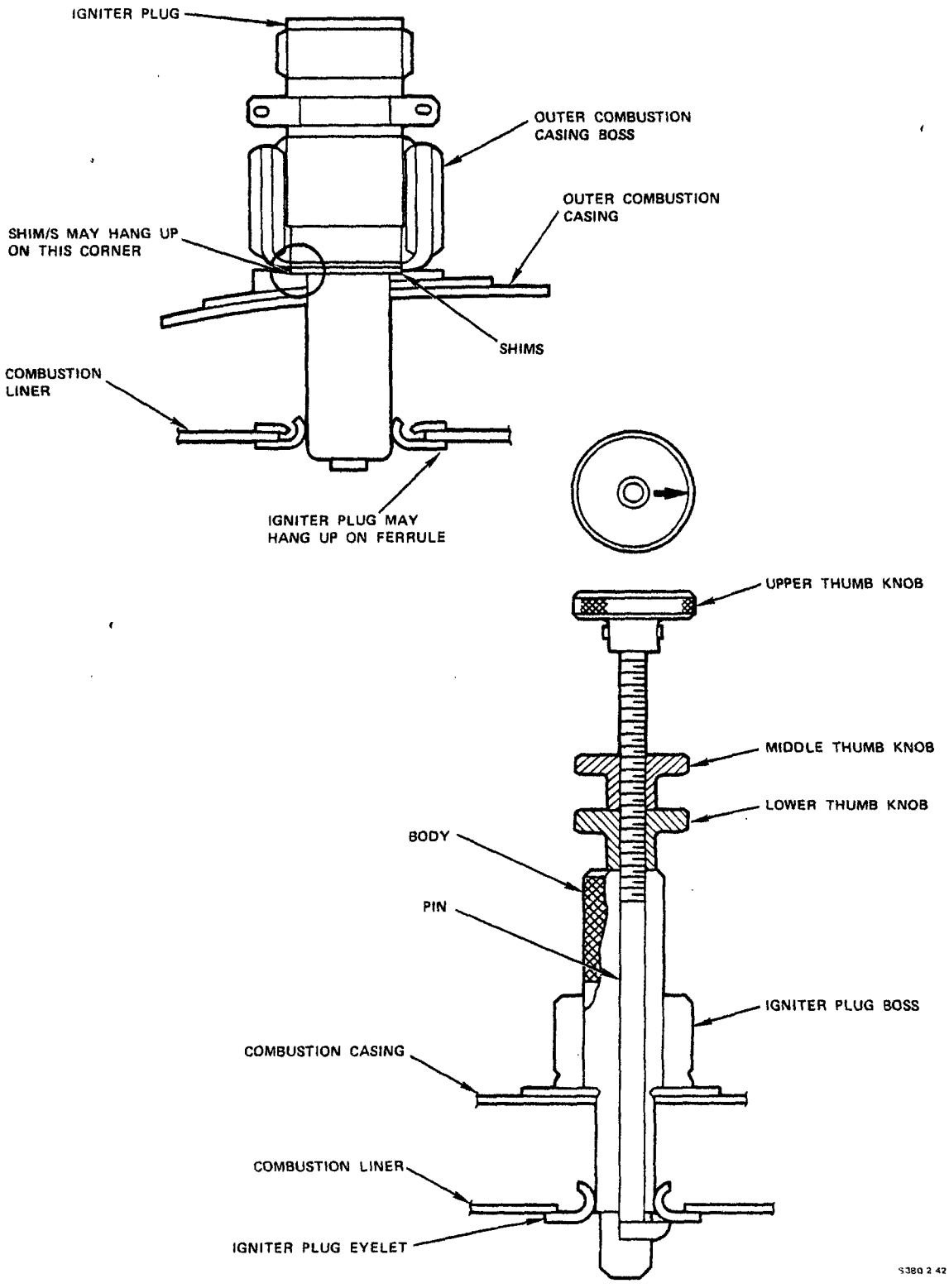
(1) Retract contact foot at lower end of pin on tool by turning upper thumb knob fully clockwise. Note position of index mark on knob.

(2) Insert tool into igniter plug boss, making sure shoulder of body is seated on seating surface inside igniter plug boss.

(3) Rotate upper thumb knob 180 degrees counter clockwise so that toe of contact foot can engage igniter plug eyelet.

(4) Pull pin out until foot engages inside surface of eyelet, then run lower thumb knob down until it contacts body of tool. Do not tighten.

(5) Turn middle thumb knob down to secure lower knob in position, then rotate upper thumb knob 180 degrees clockwise to disengage contact foot. Remove tool from igniter plug boss.



Installing Spark-igniter Plugs
Figure 74-00-401

(6) Measure distance between inside surface of contact foot and seating shoulder on body and record dimension.

(7) Measure length from plug tip to seating shoulder on plug and record dimension.

NOTE: Measure from end of outer electrode not inner electrode.

(8) Subtract dimension obtained in step (6) from dimension obtained in step (7). Select washers to reduce the difference to 0.010-0.015 inch. (Dimension from plug tip to washer face to be 0.010-0.015 inch greater than dimension in step (6).)

(9) Lubricate threads of igniter plug with Milk of Magnesia (non-flavored type) making sure that none is applied to center electrode or insulator.

CAUTION: AVOID UNDUE FORCE WHEN THREADING IGNITER PLUG INTO BOSS TO AVOID DAMAGE TO COMBUSTION CHAMBER LINER. IF IGNITER PLUG DOES NOT THREAD EASILY INTO BOSS, CHECK FOR SHIM INTERFERENCE OR INTERFERENCE WITH COMBUSTION LINER FERRULE.

(10) Torque igniter plug to 225-275 inch-pounds.

(11) Connect igniter lead and torque to 125-150 inch-pounds.

(12) Close engine access doors.

3. Removing Engine Ignition Unit

A. Remove upper engine access door.

B. Make sure all switches and electrical power are off.

C. Disconnect input electrical plug and igniter leads from ignition unit.

D. Remove four bolts and nuts, with isolators and ferrules, securing unit to mounting brackets and remove unit.

4. Installing Engine Ignition Unit

A. Position ignition unit with isolators and ferrules on mounting brackets.

B. Secure unit to brackets with bolts and nuts. Torque nuts to 8-10 inch-pounds.

C. Connect igniter plug leads to unit and torque to 125-150 inch-pounds.

D. Connect input lead electrical connector to unit.

E. Perform a test of ignition system.

F. Close engine access doors.

1. Testing Engine Ignition System

WARNING: DO NOT CHECK IGNITION SYSTEM AFTER AN ATTEMPTED START OR WHENEVER EXCESSIVE FUEL OR FUEL VAPOR IS IN ENGINE OR TAILPIPE.

A. To test engine ignition system, proceed as follows:

(1) To prevent operation of fuel boost pumps during the following test, pull "BOOST PUMP - LH FUEL" and "BOOST PUMP - RH FUEL" circuit breakers.

(2) Push in "LH ENGINE - IGN & BOOST" and "RH ENGINE - IGN & BOOST" circuit breakers.

(3) If battery power is used, position "ELECTRIC MASTER" switch and battery switches at ON.

(4) To prevent opening fuel shutoff valve, pull "LH ENGINE - MASTER" circuit breaker.

(5) Place "LH ENGINE - ENGINE MASTER" switch at ON and "AIRSTART" switch at LH ON.

(6) Advance left engine throttle from CLOSED to IDLE. If the spark-igniter plugs are firing, the firing can be heard.

(7) Move left engine throttle to CLOSED. Spark igniters should stop firing.

(8) Place "LH ENGINE - ENGINE MASTER" switch at OFF.

(9) Pull "RH ENGINE - MASTER" circuit breaker and place "RH ENGINE - ENGINE MASTER" switch at ON.

(10) Place "AIRSTART" switch at RIGHT HAND ON.

(11) Advance right engine throttle from CLOSED to IDLE. If spark-igniter plugs are firing, the firing can be heard.

(12) Return right throttle to CLOSED.

(13) Place all switches at OFF and reset circuit breakers.

2. Testing Engine Ignition Unit

(1) Remove spark-igniter plugs. (Refer to "Removing Spark-Igniter Plugs".)

(2) Connect igniter leads to plugs and ground shell of both plugs securely to engine.

(3) Station a man in vicinity of engine to watch spark-igniter plugs.

WARNING: STAY CLEAR OF SPARK-IGNITER PLUGS DURING TESTS, AS THE DISCHARGE CAN BE LETHAL.

(4) Pull "BOOST PUMP - LH FUEL" and "BOOST PUMP - RH FUEL" circuit breakers.

(5) Position "ELECTRIC MASTER" switch and battery switches to ON.

(6) Advance throttle on engine being checked to idle.

(7) Depress engine start button. Strong blue rapid firing arcs should be visible. If either igniter exhibits a weak spark, igniter should be replaced. If both are weak, the ignition unit or airplane electrical system may be malfunctioning.

(8) Pull out start button. Both spark-igniters should stop firing.

(9) Position "AIRSTART" switch at ON. Both spark-igniters should fire.

(10) Position "AIRSTART" switch at OFF. Both igniters should stop firing.

(11) Position throttle lever at CLOSED and turn all switches off.

(12) Remove electrical power and push in all circuit breakers.

(13) Install spark-igniter plugs and install engine access doors.

3. Inspect Igniter Plugs

A. The ignition system limits are as follows:

2 minutes ON followed by 3 minutes OFF
then
2 minutes ON followed by 23 minutes OFF
or alternately
5 minutes ON followed by 25 minutes OFF.

If the above limits are exceeded, inspect the igniter plugs as follows:

(1) Remove igniter plugs. (Refer to Removing Engine Spark-igniter Plugs.)

(2) Check outer electrode for cracks, breakage or looseness. Check weld for cracks. Any cracks are cause for rejection.

(3) Check outer electrode for chafing exceeding 0.030-inch deep on diameter.

(4) Check that center electrode is not loose and that it is not eroded more than 0.040-inch below outer electrode.

(5) Check ceramic insulator for cracks, looseness, or missing material. Any cracks are cause for rejection. Up to 25 percent of original material may be missing without rejection.

(6) Install serviceable igniter plugs. (Refer to Installing Engine Spark-igniter Plugs.)

CHAPTER

75

AIR

NOTICE

The following information applies to NA 265-70 airplanes only.

CHAPTER 75

AIR

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AIR

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*The asterisk indicates pages revised, added, or deleted by the current revision.

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AIR GENERAL - DESCRIPTION

1. General

Air is directed into the starter-generator and the engine compartment forward of the fire seal for cooling purposes. Hot air is extracted from the engine compressor section through an air extraction manifold for use in the engine anti-icing system and the airplane environmental system. The hot air is routed forward to the engine air inlet duct, where it is distributed to the engine inlet guide vanes, intake duct splitter vanes, and the leading edge of the engine air intake duct assembly. (Refer to 75-10-0.) Hot air is also routed aft from the air extraction manifold through the engine bleed-air shutoff valves in the right- and left-hand pylons and a emergency pressurization valve in the right-hand pylon for use in the airplane environmental system. (Refer to Chapter 21.)

ENGINE ANTI-ICING SYSTEM

DESCRIPTION AND OPERATION

GENERAL

The engine anti-icing system utilizes compressor discharge air to heat the engine inlet guide vanes, engine inlet nose fairing and splitter vanes, and engine nose cone, to remove and prevent ice formation. The anti-icing system is controlled by the ANTI-ICE — ENGINE INLETS switch on the anti-ice control panel. The ANTI-ICE — ENGINE INLETS switch is normally positioned at OFF. Two caution lights are included in the system: RH ENG DE-ICE FAIL and LH ENG DE-ICE FAIL. The RH ENG DE-ICE FAIL and LH ENG DE-ICE FAIL lights come on when the engine inlet anti-icing valve fails to open when an ice condition exists and the ANTI-ICE — ENGINE INLETS switch is at the ON position. DC power for operation of the engine inlet anti-icing system is supplied by the essential bus.

The operational sequence of the engine anti-icing system is similar for both engines. Only the left engine inlet anti-ice system is described.

Placing the ANTI-ICE — ENGINE INLETS switch at ON energizes both the left and the right engine inlet anti-ice valve solenoids, the left and right engine inlet anti-ice fail indicator control relays, the left and right engine inlet guide vane anti-ice valve indicator relays, and the left and right engine inlet anti-ice valve indicator relays. (See Figure 75-10-2.) The engine inlet anti-ice valves open, allowing heated discharge air from the compressor to flow through the duct into the engine inlet nose fairing air chamber to melt the ice formation on the air inlet nose fairing. This air is also discharged into the splitter vanes to melt the ice formations on the vanes. Placing the ANTI-ICE — ENGINE INLETS switch at ON removes power from the coils of the left and/or right engine guide vane anti-ice valves. When power is removed from the valve, it returns to the normally open position, allowing the heated air to flow and deice the engine air inlet guide vanes.

The left and right engine inlet anti-ice fail indicator control relays are also energized by the switch. This provides a path for power through both engine inlet guide vane anti-ice valves, if the engine inlet anti-ice valves should fail to open. This power is routed to illuminate the LH ENG DE-ICE FAIL or the RH ENG DE-ICE FAIL lights in the cockpit. (See Figure 75-10-1.)

ENGINE INLET ANTI-ICE VALVE

An engine inlet anti-ice valve is installed on the upper right side of the compressor section of each engine. (See Figure 75-10-1.) This valve is a pressure-limiting type valve, used to control the heated airflow from the compressor to the engine inlet air duct and splitter vanes. The valve consists of a two-port body assembly, a regulator assembly, a spring-loaded poppet valve, an integral pilot solenoid, and a position indicator switch. The valve assembly is normally closed and is designed to open when the solenoid is energized and to operate automatically in limiting the airflow to the engine inlet duct fairing. The position indicator switch within the valve furnishes a ground for the engine inlet anti-ice valve indicator relays when the engine inlet anti-ice valves are closed.

ENGINE GUIDE VANE ANTI-ICE VALVE

An engine guide vane anti-ice valve is mounted in the compressor air discharge line on the left side of the compressor section of each engine. (See Figure 75-10-1.) The valve is used to control the heated airflow from the compressor to the engine guide vanes and engine inlet nose cone. Airflow from the compressor opens or closes the valve when the engine is operating, depending on the position of the solenoid within the valve body, which is controlled by the position of the ANTI-ICE — ENGINE INLETS switch. With this switch at OFF and the engine operating, the valve solenoid is energized and the valve closes. Positioning the switch at ON de-energizes the solenoid, and the valve opens.

The position indicator switch within the valve furnishes a ground for the engine inlet guide vane anti-ice valve indicator relays when the engine inlet guide vane anti-ice valves are open. If the valves fail to position open or closed in relation to the ANTI-ICE — ENGINE INLETS switch position, a circuit is completed to the respective engine anti-ice fail light.

ENGINE INLET ANTI-ICE FAIL INDICATOR CONTROL RELAYS

Two engine inlet anti-ice fail indicator control relays (K693 and K695) are provided in the engine inlet anti-ice system, one for the left engine and one for the right engine. (See Figure 75-10-2.) These relays are mounted in the electrical power distribution box at FS 333. When an icing condition exists, indicated by the illumination of the ICING CONDITION caution light, the ANTI-ICE — ENGINE INLETS switch should be moved to the ON position, energizing this relay. Power is then routed to the coils of the engine inlet guide vane anti-ice valve indicator relays (K696 and K697) and the engine inlet anti-ice valve indicator relays (K692 and K694).

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DESCRIPTION AND OPERATION (Continued)

ENGINE INLET GUIDE VANE ANTI-ICE VALVE INDICATOR RELAYS

Two engine inlet guide vane anti-ice valve indicator relays (K696 and K697) are provided in the engine inlet anti-ice system, one for each engine. (See Figure 75-10-2.) These relays are mounted in the electrical power distribution box at FS 333. This relay works in conjunction with the engine inlet anti-ice valve indicator relay (K692 or K694) to complete an electrical circuit to the LH ENG DE-ICE FAIL or the RH ENG DE-ICE FAIL caution light in the cockpit. The relay is energized when the ANTI-ICE — ENGINE INLETS switch is at OFF or ON and the engine inlet guide vane anti-ice valve is open.

ENGINE INLET ANTI-ICE VALVE INDICATOR RELAYS

Two engine inlet anti-ice valve indicator relays (K692 and K694) are provided in the engine inlet anti-ice system, one for each engine. (See Figure 75-10-2.) These relays are mounted in the electrical power distribution box at FS 333. This relay works in conjunction with the engine inlet guide vane anti-ice valve relay (K696 or K697) to complete an electrical circuit to the LH ENG DE-ICE FAIL or the RH ENG DE-ICE FAIL caution light in the cockpit. This relay is energized when the engine inlet anti-ice valve is closed and the ANTI-ICE —ENGINE INLETS switch is at either OFF or ON.

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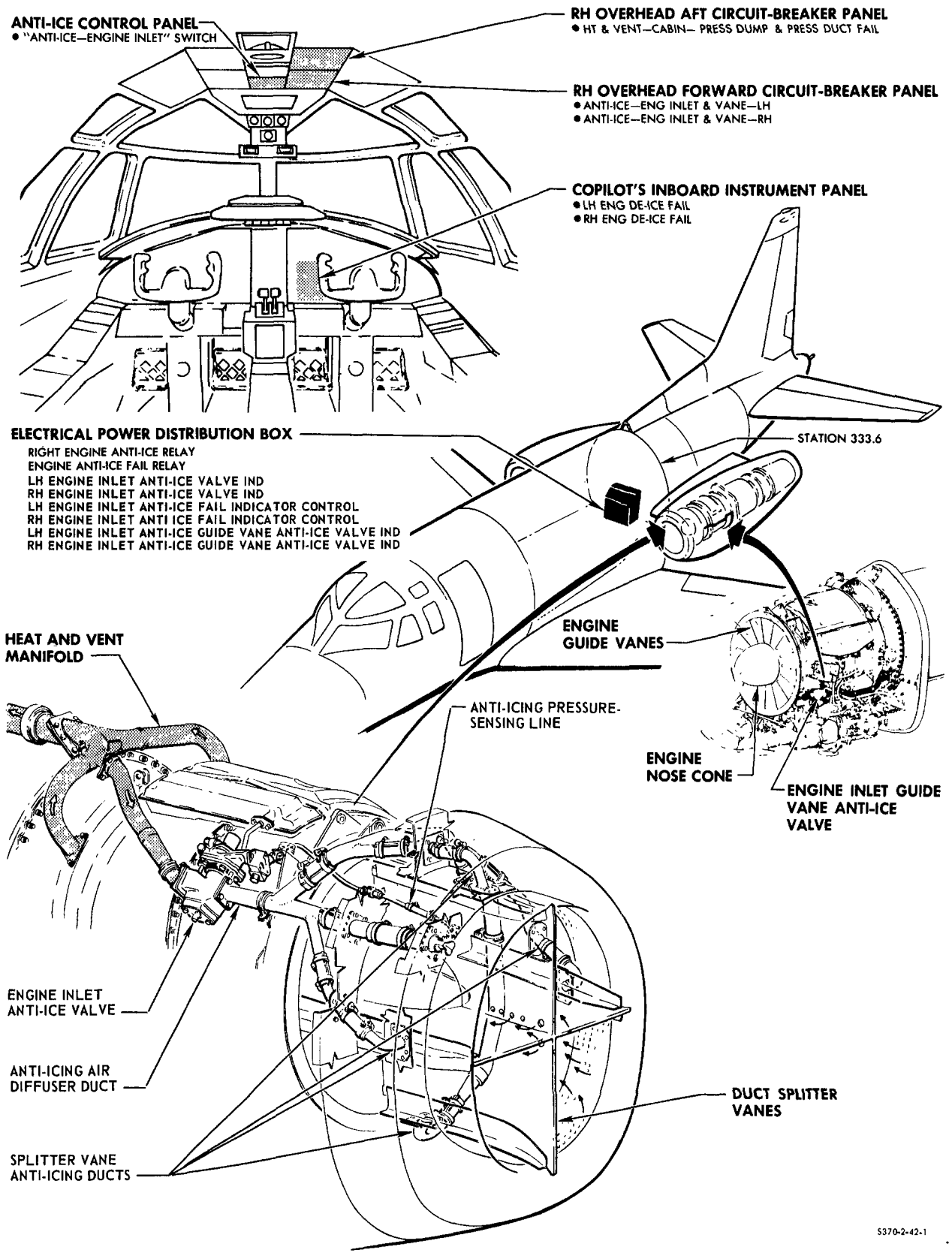
ANTI-ICE CONTROL PANEL
• "ANTI-ICE—ENGINE INLET" SWITCH
RH OVERHEAD AFT CIRCUIT-BREAKER PANEL
• HT & VENT—CABIN— PRESS DUMP & PRESS DUCT FAIL
RH OVERHEAD FORWARD CIRCUIT-BREAKER PANEL
• ANTI-ICE—ENG INLET & VANE—LH
• ANTI-ICE—ENG INLET & VANE—RH
COPILOT'S INBOARD INSTRUMENT PANEL
• LH ENG DE-ICE FAIL
• RH ENG DE-ICE FAIL
ELECTRICAL POWER DISTRIBUTION BOX
RIGHT ENGINE ANTI-ICE RELAY
ENGINE ANTI-ICE FAIL RELAY
LH ENGINE INLET ANTI-ICE VALVE IND
RH ENGINE INLET ANTI-ICE VALVE IND
LH ENGINE INLET ANTI-ICE FAIL INDICATOR CONTROL
RH ENGINE INLET ANTI ICE FAIL INDICATOR CONTROL
LH ENGINE INLET ANTI-ICE GUIDE VANE ANTI-ICE VALVE IND
RH ENGINE INLET ANTI-ICE GUIDE VANE ANTI-ICE VALVE IND
STATION 333.6
HEAT AND VENT MANIFOLD
ENGINE GUIDE VANES
ANTI-ICING PRESSURE-SENSING LINE
ENGINE NOSE CONE
ENGINE INLET GUIDE VANE ANTI-ICE VALVE
ENGINE INLET ANTI-ICE VALVE
ANTI-ICING AIR DIFFUSER DUCT
DUCT SPLITTER VANES
SPLITTER VANE ANTI-ICING DUCTS
S370-2-42-1

Figure 75-10-1 — Engine Inlet Anti-icing System

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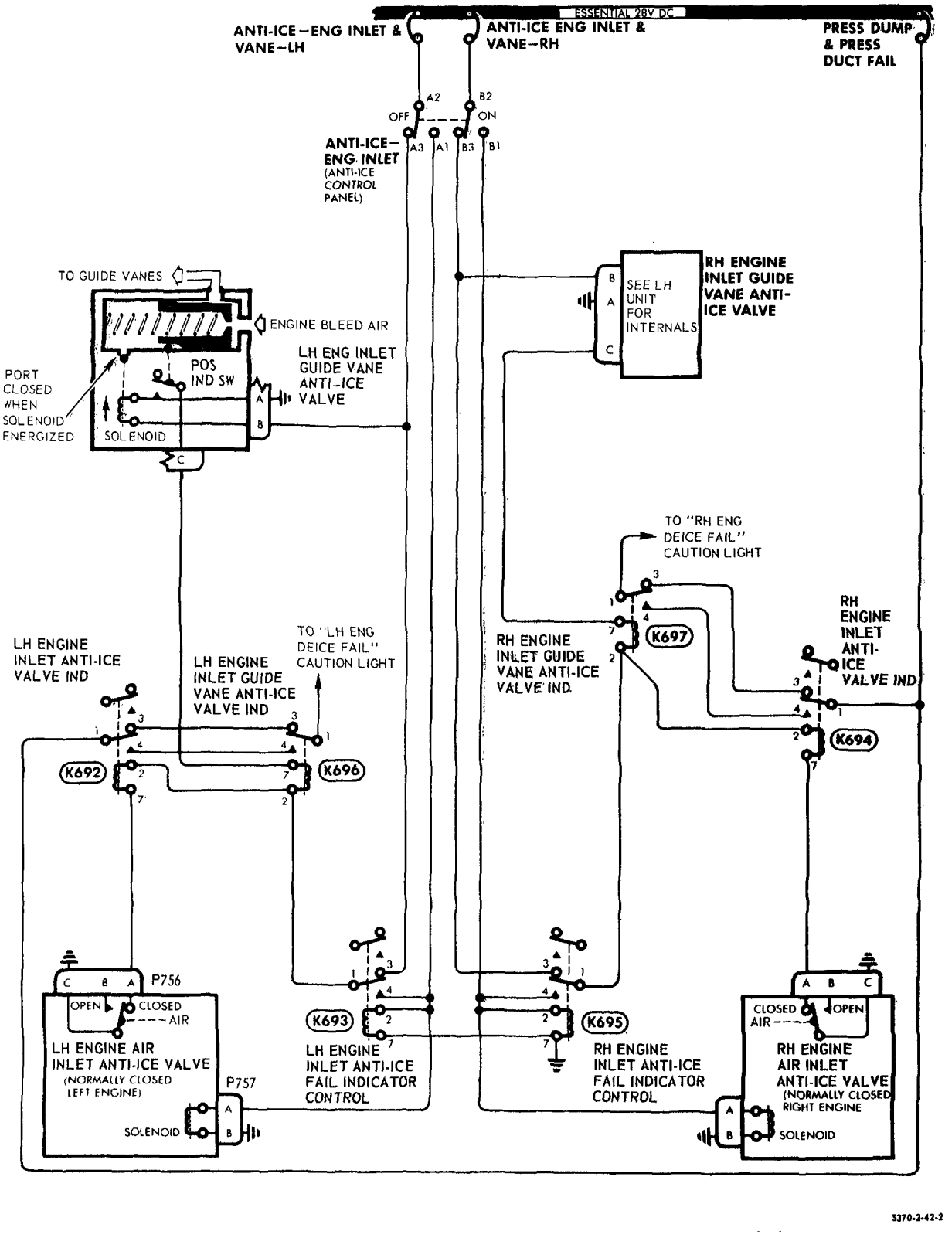

Figure 75-10-2 — Engine Inlet Anti-icing System Electrical Schematic

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ENGINE ANTI-ICING SYSTEM - TROUBLE SHOOTING

1. Trouble Shooting Engine Anti-icing System

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. ENGINE ANTI-ICE FAIL LIGHTS REMAIN ON DURING ENGINE OPERATION.		
(1) Engine rpm too low.		Increase engine operating rpm.
(2) Faulty "PRESS DUMP & DUCT FAIL VAC PUMP" or "ICE CONTROL INLET & VANE - RH" circuit breaker.	Visually check circuit breakers.	Replace defective circuit breakers.
(3) Faulty "ANTI-ICE - ENGINE INLETS" switch.	Place "ANTI-ICE - ENGINE INLETS" switch at ON. With multimeter, check for 28-volt dc at terminal "B3."	If power is not available, replace switch.
(4) Faulty engine inlet anti-ice fail indicator control relay.	Place "ANTI-ICE - ENGINE INLETS" switch at ON. With multimeter, check for 28-volt dc at terminals "7" and "2." If power is available, check for power at terminals "1" and "4" of inlet anti-ice fail indicator control relay.	Replace relay.
(5) Faulty electrical circuit to engine inlet anti-ice air valve.	Place "ANTI-ICE - ENGINE INLETS" switch at ON. With multimeter, check for 28-volt dc at pin "Z" of engine disconnect plug P639 (left engine) and P640 (right engine).	If power is not available, repair faulty circuit. (Refer to Wiring Diagram Manual.)

ENGINE ANTI-ICING SYSTEM - MAINTENANCE PRACTICES

1. Removing Engine Guide Vane Anti-icing Valve

A. The guide vane anti-icing valve is mounted in the anti-icing air line on the lower forward side of the engine compressor case. To remove the guide vane anti-icing valve from the right-hand engine, the engine must be removed from the engine pod. The valve assembly is accessible on the left engine when the lower engine access door is open. To remove valve, proceed as follows:

(1) Disconnect electrical plug from anti-icing valve solenoid.

(2) Remove bolts and locknuts securing duct to inlet and diffuser case. Remove duct and valve from engine.

(3) Remove bolts securing valve to elbow and duct.

2. Installing Engine Guide Vane Anti-icing Valve

A. To install engine guide vane anti-icing valve, proceed as follows:

(1) Place gasket and spacer on rear flange of anti-icing valve and secure valve to duct with bolts. Torque bolts and safety with lockwire.

(2) Insert guide into valve. Secure guide and elbow on valve with bolts. Torque bolts and safety with lockwire.

(3) Place gasket on rear flange of anti-icing duct. Position duct assembly on engine pads at inlet and diffuser cases.

(4) Secure duct assembly with bolts and locknuts.

(5) Connect electrical plug to anti-icing valve solenoid. Safety plug with lockwire.

3. Removing Engine Inlet Anti-ice Valve

A. The engine inlet anti-icing valve is installed on the upper right side of the compressor case. To remove valve, proceed as follows:

NOTE: To remove the anti-ice valve from the left engine, the left engine must be removed. The valve assembly is accessible on the right engine, with the engine installed.

(1) Disconnect two electrical plugs from valve assembly.

(2) Remove eight bolts securing valve to duct assemblies.

(3) Remove valve and gaskets.

4. Installing Engine Inlet Anti-ice Valve

A. To install engine inlet anti-ice valve, proceed as follows:

(1) Position valve between ducts with arrow on valve pointing forward.

(2) Insert gasket between aft valve flange and duct assembly. Install and tighten four bolts.

(3) Insert gasket between forward valve flange and duct assembly. Install and tighten four bolts.

(4) Connect two electrical plugs to valve assembly. Safety plugs with lockwire.

1. Checking Engine Inlet Anti-icing System Without Engine Operating

A. To check engine inlet anti-icing system without engine operating, proceed as follows:

(1) Make sure "RH ENGINE - ENGINE MASTER" and "LH ENGINE - ENGINE MASTER" switches are at OFF.

(2) Connect a 28-volt dc power source to external utility power receptacle.

NOTE: If airplane battery power is used, place "ELECTRIC MASTER" and "BATTERY" switches at ON.

(3) Close "ICE CONTROL - ENG INLET & VANE - LH," "ICE CONTROL - ENG INLET & VANE - RH," and "PRESS DUMP & DUCT FAIL VAC PUMP" circuit breakers.

(4) Place "ANTI-ICE - ENGINE INLETS" switche at ON. The "RH ENG DE-ICE FAIL" and "LH ENG DE-ICE FAIL" caution lights may be seen to momentarily blink on, then go out (this is normal); then they should remain out.

(5) While moving "ANTI-ICE - ENGINE INLETS" switch from OFF to ON, listen for an audible click at engine inlet guide vane anti-ice valve (left side of engine) and engine inlet anti-ice valve (upper right side of engine).

(6) Move "ANTI-ICE - ENGINE INLETS" switch to OFF. Disconnect external power source.

2. Checking Engine Inlet Anti-icing System With Engine Operating

A. To check engine inlet anti-icing system with engine operating, proceed as follows:

(1) Start engine. (Refer to Chapter 71.)

NOTE: The "LH ENG DE-ICE FAIL" and "RH ENG DE-ICE FAIL" caution lights will come on at low power settings, since air pressure on the downstream side of the engine anti-ice valve is insufficient to actuate the valve position indicator switch.

(2) Place "ANTI-ICE - ENGINE INLETS" anti-ice switch at ON and advance throttle lever above IDLE. The "LH ENG DE-ICE FAIL" and "RH ENG DE-ICE FAIL" caution lights should go out. This indicates that the left and right engine inlet anti-ice valves and inlet guide vane anti-ice valves are open.

CAUTION: DO NOT OPERATE ENGINE ANTI-ICING SYSTEM ON THE GROUND FOR MORE THAN 2 MINUTES CONTINUOUSLY. OPERATION OF THE SYSTEM FOR AN EXTENDED PERIOD OF TIME CAUSES OVERHEATING OF THE STRUCTURE, RESULTING IN POSSIBLE DAMAGE.

(3) Move "ANTI-ICE - ENGINE INLETS" switch to OFF.

(4) Shut down engine.

ACCESSORY COOLING

1. Starter-Generator Cooling

The starter-generator is cooled by direct air picked up from the starter-generator cooling duct on the lower leading edge of the engine air intake duct. Cooling air is ducted directly into the starter-generator through the front casting, circulates through the starter-generator, and exits downward through the rear casting and overboard through the engine cowling.

2. Engine Compartment Cooling

The engine compartment, forward of the fire seal, is cooled by air taken in by a small air duct on the lower leading edge of the engine air intake duct. The air blast is directed against a baffle fastened to the end of the intake duct. Air is then diffused by the baffle throughout the forward engine compartment.

NOTICE

The following information applies to NA 265-80 airplanes only.

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AIR

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CHAPTER 75

AIR

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AIR GENERAL

DESCRIPTION

GENERAL

On S/N 370-1 and -7 and 380-1 and Subs, heated anti-icing air is ducted forward from the compressor bleed port through an electrically controlled anti-icing valve secured to the engine mainframe at the 10 o'clock position. Anti-icing air is directed through the anti-icing valve into an anti-icing manifold brazed circumferentially around the outer casing of the engine over the inlet guide vanes. A portion of the anti-icing air in the manifold enters into the hollow inlet guide vanes through slots cut in the support bosses of the inlet guide vanes, around the baffles, then out into the airstream through holes in the outboard ends of the inlet guide vanes. The remainder of the anti-icing air in the manifold passes through the support struts then into the inner hub. Part of the air in the inner hub is directed forward into the bulletnose which has an inner and outer skin. Air flows forward through a hole in the inner skin, then aft between the inner and outer skins, and out through holes in the aft end of the bulletnose outer skin. The remainder of the anti-icing air in the inner hub flows through internal passages in the bearing housing into the double skinned inlet guide vane shroud. This air provides anti-icing of the shroud ring before being exhausted through holes in the aft end of the shroud ring.

The front and rear frames of the aft fan are cooled by air bled from two pads located at approximately the 11 o'clock and 1 o'clock positions on the compressor upper case. Fifth stage bleed air flows through two tubes aft to the aft fan section. The right tube connects to the front frame and the left tube is routed under the engine, around the right side of the aft fan and into the rear frame of the aft fan housing. This air enters into the inner boss structure of the front and rear frames by flowing through the number 2 strut of the front and rear frames.

Eighth stage air which passes through the leakage air valves is used to heat the aft fan front frame. Air flows through two tubes secured to pads at the 10 o'clock and 2 o'clock positions on the compressor upper case, aft to the forward manifold on the front frame. The air flows completely around the manifold except for that portion of the manifold between number 2 and 3 struts. From the manifold the air flows inward through all of the struts, except number 2, cooling the portion of the struts between the mid-cone and the inner manifold.

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ENGINE ANTI-ICING SYSTEM

DESCRIPTION AND OPERATION

GENERAL

The engine anti-icing system utilizes compressor discharge air to heat the engine inlet guide vanes, bulletnose, and the inlet guide vane shroud ring. The engine anti-icing system is controlled by the ENGINE ANTI-ICE switch located on the anti-ice control panel. The ENGINE ANTI-ICE switch is normally positioned at OFF. With the switch in the OFF position, 28 VDC power from the essential bus is applied to the engine anti-ice solenoid valves energizing them to the closed position. At the same time power is also routed to energize the LH and RH air intake anti-ice fail relays (K1060 and K1064) removing power to the engine anti-ice fail lights in the caution and warning panel.

The operaional sequence of the engine anti-ice system is similar for both engines. Only the left engine inlet anti-icing system is described.

Placing the ENGINE ANIT-ICE switch at ON removes power from the engine anti-icing solenoid valve and opens the intake anti-ice valve. This allows heated compressor bleed air to flow through the anti-icing line between the compressor bleed port and the anti-icing valve. Power is also routed from the ENGINE ANTI—ICE switch through the contacts of LH engine anti-ice solenoid fail relay (K1063) to the coils of LH, time delay relay (TM503). Power is routed through the pressure switch to energize the coil of the engine anti-ice solenoid fail relay (K0163) removing the power through the contacts of relay K1063 to the coil of the time delay relay TM503. Power is also provided to the open portion of the LH intake anti-ice valve, which in turn directs power to energize the coils of the LH air intake anti-ice fail relay (K1064). With relay K1064 energized power is no longer available to illuminate the LH engine anti-ice fail light in the caution and warning panel. If the thermal switch does not close within 30 seconds after the ENGINE ANTI-ICE switch is turned ON, the time delay relay will energize and direct power to illuminate the engine anti-ice fail light on the caution and warning panel.

On S/N 380-3 and subs; when the heated air flowing through the anti-icing line reaches a pressure of 10-12 psi, the pressure switch (S1027), mounted on the compressor bleed air line will close allowing power from the intake anti-ice valve to actuate the engine anti-ice solenoid fail relay (K1063). When relay (K1063) is energized, power is removed from the anti-ice fail lights on the caution and warning panel in the cockpit. If the pressure switch fails to close, relay K1063 remains de-energized, allowing power to be routed to the anti-ice fail light.

ENGINE ANTI-ICE SOLENOID VALVE

The electrically energized engine anti-ice solenoid valve controls the off-on flow of compressor bleed air to the anti-icing control valve mounted at the 10 o'clock position on the left side of each engine. When the ENGINE ANTI-ICE switch is positioned at OFF, the engine anti-ice solenoid valve is energized to the closed position thereby preventing bleed air from passing through the anti-icing control valve. In the event of an electrical power failure, when electrical power is removed from the solenoid valve, it will automatically position itself in the open position, assuring that anti-icing bleed air will be available.

ENGINE ANTI-ICE THERMAL SWITCH — S/N 380-1 and -2

Deleted

ENGINE ANTI-ICE PRESSURE SWITCH — S/N 380-3 and SUBS

The engine anti-ice pressure switch (S1027) is mounted on the anti-icing supply line downstream of the engine anti-icing solenoid valve. The pressure switch is activated when the anti-icing valve is open and air pressure within the anti-icing line reaches 10-12 psi. When the pressure switch is activated it provides power to actuate the engine anti-ice solenoid fail relay (K1063) whch in turn removes power from the engine anti-ice fail lights on the caution and warning panel in the cockpit.

TIME DELAY RELAYS — S/N 380-1, and -2

Deleted

TIME DELAY RELAY — S/N 380-1, -2, -42 AND SUBS

Time delay relays (TM503 and TM504) are mounted in the electrical equipment rack on the right side of the fuselage at station 158. These relays are set to provide 3 (±1/2) second delay to prevent a momentary illumination of the ANTI-ICE FAIL caution light that will occur when the ENGINE ANTI-ICE switch is moved to OFF or when the power levers are advanced. This is due to the momentary pressure drop across the pneumatic pressure switches.

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DESCRIPTION AND OPERATION (Continued)

ENGINE ANTI-ICE SOLENOID FAIL RELAYS — S/N 380-1 and -2

The right hand and left hand engine anti-ice solenoid fail relays (K1062 and K1063) are mounted in the electrical equipment rack on the right hand side of the fuselage at station 150. These relays control the operation of the time delay relays (TM503 and TM504). When the thermal switch on the compressor bleed air line closes, power is routed through the thermal switch to energize the engine anti-ice solenoid fail relays which in turn removes power from the time delay relay coils.

AIR INTAKE ANTI-ICE FAIL RELAYS — S/N 380-1 and -2

The right hand and left hand air intake anti-ice fail relays (K1060 and K1064) are mounted in the electrical equipment rack on the right hand side of the fuselage at station 150. The relays are energized through the ENGINE ANTI-ICE switch and the limit switches in the intake anti-icing valve. When the relays are energized, power is removed from the anti-ice fail lights on the caution and warning panel.

AIR INTAKE ANTI-ICE FAIL RELAYS — S/N 380-3 AND SUBS.

The right hand and left hand engine anti-ice solenoid fail relays (K1062 and K1063) are mounted in the electrical equipment rack on the right hand side of the fuselage at station 150. These relays are energized by the pressure switch. When energized, electrical power is removed from the anti-ice fail light.

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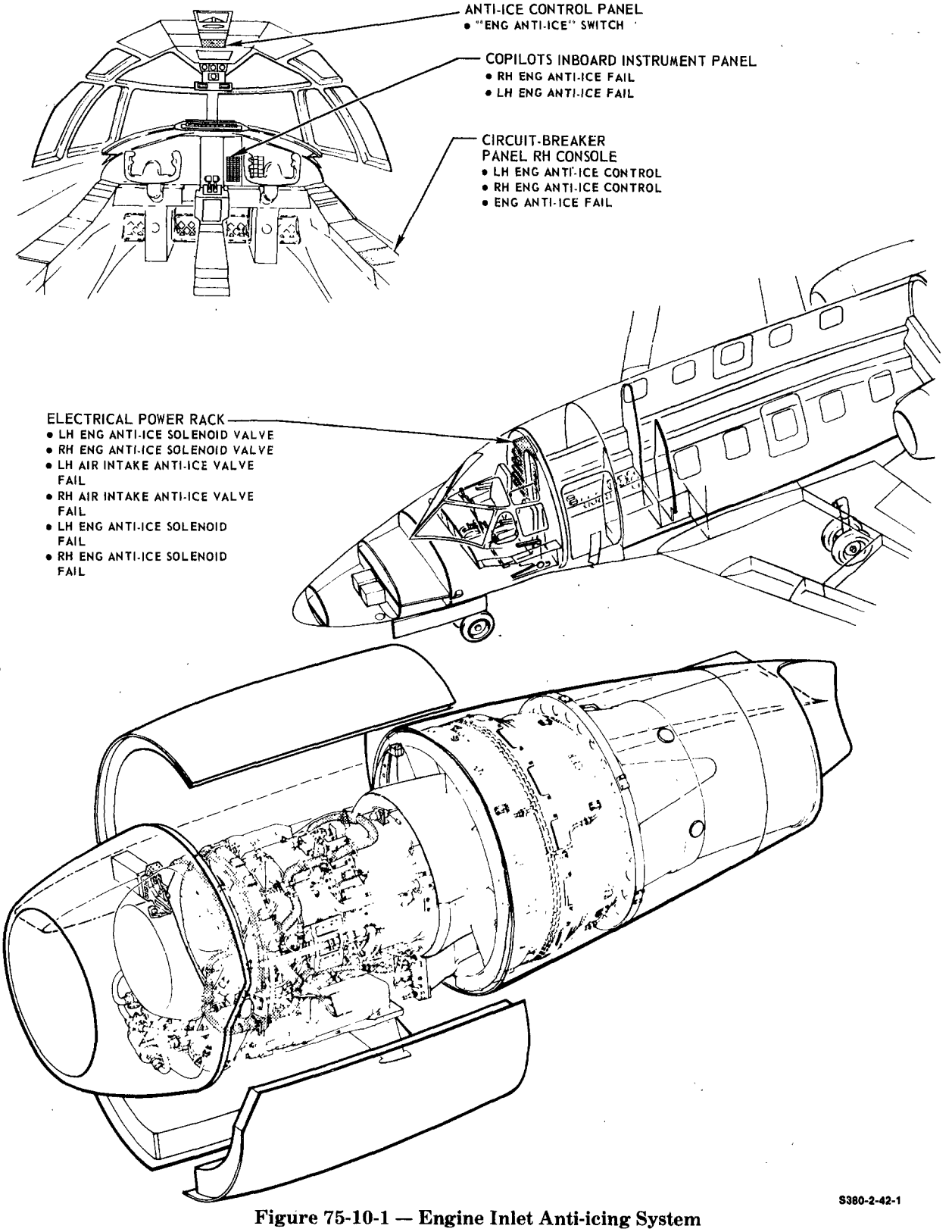
ANTI-ICE CONTROL PANEL
• "ENG ANTI-ICE" SWITCH
COPILOTS INBOARD INSTRUMENT PANEL
• RH ENG ANTI-ICE FAIL
• LH ENG ANTI-ICE FAIL
CIRCUIT-BREAKER
PANEL RH CONSOLE
• LH ENG ANTI-ICE CONTROL
• RH ENG ANTI-ICE CONTROL
• ENG ANTI-ICE FAIL
ELECTRICAL POWER RACK
• LH ENG ANTI-ICE SOLENOID VALVE
• RH ENG ANTI-ICE SOLENOID VALVE
• LH AIR INTAKE ANTI-ICE VALVE FAIL
• RH AIR INTAKE ANTI-ICE VALVE FAIL
• LH ENG ANTI-ICE SOLENOID FAIL
• RH ENG ANTI-ICE SOLENOID FAIL
S380-2-42-1

Figure 75-10-1 — Engine Inlet Anti-icing System

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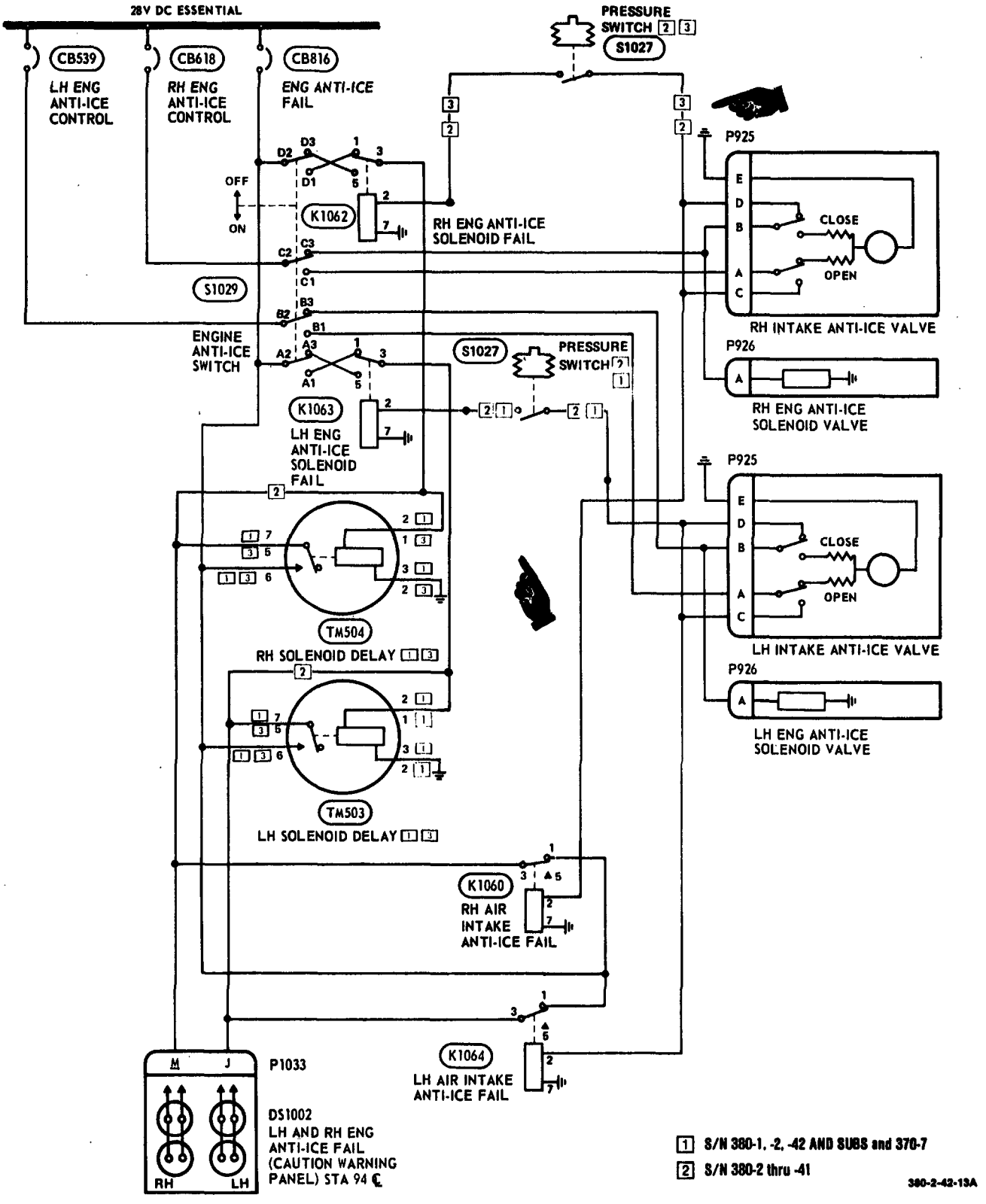

Figure 75-10-2 — Engine Inlet Anti-icing System Electrical Schematic

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6A. Air Intake Anti-ice Fail Relays - NA-380-3 and Later Airplanes

The right hand and left hand engine anti-ice solenoid fail relays (K1062 and K1063) are mounted in the electrical equipment rack on the right hand side of the fuselage at station 150. These relays are energized by the pressure switch. When energized, electrical power is removed from the anti-ice fail light.

ENGINE ANTI-ICING SYSTEM

TROUBLE SHOOTING

TROUBLE SHOOTING ENGINE ANTI-ICING SYSTEM

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
ENGINE ANTI-ICE FAIL LIGHTS REMAIN ON DURING ENDING OPERATION		
1. Engine rpm too low.		Increase engine operating rpm.
2. Faulty ENGINE ANTI-ICE switch (S1029).	With ENGINE ANTI-ICE switch at OFF, check for 28 VDC at Pin B of P925 at intake anti-ice valve.	If power is not available, replace switch.
	Check for 28 VDC at terminal 2 of K1060 and K1064.	If power present, replace K1060 or K1064. If power is not present, check anti-ice valve for operation.
ENGINE ANTI-ICE FAIL LIGHTS COME ON AND REMAIN ON WHEN ANTI-ICE CONTROL SWITCH IS PLACED AT ON		
1. Faulty engine anti-ice solenoid valve.	Have assistant in cockpit actuate ENGINE ANTI-ICE switch while listening for solenoid operation. Solenoid must be de-energized when switch is at ON.	Replace defective anti-ice solenoid valve(s).
2. Faulty switch (S1027).	Perform continuity check of switch.	Replace defective switch(es.)
ENGINE ANTI-ICE FAIL LIGHTS COME ON WHEN ANTI-ICE CONTROL SWITCH IS PLACED AT ON THEN GO OUT WITHIN 30 SECONDS — S/N 380-1 and -2		
1. Faulty time delay relays (TM503 and TM504).	Perform continuity check of time delay relays.	Replace defective time delay relay(s).

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ENGINE ANTI-ICING SYSTEM

MAINTENANCE PRACTICES

ENGINE ANTI-ICE SOLENOID VALVE

Removal:

The engine anti-ice solenoid valve is mounted on the forward left hand side of the engine compressor case at approximately the 10 o'clock position. To remove the engine anti-ice solenoid valve from the right hand engine, the engine upper access door must be removed. On the left hand engine it is only necessary to open the engine upper access door to remove anti-ice solenoid valve. To remove engine anti-ice solenoid valve, proceed as follows:

1. Disconnect electrical plug from solenoid portion of engine anti-ice solenoid valve.
2. Remove 2 bolts securing solenoid clamp to solenoid support bracket.
3. Remove 3 bolts securing the compressor bleed air supply line to attachment pad on aft side of engine anti-icing valve.
4. Remove bolts and nuts (2 each) securing aft side of engine anti-icing valve to machined ring of engine compressor case.
5. Remove bolt securing spacer to machined ring of compressor case.
6. Remove 3 bolts securing engine anti-icing valve to anti-icing manifold.
7. Remove engine anti-icing valve from engine and discard used gasket.

Installation:

1. Place spacer on forward side of machined ring on compressor case and secure in place using bolt and nut previously removed, through center hole in spacer.
2. Place a new gasket on forward attaching pad of engine anti-icing valve and secure valve to anti-icing manifold with 3 bolts. Do not tighten bolts.
3. Install bolts and nuts (2 each) through aft side of engine anti-icing valve, spacer and machined ring on compressor case. Use shims as necessary to eliminate any gaps between engine anti-icing valve attaching lugs and spacer. Do not tighten nuts and bolts.
4. Place a new gasket on aft pad of engine anti-icing valve and connect compressor bleed air supply line to engine air inlet valve. Do not tighten bolts.
5. Install clamp over solenoid and attach to bracket with 2 bolts. Torque bolts to 24-27 in.-lbs and safety with lockwire.
6. Torque bolts securing engine anti-ice valve to anti-ice manifold to 20-25 in.-lbs and safety with lockwire.
7. Torque nuts and bolts through aft side of engine anti-ice valve, spacer, and machined ring on compressor case to 35-40 in.-lbs.
8. Torque nuts and bolts securing compressor bleed air supply line to engine anti-ice valve to 20-25 in.-lbs and safety with lockwire.
9. Connect electrical plug to receptacle on solenoid.
10. Perform operation check of engine anti-ice valve (refer to 75-10-00).

COMPRESSOR BLEED AIR SUPPLY LINE

Removal:

The compressor bleed air supply line extends from the compressor bleed port to the engine anti-icing valve (see Figure 72-10-3). Access to the compressor bleed air supply line is through the engine upper access door. To remove the compressor bleed air supply line, proceed as follows:

1. Remove 3 bolts securing compressor bleed air line to aft side of engine anti-icing valve.
2. Loosen clamp securing tube to hose section.
3. Remove air tube from hose. Discard gasket from end of tube attaching to aft pad of engine anti-icing valve.
4. Remove 4 bolts securing compressor bleed air tee to engine compressor section. Discard gasket.

Installation:

1. Place new gasket on compressor bleed air supply line attachment pad and secure tee to pad with 4 bolts lubricated with Fel Pro C-5 (or equivalent). Torque bolts to 16-19 in.-lbs and safety with lockwire.
2. Insert compressor bleed air supply line into hose section.
3. Place a new gasket on aft pad of engine anti-ice valve and install compressor bleed air line to valve with 3 bolts. Torque bolts to 20-25 in.-lbs and safety with lockwire.
4. Tighten clamp to 30-40 in.-lbs and safety with lockwire.

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INSPECTION/CHECK

CHECK ENGINE ANTI-ICING SYSTEM

CAUTION

Do not operate the engine anti-icing system for more than two minutes continuously.

NOTE

Illumination of either caution light indicates that the corresponding engine anti-ice valve is not operating correctly.

1. Start engines (refer to chapter 71).
2. Operate both engines at approximately 90 percent rpm. Make certain that RH ENG ANTI-ICE CONTROL, LH ENG ANTI-ICE CONTROL and ENG ANTI-ICE FAIL circuit breakers on right hand console are pushed in and that the ENGINE ANTI-ICE switch on the anti-ice control panel is at OFF. The RH and LH anti-ice fail caution lights on the caution and warning panel should be out.
3. Pull out the RH and LH engine anti-ice circuit breakers. Note that both the RH and LH engine anti-ice fail caution lights are illuminated.
4. Push in RH and LH engine anti-ice circuit breakers. Note that both RH and LH engine anti-ice fail caution lights go out.
5. Position the ENGINE ANTI-ICE switch to ON. The RH and LH caution lights should not come on [1]. Note that for both engines the engine pressure ratio (EPR) decreases slightly, and the exhaust gas temperature increases slightly.
6. Reposition ENGINE ANTI-ICE switch at OFF.
7. Shut down engines.
8. Secure airplane as desired.

[1] S/N 370-7; 380-1, -2 and -42 and Subs.

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CHAPTER 76

ENGINE CONTROLS

CHAPTER 76

ENGINE CONTROLS

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CHAPTER 76

ENGINE CONTROLS

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ENGINE CONTROLS

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ENGINE CONTROLS

DESCRIPTION

GENERAL

The engine controls consist of a system of linkages, bellcranks, and telescopic cables which transmit mechanical motion of the cockpit power levers to the hydromechanical engine fuel control unit. Latch assemblies for each power lever prevent power lever movement from the OFF position, or from IDLE into the OFF position, unless the latch has been released. The friction control knob on the left side of the quadrant, when rotated in a cw direction, increases resistance of the levers to movement from their preset positions. A mechanical stop, a component of the gust lock system, is mounted inside the throttle quadrant and prevents engine advancement to take-off power when the gust locks are engaged. Initial movement of the throttle lever from the OFF position operates a quadrant-mounted microswitch energizing the airplane fuel boost pump and engine ignition systems, and opening the shutoff valve in the hydromechanical engine fuel control unit. As the throttle is advanced, the movement is mechanically transmitted to the fuel control unit arm, positioning it at the desired setting. Throttle lever movement from the full fwd position will, at 95% engine rpm, operate a quadrant-mounted microswitch in the landing gear warning circuit.

ENGINE CONTROL QUADRANT

The engine control quadrant assy, on top of the cockpit center pedestal, consists of two throttle levers on which are mounted two thrust reverser control levers, two adjustable reverse thrust stops (S/N 370-2 thru -6, -8, and -9), a speed brake switch, and a friction control knob; an interlock solenoid and eight microswitches are within the quadrant. The two throttle levers and reverser control levers are connected to their respective engine fuel control units by mechanical linkages. Each throttle includes a separate safety latch button on the aft end of the quadrant. These latches must be released before the power levers can be moved from the closed position to IDLE or, on engine shutdown, from IDLE back to OFF. These latches are released by applying a downward pressure on the latch button. On S/N 370-2 thru -6, -8, and -9, each reverser control lever includes an adjustable stop assy. This stop is adjusted to prevent overboosting of the engine during max reverse-thrust operation. Each stop consists of a field temperature scale calibrated in degrees Fahrenheit, a field pressure altitude scale calibrated in thousands of feet on the end of an adjustable bellcrank, and a stop lug on the reverser control lever. Reverser control lever max travel is limited by the stop lug contacting the bellcrank. On S/N 370-1, -7, S/N 380-1 and subs, a fixed stop is mounted on the reverser control lever. This stop limits engine rpm to 97.5% during reverse thrust operation. Adjustment of this stop is accomplished during ground maintenance. A solenoid-actuated lock is provided for each reverser control lever. When the solenoids are de-energized, the locks prevent the reverser control levers from being moved past the idle reverse-thrust position. A friction control knob for the two throttle levers is on the left side of the quadrant. The microswitches within the quadrant are actuated by throttle lever and reverser control lever movement and function in the fuel boost pump, ignition, landing gear warning, and thrust reverser systems. The thumb-operated speed brake switch is mounted on the left throttle lever. A mechanical blocking component of the gust lock system prevents throttle advancement to take-off power if the gust locks have not been released. (See figures 76-00-2 or 76-00-4.)

TELESCOPIC CONTROL

Mechanical movement originating in the fwd bellcrank in the fwd cockpit area is transmitted through the telescopic control unit to the aft bellcrank in the fuselage at the engine pod pylon. The cable operates inside a conduit, keeping friction at a min. On S/N 370-1, -7, S/N 380-1 and subs, the aft bellcrank in the cabin (opposite the pylon) incorporates a thrust reverser and speed brake limit switch that prevents stowing of the thrust reversers above 80% engine rpm and illuminates the speed brake open caution light providing the speed brake is extended. (See figure 76-00-2.)

THROTTLE LEVER/FUEL CONTROL TRAVEL — S/N 370-2 THRU -6, -8, AND 9

A bellcrank assy on the engine cross shaft is constructed of a serrated crank mated to an adjustable clevis. The clevis may be shifted as necessary to obtain the required amount of travel to align the throttle and fuel control unit movements. An index on each part, when aligned, serves as an original point of reference. Extending the clevis increases the amount of throttle lever travel, and shortening the clevis increases the fuel control arm movement.

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DESCRIPTION (Cont'd)

THROTTLE LEVER/FUEL CONTROL TRAVEL — S/N 370-1 AND -7, AND S/N 380-1 AND SUBS

An arm and rod end on the outbd end of the connecting shaft through each pylon transmits throttle motion to the fuel control. (See figure 76-00-2.) Arm/rod end total length can be adjusted as required to obtain equal LH and RH throttle movement with equal LH and RH fuel control movement. Extending the rod end reduces throttle movement required for full travel of the fuel control; shortening the rod end increases throttle movement required for full travel of the fuel control.

WOODARD FAN SPEED SYNCHRONIZER SYSTEM — MODEL NA 265-80 AIRPLANES (OPTIONAL EQUIPMENT)

The Woodard fan speed synchronizer system consists of a control switch, indicator light, master unit in the cockpit, an actuator, a flexible shaft, a fuel control trimmer on the left engine, and the necessary electrical wiring to complete the system. (See figure 76-00-7.)

The synchronizer system utilizes the fan rpm magnetic pickup sensor on each engine to automatically adjust the fuel control linkage of the left engine, thus adjusting the left engine fan rpm to match the right engine fan rpm. The fan rpm of the left engine is varied over a limited range (approximately 1/2% rpm) to match any variation in fan speed of the right engine. The limited range of rpm variation of the left engine prevents the engine from losing more than approximately 1/2% rpm in case the right engine is shut down with the synchronizing system energized.

The engine fan speed synchronizer control switch and indicator light are mounted on the instrument control panel above the engine instruments. The switch energizes the system providing electrical power is available on the essential bus and the engine synchronizer circuit breaker is engaged. The switch should be in the OFF position during take-off, landing, and any time the engine (turbine) rpm is below 60%. The engine fan speed synchronizer master control unit is located at FS 78 (adjacent to LH rudder pedals). The control unit receive electrical signals (pulses) from the RH and LH engine fan rpm magnetic pickups (sensors). The pulse rates of these signals are compared within the control unit, resulting in a correction voltage (if engine fans are out of sync) being applied to the fan synchronizer actuator.

The engine fan speed synchronizer control actuator is installed on the inbd side of the top nacelle door of the LH engine. (See figure 76-00-7.) The actuator is a stepping type motor that operates on signals supplied by the control unit. When the actuator is energized, rotary motion of the motor is transmitted to the fuel control trimmer assy by a flexible shaft. During operation, when the fan synchronizer switch is placed in the OFF position, the actuator remains energized until the trimmer assy is centered within its operating range. This ensures the proper rigging between the engine power lever and engine fuel control during the min and max engine rpm settings.

The fuel control trimmer assy is installed on the LH engine main fuel control arm. The trimmer is extended or lengthened by actuation of the actuator as required to maintain a fan speed synchronized condition with the RH engine. The trimmer assy is connected to the power control lever system by a turnbuckle and an eye fitting. The eye fitting provides the necessary adjustment for rigging the trimmer assy to the engine power control system.

During normal operation (engines operating above 60% rpm and within 1/2% rpm) when the FAN SYNC switch is positioned at ON, the control unit is energized. (See figure 76-00-7.) At the same time electrical signals (pulses) from the fan speed magnetic pickups (sensors) are applied to the control unit. The control unit compares the two pulses for rate changes. If a rate change exists (engine not synchronized), the control unit detects this rate change and provides a correction voltage which is applied to the actuator. The actuator is then energized, which in turn actuates the trimmer assy resulting in a slight movement of the main fuel control arm. The left engine speed is then varied proportionately to the pulse rate change resulting in an increase or decrease of the left engine fan rpm. The left engine fan rpm is then matched to the right engine fan rpm to obtain a synchronized condition.

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ENGINE CONTROLS

FAULT ISOLATION

ENGINE CONTROLS FAULT ISOLATION

Engine control system troubles can be readily determined by isolation of the system segments. Disconnect telescopic cable at aft bellcrank and operate throttle to determine trouble locations. Continue to break down linkages until defective component is found. Replace faulty part(s) and rig system.

FAN SPEED SYNCHRONIZER SYSTEM FAULT ISOLATION — MODEL NA 265-80 AIRPLANES (OPTIONAL EQUIPMENT)

Trouble shooting of the engine synchronizer system consists of continuity checks of the system wiring or replacement of the control unit or actuator. The engine synchronizer control unit or actuator is at fault if fan rpm indications are normal during engine operation and the fan speed magnetic pickups (sensors) are known to be serviceable. When performing a continuity check of the system wiring, refer to the wiring diagram manual.

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ENGINE CONTROL

ADJUSTMENT/TESTS

ENGINE CONTROL RIGGING

Engine Control Rigging S/N 370-2 Thru -6, -8, and -9

NOTES

- ***Prior to rigging engine control system and subsequent to installing an engine and/or fuel control, fuel control linkage should be checked for proper rigging and adjusted as required. Refer to Adjusting Fuel Control Linkage — S/N 370-2 thru -6, -8, and -9.***
- ***See Figure 76-00-1.***

1. Disconnect fwd telescopic units from bellcranks.
2. Depress latch locks and place power levers in idle-cutoff position. Release latch locks.
3. Adjust fwd throttle rods, to obtain slip-fit of 5/16-in. diameter rig pins in both bellcranks. Secure locknuts on rods.
4. Disconnect aft telescopic units from bellcranks.
5. Install 5/16-in. rig pins in upper arms.
6. Ensure that index marks on clevis and bellcrank are aligned.
7. Adjust aft vertical rods, to position fuel control indicator to OFF position. Secure locknuts on rods.

CAUTION

Hold head of fwd and aft clamp bolts while tightening. Torque nut to 35-40 in.-lbs.

8. Position cables on slide of fwd telescopic units. Secure cable clamping bolts.

NOTE

Fwd bellcranks are not adjustable.

9. Attach fwd telescopic units to bellcranks.
10. Position cables on sliders of aft telescopic units. Secure clamping bolts.
11. Adjust length of aft telescopic units so units fit bellcranks. Secure locknuts and rod-end check nuts.
12. Remove all rig pins. Refer to steps 3 and 5.
13. Move throttle control lever fwd until engine fuel control engages the max travel stop. The throttle control lever should be 7/32 (+1/32/-1/16) in. from fwd mechanical stop of lever. The fwd stop requirements can be adjusted by the serrated clevis and bellcrank as outlined in steps 14 and 15. (See Figure 76-00-1.)
14. Loosen serrated clevis clamp bolts and establish new radius. Throttle control lever travel can be lengthened by increasing radius of arm; throttle control lever travel can be shortened by decreasing the radius of arm.
15. After adjusting to a new radius, install rig pins and adjust vertical rods per step 7. Remove rig pins after adjusting.

NOTE

Make sure that friction control is released before force check.

16. Apply a scale tangentially in line with throttle lever at center of throttle grip. Max force required to operate throttle control, in either direction, within 1/8 in. of stops should be 7.5 lbs.
17. To adjust microswitches and latch stops within the quadrant, refer to Adjusting Engine Control Quandrant.

Engine Control Rigging S/N 370-1, -7 and S/N 380-1 and Subs (Without Woodard Fan Speed Synchronizer)

NOTE

See Figure 76-00-2.

1. Disconnect fwd telescopic units from bellcranks.
2. Depress latch lock and position power lever in idle-cutoff position. Release latch lock.
3. Loosen locknuts on fwd power lever rod and adjust to obtain a slip-fit of the 5/16-in. diameter rig pins on both ends of torque tube. Tighten locknuts. Leave rig pins installed.
4. Remove necessary trim panels to gain access to rear bellcrank assembly. Bellcrank is located in cabin area between FS 306 and 316.
5. Remove cover plate from bellcrank and switch assembly and disconnect telescopic unit from control arm. Remove arm from splined shaft
6. Rotate engine fuel control arm CCW as required to position are 0.000 to 0.006 in. from external mechanical stop. (Fuel control quadrant should indicate 0°.)

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ADJUSTMENT/TESTS (Continued)

Engine Control Rigging S/N 370-1, -7 and S/N 380-1 and Subs (Without Woodward Fan Speed Synchronizer) (Cont'd)

7. Install control arm on splined shaft. Control arms should be approximately 25° fwd from the control arm vertical position. Tighten control arm retaining nut.

CAUTION

Hold head of fwd and aft clamp bolts while tightening nut. Torque nut to 35-40 in.-lbs.

8. Position cables on sliders of fwd telescopic units. Secure cable clamping bolts.

NOTE

Fwd bellcrank is not adjustable.

9. Install fwd telescopic unit to fwd bellcrank. Tighten and safety nut.
10. Position cables on sliders of aft telescopic unit. Secure clamping bolt.
11. With fuel control arm and control arm positioned as per steps 6 and 7, adjust length of aft telescopic unit as required to install connnecting bolt. Tighten nut and install cotter pin. Tighten rod end bearing locknut.
12. Remove rig pins installed in step 3.

NOTE

A rig pin (2C5320) installed in fuel control stop arm may be used to accurately establish the 13° position.

13. Position power lever so fuel control lever indicates 13°.
14. On left engine ensure that length (center to center of push-pull rod attaching holes) of push-pull rod (turnbuckle link) is 5.12 in. (See Figure 76-00-2.)
15. Adjust the quadrant idle stops to contact idle stop bellcrank.
16. Position power lever fwd until the max travel stop is within 0.000 to 0.010 in. of contacting stop. The power lever should be 1/4 (±1/8) in. from its fwd mechanical stop.

NOTE

Make sure friction control is released before force check.

17. Apply a scale tangentially in line with power lever at center of power lever grip. Max force required to operate throttle control, in either direction within 1/8 in. of stops should be 7.5 lbs.
18. Move power lever through full travel and check that fuel control lever matches throttle position. Idle cutoff position 0°, IDLE position 13°, and full open position is 90°.
19. Check complete system for security and proper safeties. Ensure all locking tabs are bent over the jam nuts.
20. Adjust thrust reverser and speed brake 80% rpm limit switch. (Refer to Adjusting 80% Rpm Limit Switch S/N 370-1, -7, S/N 380-1 and Subs.) Check for proper operation of boost and ignition, and landing gear warning switches on the quadrant.
21. Install cover plate over bellcrank assembly in cabin. Install trim panels.
22. Start engine and position THRUST REVERSER control switch at ON.
23. Check thrust reverser max rpm adjustments per steps A thru F.
    A. Place applicable reverser thrust lever to deploy. Check that UNLOCK indicator light (red) and the DEPLOY indicator light (blue) come on. Thrust reverser should deploy within two seconds.

CAUTION

Do not operate engine at 72% rpm with reverser deployed for more than 5 to 10 seconds.

    B. When DEPLOY light comes on increase rpm of engine to 72% rpm by moving the reverse thrust control lever aft past the interlock stop.
    C. When 72% rpm engine speed is obtained stow reverser by setting the thrust reverser control switch to EMERG STOW.
    D. Check that DEPLOY and UNLOCK indicator lights are out, and ARM indicator light stays on.
    E. After thrust reverser is stowed, advance the reverser thrust control lever to the max position (mechanical stop).

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ADJUSTMENT/TESTS (Continued)

Engine Control Rigging S/N 370-1, -7 and S/N 380-1 and Subs (Without Woodward Fan Speed Synchronizer) (Cont'd)

Step 23 (Cont'd)

F. After engine rpm has stabilized, check engine rpm to see if it is within 95-97% rpm, but does not exceed 97.5% rpm.

NOTE

If the reverser control lever is not in alignment at max reverser thrust, the stop attached to the reverser control lever may be shifted up or down to align reverser control lever. Tighten stop screw after adjustment.

24. Adjust reverser stop adjusting screw (Figure 76-00-2) as necessary to limit reverser control lever movement to 97.5T engine rpm.
25. Reduce engine speed to idle rpm and stow thrust reverser. Set THRUST REVERSER control switch to OFF.
26. Shut down engine.

Engine Control Rigging S/N 370-1, -7, and S/N 380-1 and Subs (With Woodard Fan Speed Synchronizer)

NOTE

See Figure 76-00-2.

1. Disconnect fwd telescopic units from bellcranks.
2. Depress latch look and position engine power levers in idle-cutoff position.

NOTE

The trimmer assembly and actuator will automatically center by setting the FAN SYNC switch at OFF and applying electrical power to airplane.

3. On left engine, check that rod end (trimmer assembly) at fuel control is centered within its operating range and total length (center to center of eye bolts on trimmer assembly) is 5.12 in.
4. Remove side panels on center pedestal and verify slip-fit of rig pins at fwd bellcrank. If slip-fit is not obtained, loosen locknuts on fwd power lever rod and adjust as required to obtain a slip-fit of the 5/16-in. diameter rig pins on both ends of the torque tube. Tighten locknuts. Leave rig pins installed.
5. Remove necessary trim panels to gain access to rear bellcrank assembly. This bellcrank is located in the cabin area between FS 306 and 316.
6. Remove cover plate from bellcrank and switch assembly and disconnect teleflex cable from control arm. Remove control arm from splined shaft.
7. Rotate engine fuel control arm ccw as required to position arm 0.000 to 0.006 in. from the external mechanical stop (fuel control quadrant should indicate 0°).
8. Install control arm on splined shaft. Control arms to be approximately 25° fwd from the control arm vertical position. Tighten control arm retaining nut.

CAUTION

Hold heads of fwd and aft clamp bolts while tightening nut. Torque nut to 35-40 in.-lbs.

9. Position cables on sliders of fwd telescopic units. Secure cable clamping bolts.

NOTE

Fwd telescopic unit is not adjustable.

10. Install fwd telescopic unit to fwd bellcrank. Tighten and safety nut.
11. Position cables on sliders of aft telescopic unit. Secure clamping bolt.
12. With engine fuel control arm positioned per step 7 and control arm positioned per step 8, adjust length of aft telescopic unit as required to install connecting bolt. Tighten nut and install cotter pin. Tighten rod end bearing locknut.
13. Remove rig pins installed in step 4.

NOTE

A rig pin (2C5320), installed in engine fuel control arm, may be used to accurately establish the 13° position.

14. Position engine power lever so fuel control indicator is at the 13° mark. Adjust power lever idle stops to contact stop pin.

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ADJUSTMENT/TESTS (Continued)

Engine Control Rigging S/N 370-1, -7 and S/N 380-1 and Subs (With Woodward Fan Speed Synchronizer) (Cont'd)

15. Remove fuel control arm to trimmer attach bolt. Lengthen trimmer 2-1/2 turns and temporarily install attach bolt.
16. Position power lever full fwd (max power). Check that stop lug on engine fuel control arm is within 0.000 to 0.010 in. of contact with max rpm mechanical stop.
17. Remove engine fuel control arm to trimmer attach bolt. Shorten trimmer assembly five turns and temporarily install attach bolt.
18. Position engine power lever full fwd (max power). Check that stop lug on engine fuel control arm is within 0.000 to 0.010 in. contact with the max rpm mechanical stop.
19. Remove engine fuel control arm to trimmer attach bolt and center trimmer assembly within its operating range. Install attach bolt and safety.
20. Position engine power lever fwd until the max travel stop is with 0.000 to 0.010 in. of contacting stop. The engine power levers should be 1/4 (±1/8) in. from its fwd mechanical stop.

NOTE

Ensure friction control is released before force check.

21. Apply a scale tangentially in line with engine power lever at center of grip. Max force required to operate throttle control, in either direction within 1/8 in. of stops should be 7.5 lbs.
22. Move engine power lever through full travel and check that engine fuel control lever matches the throttle position. Idle cutoff position 0°, IDLE position 13°, and full open position is 90°.
23. Check complete system for security and proper safeties.
24. Adjust thrust reverser and speed brake 80% rpm limit switch. (Refer to Adjusting 80% Rpm Limit Switch S/N 370-1, -7, S/N 380-1 and Subs.) Check for proper operation of the boost and ignition and landing gear warning switches with the quadrant.
25. Install cover plate over bellcrank assembly in cabin. Install trim and pedestal panels.
26. Start engine and position THRUST REVERSER control switch at ON.
27. Check engine idle RPM with thrust reverser deployed as follows:

NOTE

The indications called out in this check are not independently adjustable. Failure of this check indicates a mechanical failure in the throttle quadrant, improper engine control rigging, or a fuel control problem.

A. Simultaneously raise both thrust reverser levers against interlock stop and observe RPM on both engines.

B. Both engines should spool up together, stabilize within 2% rpm of each other, and both be within 64 to 68% rpm range.

C. Return both thrust reverser levers to stowed position.

28. Check thrust reverser max rpm adjustments per steps A thru F.

A. Set applicable reverser thrust lever to DEPLOY. Check that UNLOCK indicator light (red) and the DEPLOY indicator light (blue) come on. Thrust reverser should deploy within two seconds.

CAUTION

Do not operate engine at 72% rpm with reverser deployed for more than 5 to 10 seconds.

B. When DEPLOY light comes on, increase engine speed to 72% rpm by moving reverse thrust control lever aft past the interlock stop.

C. When 72% rpm engine speed is reached, stow reverser by switching the thrust reverser control switch to EMERG STOW.

D. Check that DEPLOY and UNLOCK indicator lights are out, and the ARM indicator light stays on.

E. After thrust reverser is stowed, advance the reverse thrust control lever to the max position (mechanical stop).

F. After engine rpm has stabilized, check engine rpm to see if it's within 95-97% rpm, but does not exceed 97.5% rpm.

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ADJUSTMENT/TESTS (Continued)

Engine Control Rigging S/N 370-1, -7 and S/N 380-1 and Subs (With Woodward Fan Speed Synchronizer) (Cont'd)

NOTE

If the reverser control lever is not in alignment at max reverser thrust, the stop attached to the reverseer control lever may be shifted up or down to align the reverser control lever. Tighten stop screw after adjustment.

29. Adjust reverser stop adjusting screw (Figure 76-00-2) as necessary to limit reverser control lever movement to 97.5% engine rpm.
30. Reduce engine speed to idle rpm. Set THRUST REVERSER control switch to OFF.
31. Perform operational check of fan synchronizer system.
32. Shut down both engines.

Fuel Control Linkage Adjustment (S/N 370-2 Thru -6, -8, and -9)

NOTE

To synchronize movement of the throttle levers, the fuel control linkage (between engine cross-shaft and fuel control) must be adjusted the same on both engines. When an engine and/or fuel control is replaced, it may be necessary to index and adjust the fuel control to cross-shaft linkage. Check and/or adjust linkage per steps 1 thru 9.

1. Carefully measure distance (A) between centers of fuel control shaft and engine cross-shaft. (See Figure 76-00-8.)
2. Next, carefully measure distance (B) between centers of fuel control arm to cross-shaft arm clevis retaining bolts. This distance (B) should be the same as the measurement (A) noted in step 1.
3. If adjustment is necessary, cut safetywire and remove retaining bolt from one end of clevis; then loosen jam nut and adjust length (B) of clevis to be the same as dimension (A) in step 1.
4. After adjustment of fuel control linkage clevis, reinstall clevis and retaining bolt. Tighten and safetywire retaining bolt and jam nut.
5. Remove cotter pin and disconnect throttle linkage aft vertical rod at engine cross-shaft bellcrank clevis. (See Figure 76-00-1.) Retain nut, washer, and bolt for reinstallation.
6. Rotate fuel control arm CW until shoulder on index pin is against fuel control mechanical stop (MIL position); then check for clearance of 0.010 to 0.063 in. at interference points on clevis on cross-shaft arm. (See Figure 76-00-8.)
7. If insufficient clearance exists at interference points, re-index fuel control arm per steps A thru G.

CAUTION

When loosening fuel control lever retaining nut, and re-indexing fuel control linkage, ensure that index pin (Hamilton Standard P/N 561531), which acts as stop for power lever, is not displaced when tension is relieved by loosening nut.

A. Rotate fuel control arm CW to MIL position.
B. Cut safetywire and loosen fuel control arm retaining nut.
C. Separate serrated faces of fuel control arm, serrated face plate, and control shaft.
D. Rotate cross-shaft arm CCW to give clearance (0.010 to 0.063 in.) at interference points.
E. Rotate serrated face plate until serrations on plate match serrations on fuel control shaft and fuel control arm. Install retaining nut.

NOTE

If linkage appears excessively stiff, actuate at least 50 times to facilitate run-in of bearings.

CAUTION

When rigging power lever linkage on engines equipped with Hamilton Standard fuel controls and before tightening shaft nut, bottom index pin with head or pointer end flush against power lever shaft to ensure proper contact of index pin with power lever stop.

CAUTION

Do not wrench Holley fuel control bleed strap signal valve screw at any time to actuate bleed strap mechanism as shearing of serration will occur in bleed actuator housing when lever and shafts are disengaged.

F. Tighten fuel control arm retaining nut (35 to 45 in-lbs) and repeat step 6 to check for proper clearance at interference points. After checking clearance, safetywire fuel control arm retaining nut.

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MAINTENANCE PRACTICES (Continued)

**FuelControl Linkage Adjustment
(S/N 370-2 Thru -6, -8, and -9) (Cont'd)**

G. If further indexing of fuel control arm is necessary, loosen fuel control arm retaining nut and repeat steps C thru F until proper clearance is obtained when index pin is against fuel control mechanical stop (MIL position); then tighten fuel control arm retaining nut (35 to 45 in.-lbs) and safetywire nut.

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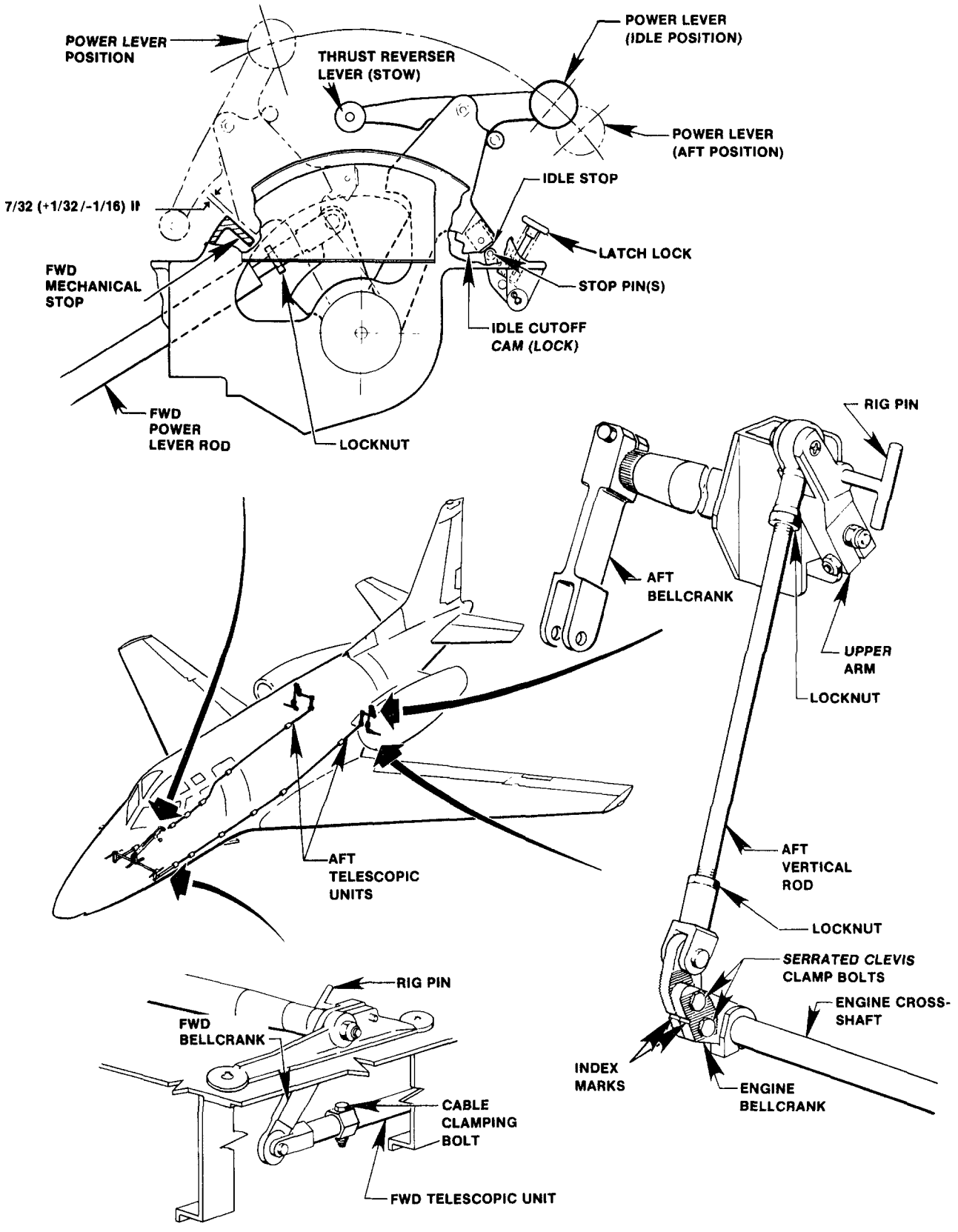

Figure 76-00-1 — Rigging Engine Controls Airplanes S/N 370-2 thru -6, -8 and -9

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ADJUSTMENT/TESTS (Cont'd)

Fuel Control Linkage Adjustment (S/N 370-2 Thru -6, -8, and -9) (Cont'd)

NOTE

The fuel control arm should be approximately 5° below horizontal when it is rotated to the full OFF position.

8. Connect throttle linkage aft vertical rod at engine cross-shaft bellcrank clevis. Install bolt, washer, and nut. Subsequent to rigging engine control system, tighten nut, and install new cotter pin.
9. Check rigging of engine control system and adjust as required. (Refer to Rigging Engine Controls S/N 370-2 Thru -6, -8 and -9.)

RH Engine Control Teleflex Installation (After SOK 18 Installed, and 380-56 and Subs)

1. Disassemble control teleflex for assembly on mounting bracket (5) and mounting block (17) per steps A thru C. (See figure 76-00-3.)
   - A. Back off nuts (3 and 15) attaching ends of casing assembly (2) to inbd and outbd swivel assemblies (7 and 19).
   - B. Remove both rod ends (8 and 20) by loosening nuts (11 and 21) securing plug bolts (9 and 26) to cable (1) and pulling rod ends (8 and 20) free.
   - C. Slide swivel assemblies (7 and 19) from cable (1).
2. Assemble the control teleflex on the RH engine per steps A thru P.
   - A. Run lower mounting nut (6) to bottom of threads on inbd swivel assembly (7).
   - B. Run lower mounting nut (18) to bottom of threads on outbd swivel assembly (19).
   - C. Insert inbd swivel assembly (7) up through hole in mounting bracket (5) and install upper mounting nut (4). Torque to 80 in.-lbs.
   - D. Insert outbd swivel assembly (19) up through holes in mounting block (17) and install upper mounting nut (16). Torque to 80 in-lbs.
   - E. Feed casing assembly (2) through grommet at top center of RH engine and feed cable (1) through inbd and outbd swivel assemblies (7 and 19).
   - F. Thread cable (1) through swivel assemblies (7 and 19).
   - G. Secure inbd and outbd ends of casing assembly (2) to inbd and outbd swivel assemblies (7 and 19) using nuts (3 and 15). Torque nuts to 50-60 in.-lbs.
   - H. Loosen upper mounting nuts (4 and 16).
   - I. On inbd end of control teleflex, slide rod end (8) onto cable (1) and tube of swivel assembly (7). Thread cable (1) through hole in plug bolt (9) until end of cable (1) is visible in lower inspection hole. Tighten nut (11) (DO NOT TURN PLUG BOLT) to 35-40 in.-lbs.
   - J. On outbd end of control teleflex, slide rod end (2) onto cable (1) and tube of swivel assembly (19). Thread cable (1) through hole in plug bolt (26) until end of cable (1) is visible in lower inspection hole. Tighten nut (21) (DO NOT TURN PLUG BOLT) to 35-40 in-lbs.
   - K. Set RH throttle lever at idle.
   - L. Install inbd rod end (8) on rod end assembly (13) using bolt (14) and nut (12). Torque to 12-15 in.-lbs. and safety with cotter pin.
   - M. Install outbd rod end (20) on fuel control lever assembly (23) using bolt (24) and nut (25). Torque to 12-15 in.-lbs and safety with cotter pin.
   - N. With RH throttle lever at idle, adjust upper and lower mounting nuts (4, 6, 16, and 18) to position fuel control lever at idle on protractor. When proper setting achieved, torque upper mounting nuts (4 and 16) to 80-100 in.-lbs.
   - O. Run throttle full travel and check that both fuel control stops are contacted, that swivel assembly (7 and 19) tube is visible in rod end (8 and 20) upper inspection hole, and rod end (8 and 20) does not contact ball joint of swivel assembly (7 and 19).
   - P. Safetywire upper and lower mounting nuts (4, 6, 16, and 18).

MAJOR CHANGE

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ADJUSTMENT/TESTS (Cont'd)

Engine Control Quadrant Adjustment

NOTE

Adjustment of the engine control quadrant is limited to the microswitches and throttle lever latch stops.

1. Adjust throttle lever idle stop per steps A thru D.
   A. Position throttle lever so that interlocking pawl in throttle lever is lined up with locking hole in support assembly.
   B. Lift reverser control lever approximately 1.0 in. to fully engage locking pawl in throttle lever and support assembly.
   C. Position throttle lever to mid-position of interlock stop limits.
   D. Adjust idle stop to contact stop pin of idle lock bellcrank.
2. Adjust landing gear warning switches per steps A thru E.

NOTE

Warning switches are set to actuate when power levers are advanced to a point 1.09 in. aft of fwd stop.

   A. Start engine. (Refer to chapter 71.) Advance power levers to obtain 95% rpm. When engines have stabilized at 95% rpm, mark position of each power lever on quadrant. (Place a piece of tape on quadrant and mark tape.)
   B. Shut down engines. (Refer to chapter 71.)
   C. Adjust landing gear warning switches to actuate when power levers advance past 95% rpm position.
   D. Check that switches remain actuated while levers are advanced to the full fwd stop. (Switches are actuated when normally open contacts are closed.)
   E. Remove tape from quadrant.
3. Adjust fuel boost and ignition switches per steps A thru C.
   A. Move throttle lever from idle stop to idle cutoff position; then position throttle lever one third this distance (travel) fwd of idle cutoff.

NOTE

Switches are deactuated when the normally closed contacts are closed. Switches should remain deactuated through the remaining fwd travel of the throttle lever.

   B. Adjust switches to deactuate.
   C. Check that switches are actuated (closed contacts open) from idle cutoff to one-third position.
4. Adjust thrust reverser arming switches per steps A thru F.
   A. Set throttle lever at IDLE.
   B. Lift reverser control lever approximately 1.0 in., or until lever contacts unlock solenoid.
   C. Push throttle lever fwd until throttle movement is stoppped by the interlocking pawl assembly.

NOTE

The swtich is actuated when the normally closed contacts are open.

   D. Adjust switch to actuate.
   E. Push reverser control levers down.

NOTE

The switch is deactuated when the normally closed contacts are closed.

   F. Advance throttle levers 0.20 in. and check that switch is deactuated.
5. Adjust reverser control lever stow and deploy switches per steps A thru F.
   A. Position throttle levers at IDLE.
   B. Check that reverser control levers are in full down (stowed) position.
   C. Lift reverser control levers 1-3/4 in.

NOTE

Switches are deactuated when the normally open contacts are open.

   D. Adjust switches to deactuate.

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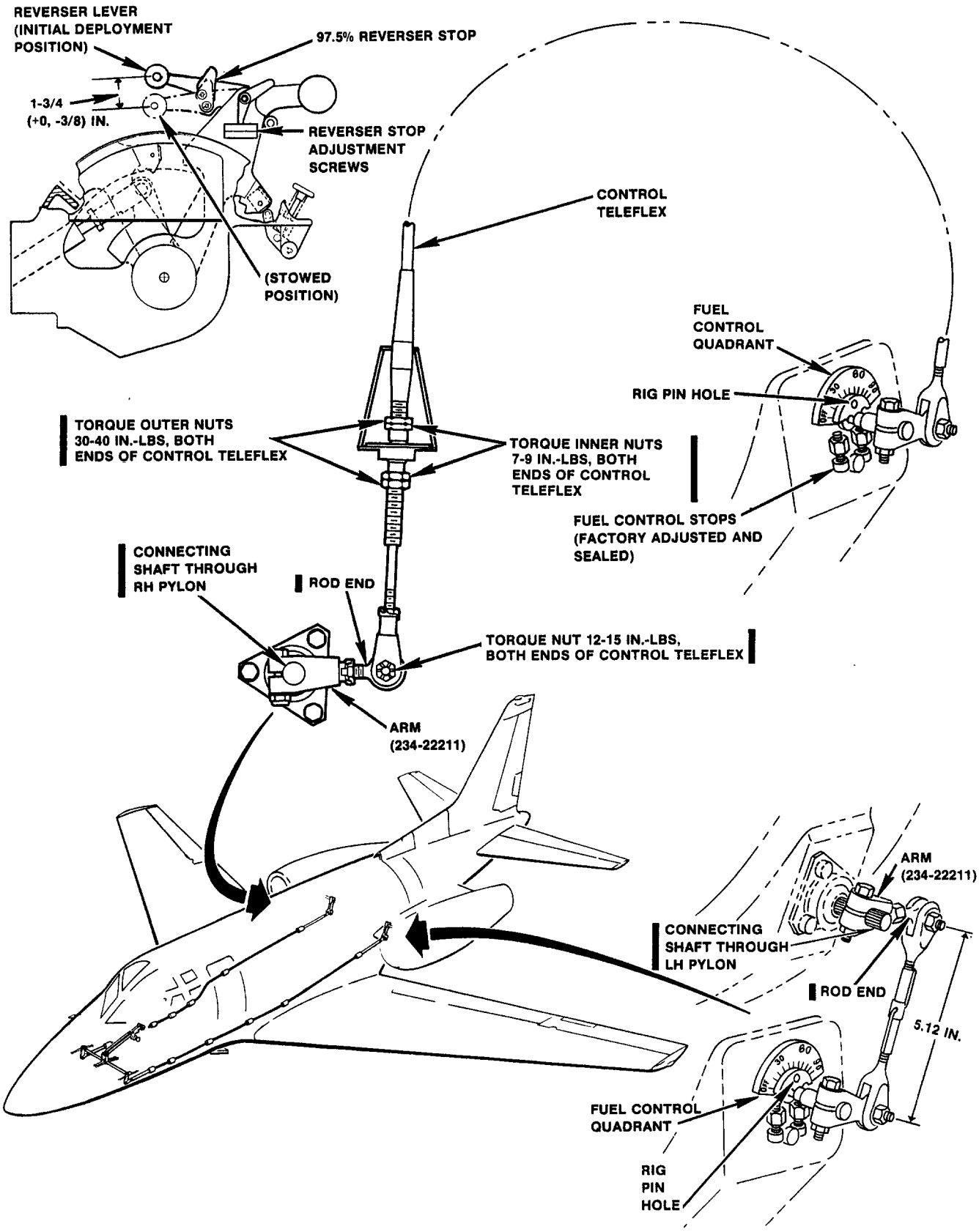
REVERSER LEVER (INITIAL DEPLOYMENT POSITION)
97.5% REVERSER STOP
1-3/4 (+0, -3/8) IN.
REVERSER STOP ADJUSTMENT SCREWS
CONTROL TELEFLEX
(STOWED POSITION)
FUEL CONTROL QUADRANT
RIG PIN HOLE
TORQUE OUTER NUTS 30-40 IN.-LBS, BOTH ENDS OF CONTROL TELEFLEX
TORQUE INNER NUTS 7-9 IN.-LBS, BOTH ENDS OF CONTROL TELEFLEX
FUEL CONTROL STOPS (FACTORY ADJUSTED AND SEALED)
CONNECTING SHAFT THROUGH RH PYLON
ROD END
TORQUE NUT 12-15 IN.-LBS, BOTH ENDS OF CONTROL TELEFLEX
ARM (234-22211)
ARM (234-22211)
CONNECTING SHAFT THROUGH LH PYLON
ROD END
5.12 IN.
FUEL CONTROL QUADRANT
RIG PIN HOLE

Rigging Engine Controls
S/N 370-1, -7 and S/N 380-1 and Subs
Figure 76-00-2 (Sheet 1 of 2)

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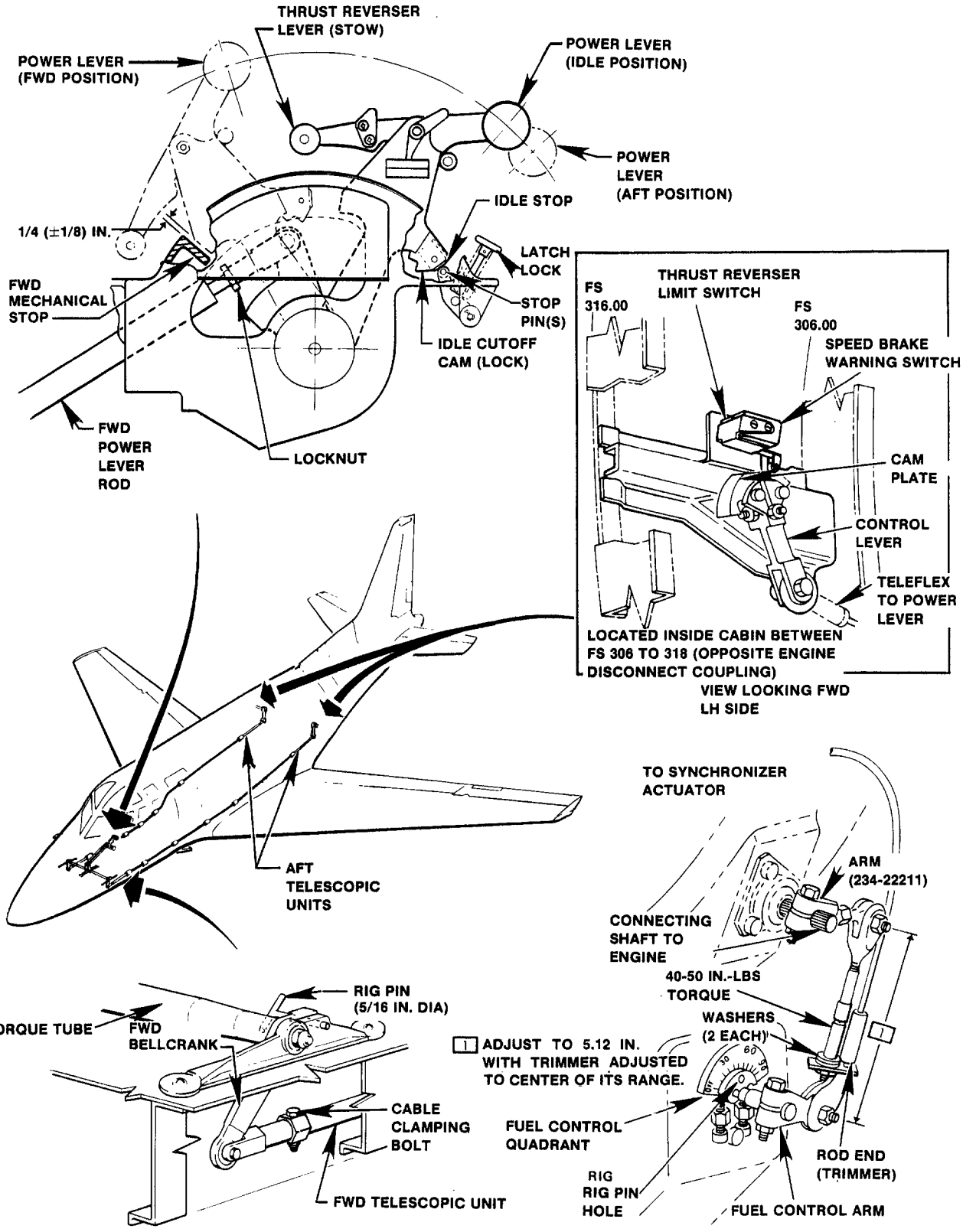

Rigging Engine Controls
S/N 370-1, -7 and S/N 380-1 and Subs
Figure 76-00-2 (Sheet 2 of 2)

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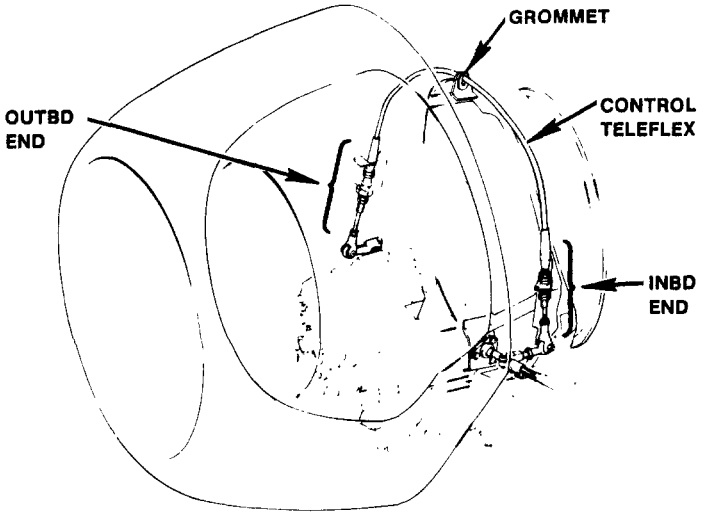
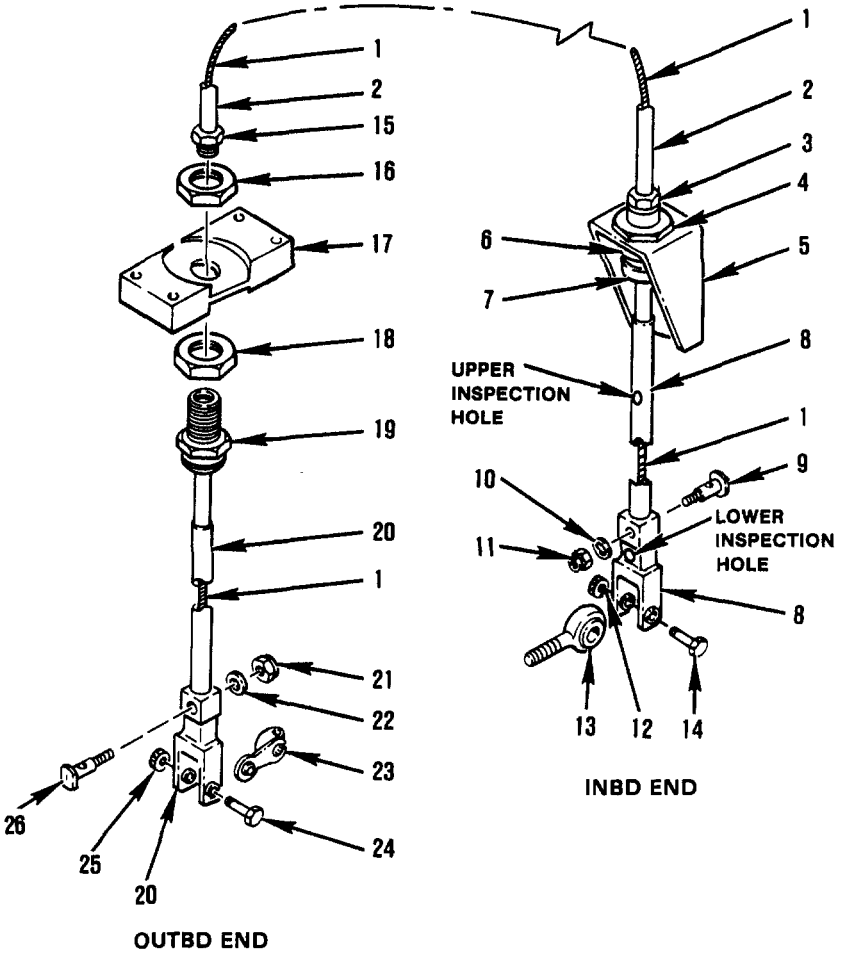


RH ENGINE



1 CABLE
2 CASING ASSEMBLY
3 NUT
4 UPPER MOUNTING NUT
5 BRACKET
6 LOWER MOUNTING NUT
7 SWIVEL ASSEMBLY
8 ROD END
9 PLUG BOLT
10 WASHER
11 NUT
12 NUT
13 ROD END ASSEMBLY
14 BOLT
15 NUT
16 UPPER MOUNTING NUT
17 MOUNTING BLOCK
18 LOWER MOUNTING NUT
19 SWIVEL ASSEMBLY
20 ROD END
21 NUT
22 WASHER
23 LEVER ASSEMBLY
24 BOLT
25 NUT
26 PLUG BOLT

NEW

Installing RH Engine Control Teleflex
After SOK 18 and S/N 380-56 and Subs
Figure 76-00-3

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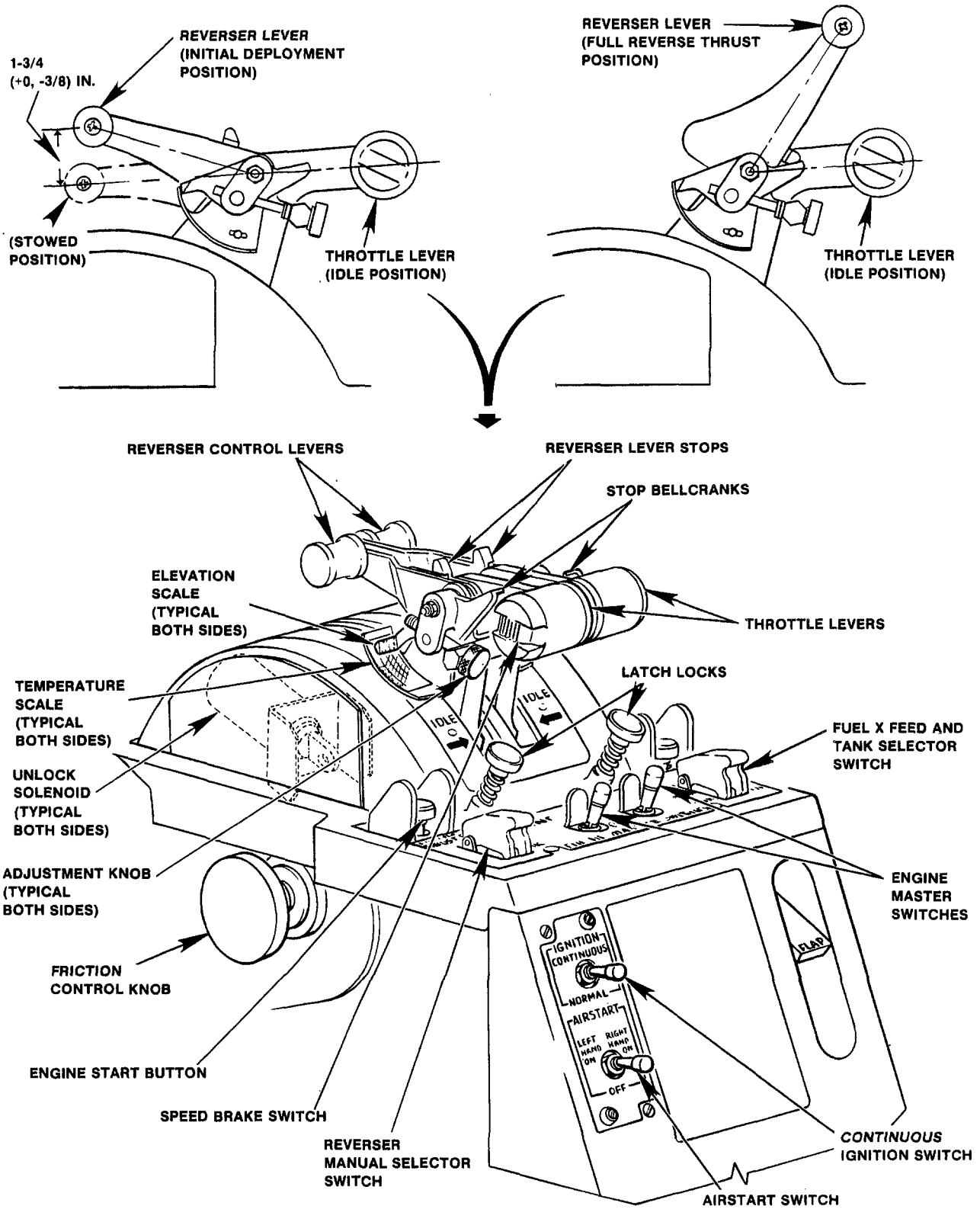

Engine Control Quadrant
S/N 370-2 thru -6, -8 and -9
Figure 76-00-4

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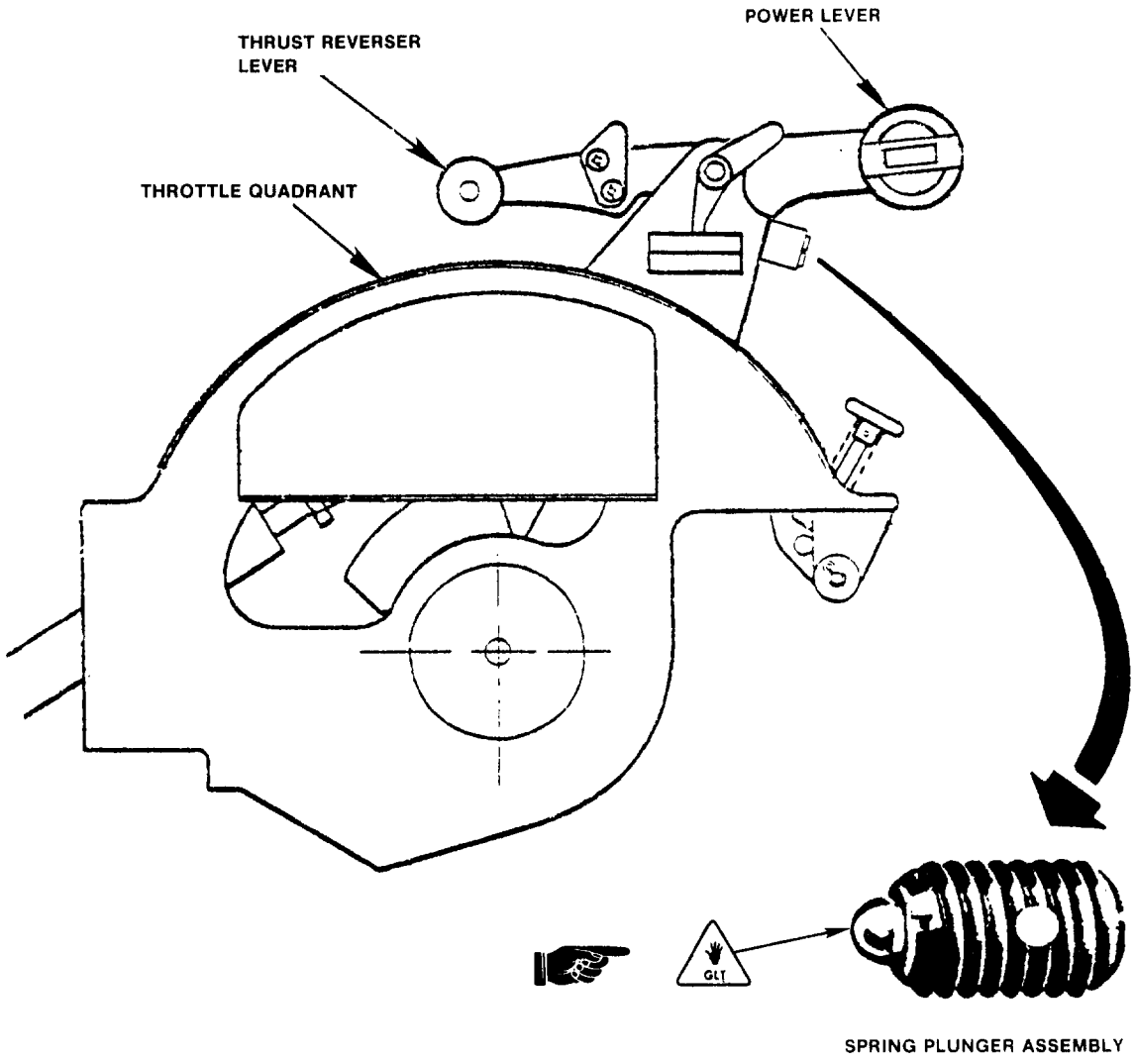
Using a push-pull type scale capable of 0 to 10 lbs, adjust spring plunger assemblies by turning in or out as required to obtain a 2-4 lb force measured at the thrust reverser knobs.



NOTE

Lubricate area on plunger assembly as shown using MIL-G-81827 grease.

Figure 76-00-5 — Adjusting Thrust Reverser Spring Plunger Assembly S/N 370-7 and 380-1 and Subs

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ADJUSTMENT/TESTS (Continued)

Engine Control Quadrant Adjustment (Cont'd)

E. Move reverser control levers up and aft to solenoid stops.

F. Move reverser control levers fwd and down, and check that switches actuate (normally open contacts closed) when reverser control levers are 1-3/4 (+0/-3/8) in. from full down position.

6. Adjust auxiliary throttle unlock solenoids per steps A thru J. (See Figure 76-00-6.)

NOTE

The procedure is the same for both unlock solenoids.

A. Using a thin-bladed knife or similar tool, carefully pry access panels from side of throttle quadrant. (Do not lose brass drive screws securing panel to quadrant.)

B. Set throttle lever at idle position and reverser control lever at stowed position.

C. Manually push in unlock solenoid plunger until it is bottomed.

D. Move thrust reverser control lever toward full reverse thrust position and check that reverser control lever arm (Figure 76-00-6) clears solenoid plunger lockpin by 0.030 to 0.100 in.

E. If clearance is not within tolerance, loosen solenoid mounting screws and move solenoid fwd or aft as required to obtain correct clearance.

F. Tighten solenoid attaching screws.

G. Recheck clearance.

H. Release solenoid plunger and verify that thrust reverser control lever arm fully engages plunger lock pin when reverser control lever is lifted.

I. Check that plunger stop pin does not contact either end of slotted hole in plunger throughout full range of plunger travel.

J. Position access panel on quadrant. Carefully tape brass drive screws in to secure panel to quadrant.

7. Install the thrust reverser control switch guard per steps 1 thru 3.

NOTE

Whenever the thrust reverser control switch and guard is reinstalled, the top mounting nut must be flush with end of the threaded portion of the switch or with a max of 1/2 thread showing above the top of the mounting nut.

1. Position the lower nut on switch shaft so that the top of the nut is approximately one half turn above the lower edge of the lever pivot hole on the switch bushing.
2. Place the lock washer on the bushing followed by the keying washer with the tab up, then place the switch through the hole in the panel and switch guard.
3. Tighten top mounting nut.

THRUST REVERSER AND SPEED BRAKE 80% RPM LIMIT SWITCH ADJUSTMENT (S/N 370-1, -7, S/N 380-1 AND SUBS)

Adjust the limit switches on airplanes equipped with Woodard fan speed synchronizer per steps 1 thru 12.

1. Center the trim assembly per steps A thru C.
   A. Apply electrical power to airplane.
   B. Push ENG SYNC circuit beaker.
   C. Set FAN SYNC switch to OFF.
2. Move power levers fwd until the engine fuel control arm indicates 41° (±1°) on quadrant.
3. Remove trim panels and guard for access to switches.
4. Loosen bolts (2) attaching cam to lever. (See Figure 76-00-2.)
5. Rotate cam until the switch actuator roller is at the top of cam ramp. (This is the highest portion of the cam.)
6. Check that switch is actuated.
7. Tighten cam attach bolts.
8. Set power lever to cut-off.
9. Check that switch is open (no continuity between common and normally open contact).
10. Check switch installation for security and safeties.
11. Replace guard and trim panels.
12. Perform operational check of thrust reverser and speed brake systems.

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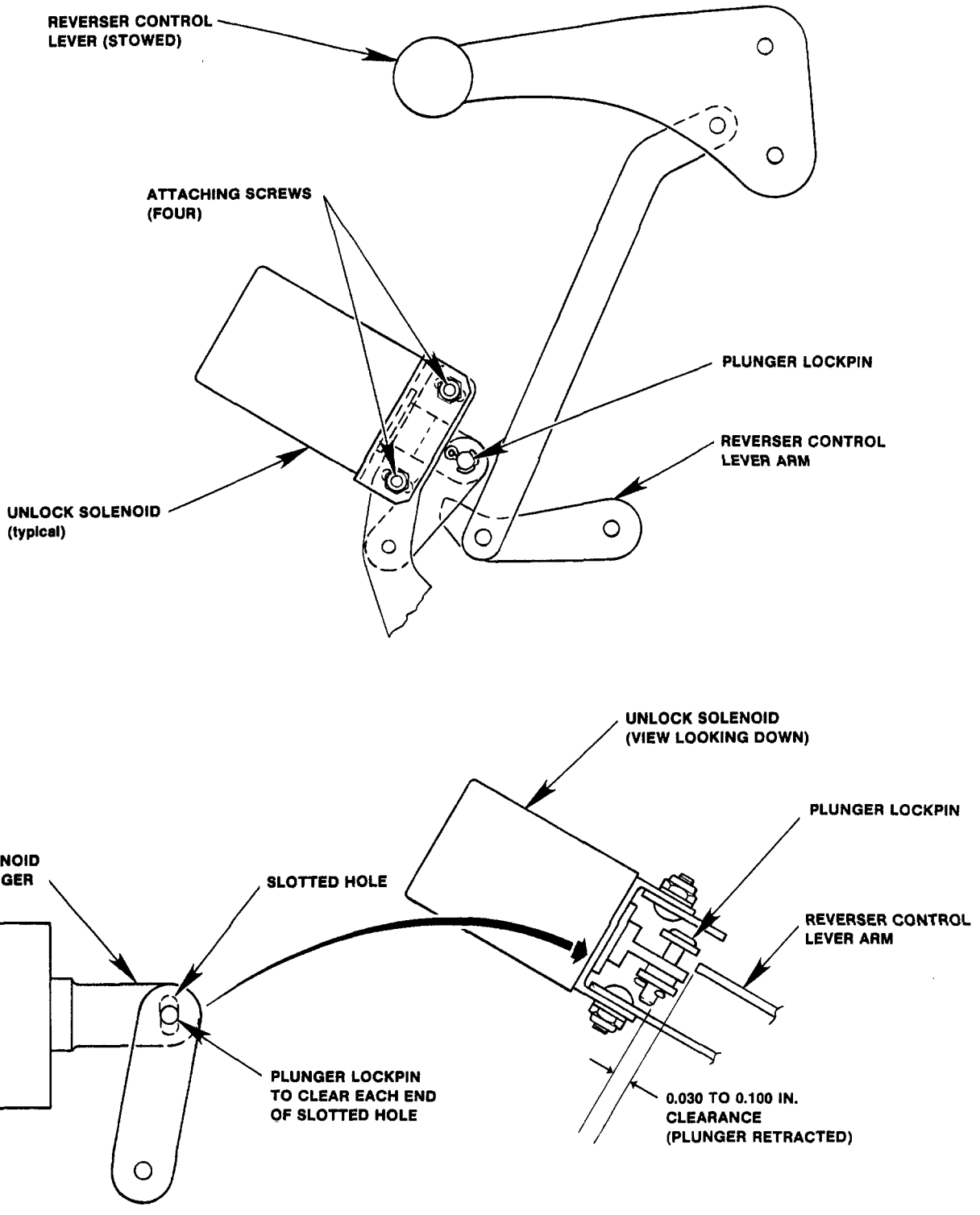

Adjusting Auxiliary Throttle Unlock Solenoid
Figure 76-00-6

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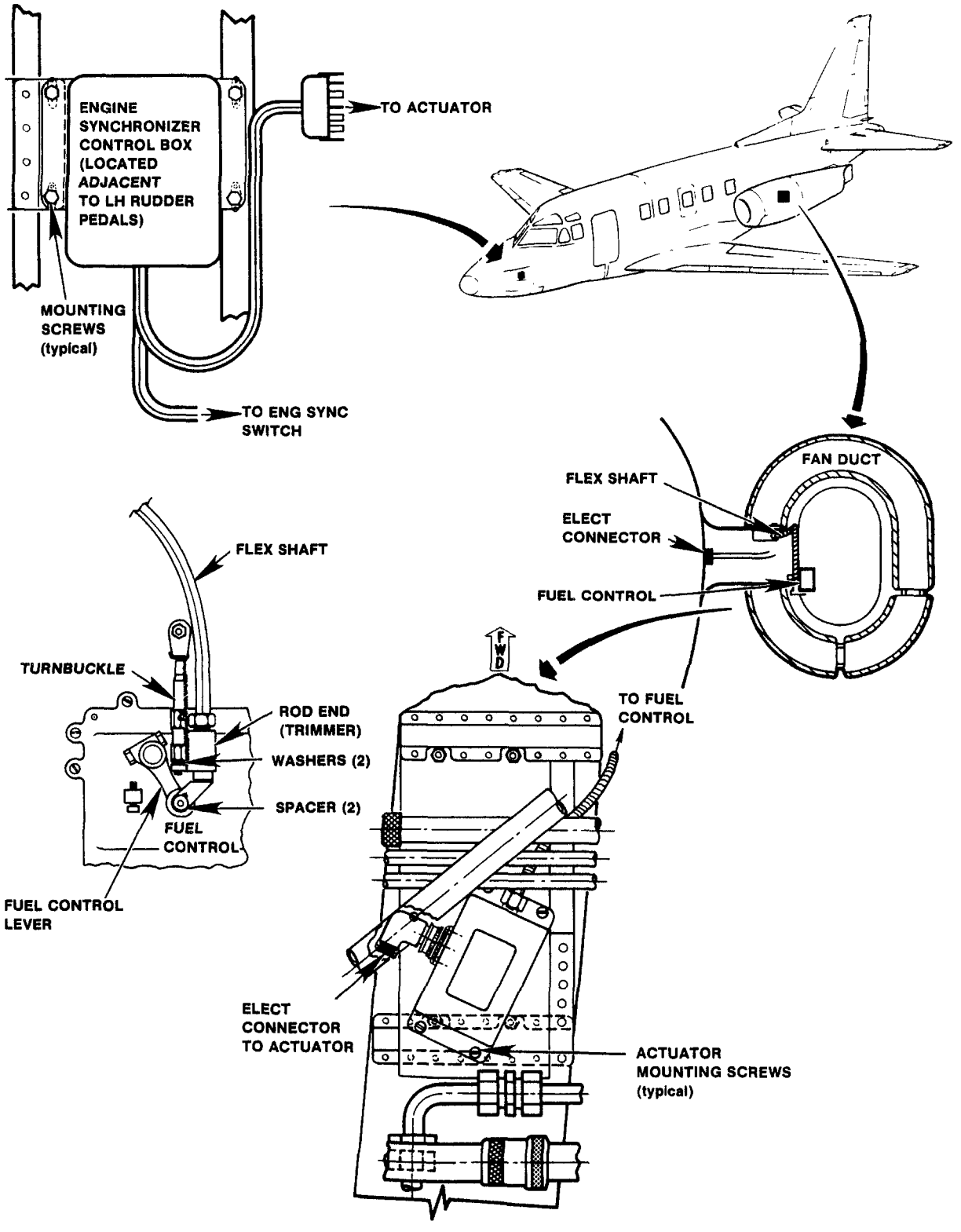

Fan Speed Synchronizer System Components
Figure 76-00-7 (Sheet 1 of 2)

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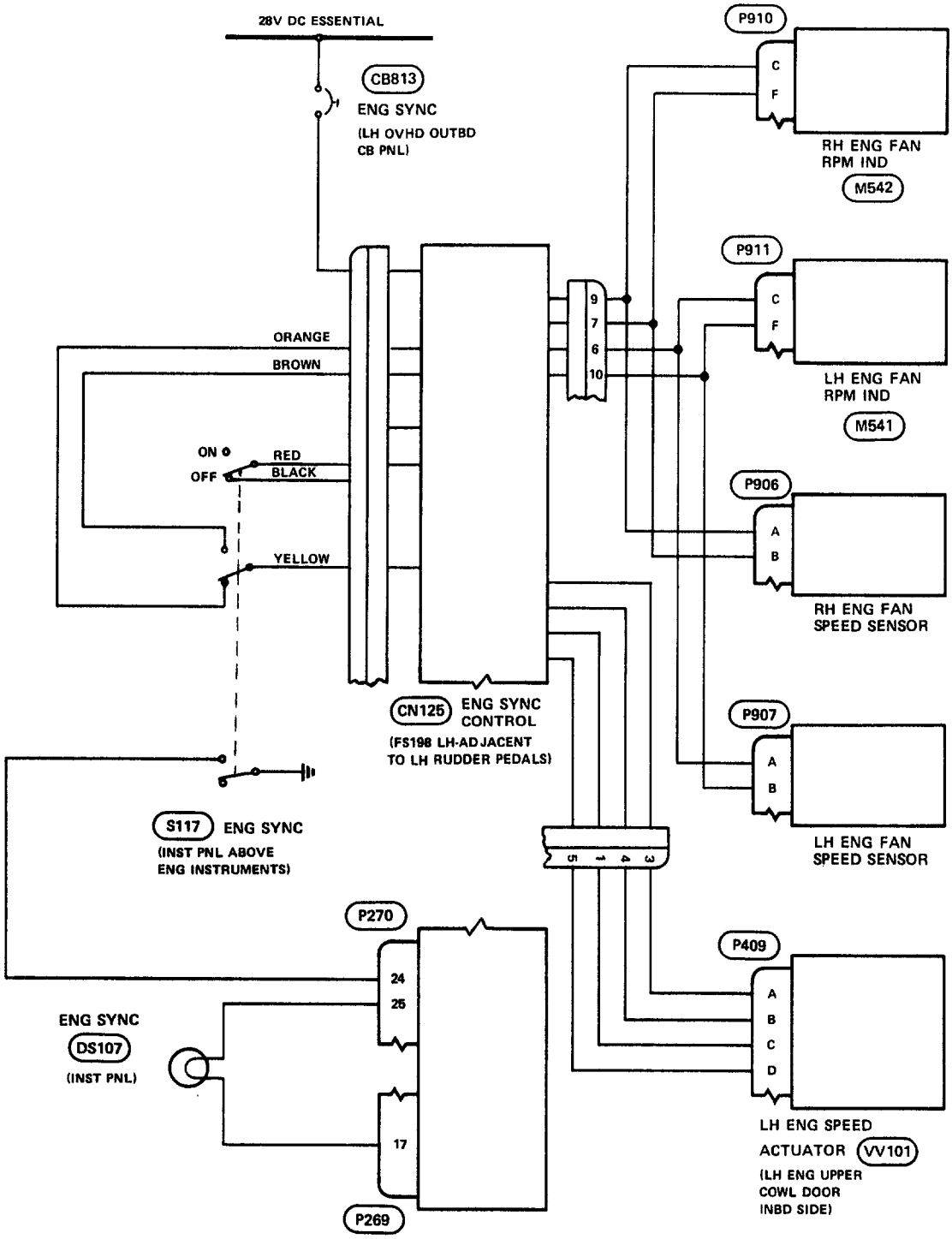

Fan Speed Synchronizer System Components
Figure 76-00-7 (Sheet 2 of 2)

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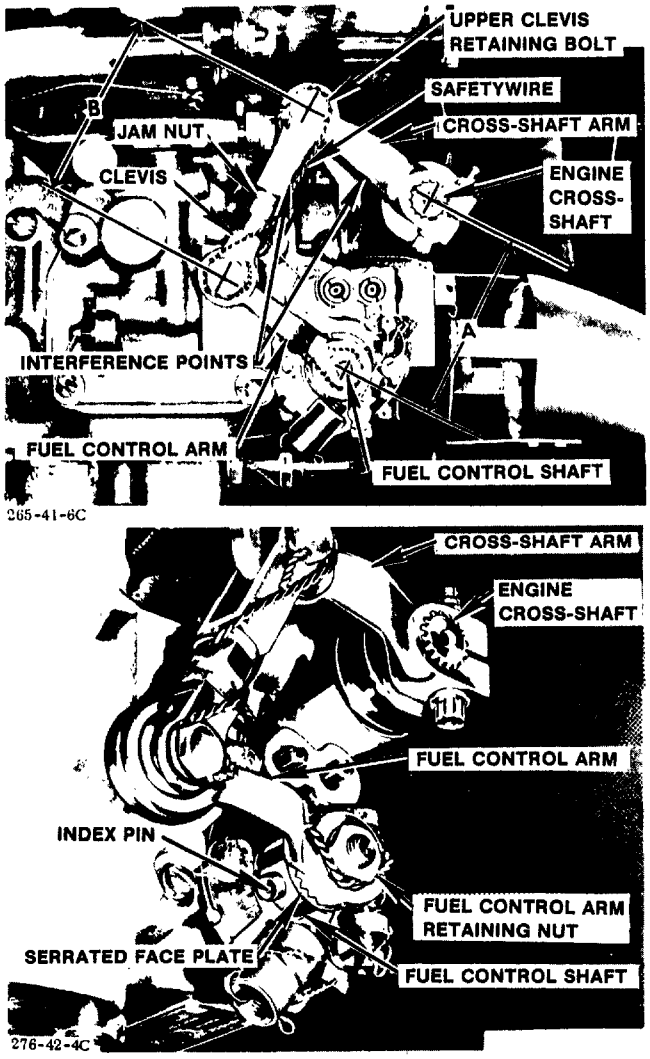

Adjusting Full Control Linkage
S/N 370-2 thru -6, -8 and -9
Figure 76-00-8

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ADJUSTMENT/TESTS (Cont'd)

QUADRANT THRUST REVERSER TEMPERATURE AND ELEVATION DECAL INSTALLATION (S/N 370-2 THRU -6, -8, AND -9)

1. Install decals per figures 76-00-4 and 76-00-9.
2. After replacement of either or both decals refer to Temperature and Elevation Thrust Reverser Stop Adjustment (S/N 370-2 Thru -6, -8, and -9).

TEMPERATURE AND ELEVATION THRUST REVERSER STOP ADJUSTMENT (S/N 370-2 THRU -6, -8 AND -9)

NOTES

- ***The following procedures must be performed if thrust reverser temperature and elevation decals are reinstalled or repositioned.***
- ***The engines must have engine inlet duct (with splitter vanes) installed, nacelle doors installed, no inlet screen, engine inlet duct anti-icing off, normal power extraction, cabin air on.***

1. Start engine(s).
2. Leave engine power levers at idle detent.
3. Position THRUST REVERSER control switch to on.
4. Raise the LH reverser lever to interlock stop.
5. After reverser has deployed increase engine rpm to approximately 45 to 50%.
6. Position THRUST REVERSER control switch to EMERG STOW and check that reverser is stowed.
7. Increase engine thrust (using reverser lever) until the determined engine pressure ratio (EPR) is attained. (See chapter 71.)
8. With the engine operating at the determined EPR, rotate the reverser lever adjustment knob until the stop bellcranks contacts the reverser lever stop. (See figure 76-00-4.)

NOTE

The intersection of the temperature and elevation lines should correspond to the ambient (field) conditions prevailing during this check.

9. Check position of temperature and elevation scales.
10. If required, adjust the position of the temperature scale decal by loosening the ramp adjustment screw (P/N NA500-A4-3) and moving the ramp to obtain the prevailing temperature and elevation relationship. (See figures 76-00-4 and 76-00-9.)
11. Tighten adjustment screw and safetywire.
12. Decrease LH engine thrust to idle by positioning the reverser levers to STOW (fwd and down) and position THRUST REVERSER control switch to ON.
13. Repeat steps 2 thru 12 operating the RH engine.

SYNCHRONIZER CONTROL BOX REMOVAL AND INSTALLATION (S/N 380-1 AND SUBS WITH WOODARD FAN SPEED SYNCHRONIZER)

NOTE

The synchronizer control box is located adjacent to the left hand rudder pedals.

1. Remove synchronizer control box per steps A thru D.
   A. Remove electrical power from airplane.
   B. Disconnect electrical plugs from control box electrical cables.
   C. Remove four mounting screws from control box.
   D. Remove control box from airplane.
2. Install synchronizer control box per steps A thru C.
   A. Position control box over mounting holes and install the four mounting screws.
   B. Connect electrical cable plugs to airplane wiring.
   C. Perform operational check of fan synchronizer system.

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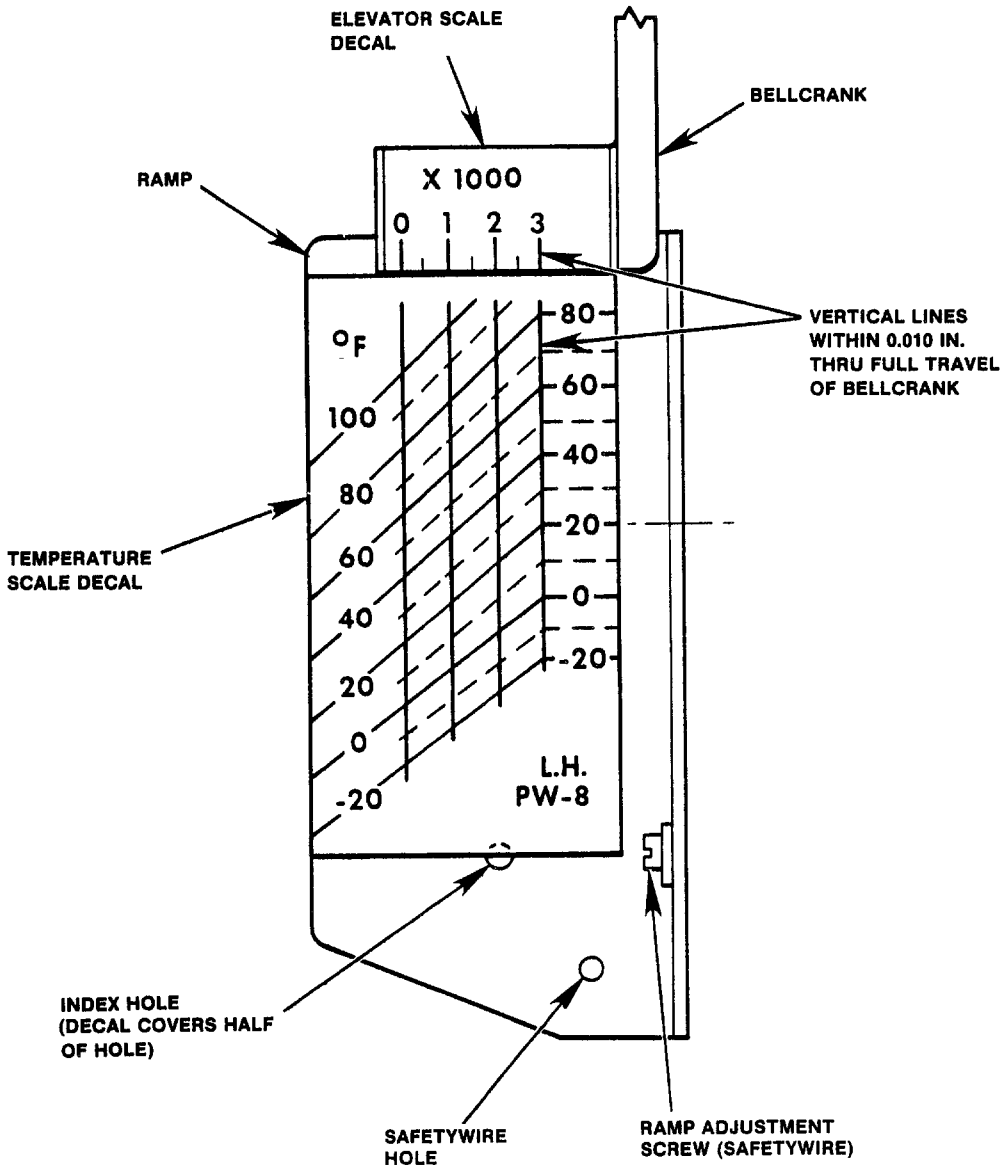

TYPICAL BOTH SIDES

Installing Quadrant Thrust Reverser Temperature and Elevation Decals
S/N 370-2 thru -6, -8 and -9
Figure 76-00-9

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ADJUSTMENT/TESTS (Cont'd)

FAN SYNCHRONIZER ACTUATOR REMOVAL AND INSTALLATION (S/N 380-1 AND SUBS WITH WOODARD FAN SPEED SYNCHRONIZER)

1. Remove fan synchronizer actuator per steps A thru E.
   A. Remove electrical power from airplane.
   B. Open left engine cowling.
   C. Disconnect flex shaft and electrical plug from actuator.
   D. Remove the four mounting screws securing actuator to fan duct.
   E. Remove actuator from airplane.
2. Install fan synchronizer actuator per steps A thru H.
   A. Position actuator on fan duct and install four securing screws.
   B. Connect electrical plug to actuator and safetywire.
   C. Position the FAN SYNC switch at OFF and apply electrical power to airplane. (This centers the actuator travel.)
   D. Remove electrical power from airplane.
   E. Turn the trimmer drive cable as required to center the trimmer assembly in its operating range.
   F. Connect flex shaft to actuator.
   G. Close engine cowling.
   H. Perform operational check of fan synchronizer system.

ROD (TRIMMER ASSEMBLY) REMOVAL AND INSTALLATION (S/N 380-1 AND SUBS WITH WOODARD FAN SPEED SYNCHRONIZER)

1. Remove rod end (trimmer assembly) per steps 1 thru 5.
   A. Open engine access doors.
   B. Remove flex cable from trimmer assembly.
   C. Remove bolts securing trimmer to fuel control arm and engine power lever arm. Retain spacing washers at fuel control arm for reinstallation.
   D. Remove trimmer from engine.
   E. Remove safetywire and unscrew turnbarrel from trimmer assembly. Retain washers for installation.
2. Install rod end (trimmer assembly) per steps A thru K.
   A. Install washers (2) on threaded end of turnbarrel and install turnbarrel in trimmer assembly. Torque turnbarrel to 40-50 in.-lbs.
   B. Install eye fitting in threaded portion of turnbarrel.
   C. Adjust threaded eye fitting to 5.12 in. (center to center of holes) between eye fitting and trimmer assembly attach point. Safetywire eye fitting to barrel and to trimmer assembly.

NOTE

The actuator may be centered by positioning the FAN SYNC switch at OFF and apply electrical power to airplane.

   D. Check that synchronizer actuator is centered within its operating range.
   E. Center the trimmer assembly within its operating range by turning the trimmer eye fitting.
   F. Install trimmer assembly on engine fuel control arm per steps (1) thru (3).
      (1) Install spacer washers.
      (2) Install nut and bolt.
      (3) Connect opposite end to power lever linkage.
   G. Check rigging of fuel control linkage.
   H. Inspect complete sysem for proper safetywire, cotter pin and jam nut installation.
   I. Check system for freedom of operation.
   J. Close engine cowling.
   K. Perform operational check of fan synchronizer system.

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ENGINE CONTROLS

INSPECTION/CHECK

FAN SYNCHRONIZER SYSTEM OPERATIONAL CHECK (NA 380 AIRPLANES WITH WOODARD FAN SPEED SYNCHRONIZER)

Operational Check

NOTE

Operational check of the fan synchronizer system consists of a max power/idle power and engine cutoff check. Both procedures are required to ensure that the engine will not shut down at idle power and that max power can be obtained.

1. Position airplane on run-up pad and place wheel chocks at the main landing gear wheels.
2. Set FAN SYNC switch to OFF and ENG SYNC circuit breaker is in.
3. Start both engines. (Refer to chapter 71.)
4. Advance both power levers to 65% rpm (gas generator speed). Reposition either power lever as required to obtain identical rpm (as close as possible) of both engines.
5. Set FAN SYNC switch to ON. Ensure the SYNC ON indicator light is on.
6. Slowly advance LH engine power lever to 70%. This allows the trimmer to adjust to its shortest length.
7. Advance LH engine power lever to max power. Ensure that max power is obtained, then pull the ENG SYNC circuit breaker.

NOTE

The synchronizer system will required rerigging and another operational checkout if max power was not obtained or the engine shut down at idle power.

8. Retard LH engine power lever to IDLE rpm. Engine should not shut down.
9. Push in ENG SYNC circuit breaker and set FAN SYNC switch to OFF.

Engine Cut-off Check

1. Manually synchronize RH and LH engines at 65% rpm.
2. Set FAN SYNC switch to ON.
3. Slowly retard LH engine rpm to 60%. This allows the trimmer to adjust to its longest length.

NOTE

The synchronizer system should be re-rigged and the cut-off check shall be performed if engine did not shut down.

4. Retard engine power lever to CUT-OFF. Ensure that engine shuts down.
5. Shut down RH engine.
6. Advance LH engine power lever approximately one in. and set FAN SYNC switch to OFF subsequent to engine rotation. This allows the trimmer and actuator to center itself within their operating range.

FAN SYNCHRONIZER SYSTEM FUNCTIONAL TEST (NA 380 AIRPLANES WITH WOODARD FAN SPEED SYNCHRONIZER)

Functional Test

1. Start RH and LH engines. (Refer to chapter 71.)
2. Advance engine power levers to 65% rpm (gas generator speed).
3. Synchronize engines manually as close as possible.
4. Monitor fan rpm indicators and set FAN SYNC switch to ON.
5. Monitor both fan rpm indicators. Rpm indications should be identical.
6. Advance RH engine power lever slightly. Fan rpm should remain unchanged.
7. Set FAN SYNC switch to OFF. Check for an out of sync fan condition.
8. Shut down engines and secure airplane as required.

MAJOR CHANGE

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CHAPTER

77

ENGINE INDICATING

SABRELINER

MAINTENANCE MANUAL

CHAPTER 77

ENGINE INDICATING

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ENGINE INDICATING

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*The asterisk indicates pages revised, added, or deleted by the current revision.

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ENGINE INDICATING - DESCRIPTION

1. General

A constant indication of the operating condition of the engines is provided by the engine indicating instruments. Individual engine speed, exhaust temperature, engine pressure ratio, rate of fuel flow and engine oil pressure and temperature are shown on gages or indicators mounted on the instrument panel in the cockpit. The engine indicating systems are shown schematically in figure 77-00-1.

2. Oil Pressure Indicators

The oil pressure indicators are hermetically sealed, synchro-type instruments. The indicators are marked to show engine oil pressure in pounds per square inch. The indicators are clamp-mounted and are equipped with integral lighting.

3. Oil Pressure Transmitters

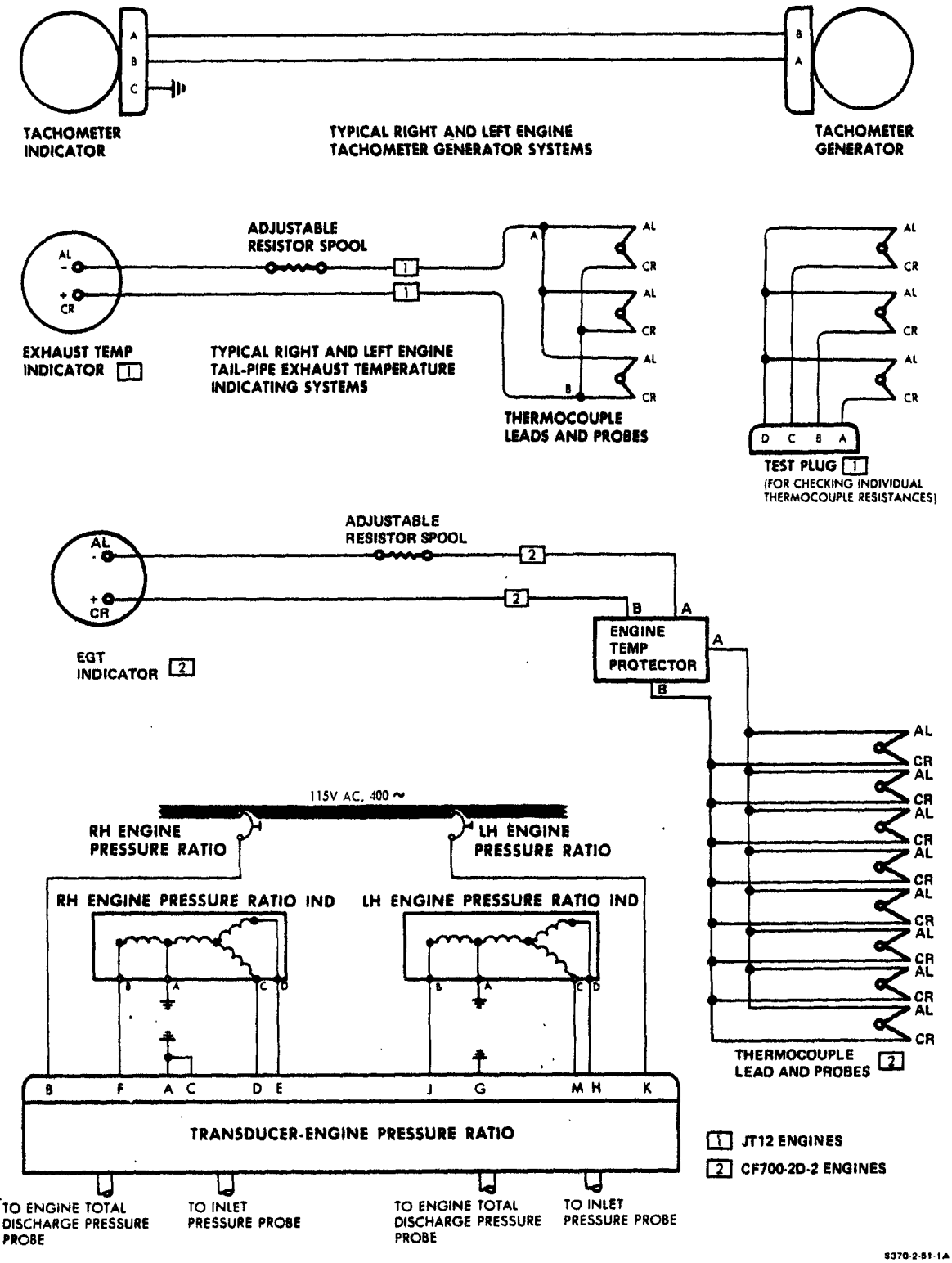
The oil pressure transmitters are hermetically sealed units and are installed on the engines. The transmitters are operated by engine oil pressure acting upon Bourdon tubes and mechanisms, positioning the rotor of the respective transmitter synchro motor. On NA-370-2 through -6, -8 and -9 airplanes, the transmitters may be reached through the lower engine access doors. On NA-370-1 and -7, NA-380-1 and later airplanes, access to the transmitter is through the engine upper access door.

4. Oil Low-pressure Warning Switch - Aircraft NA-370-2 Through -6, -8 and -9

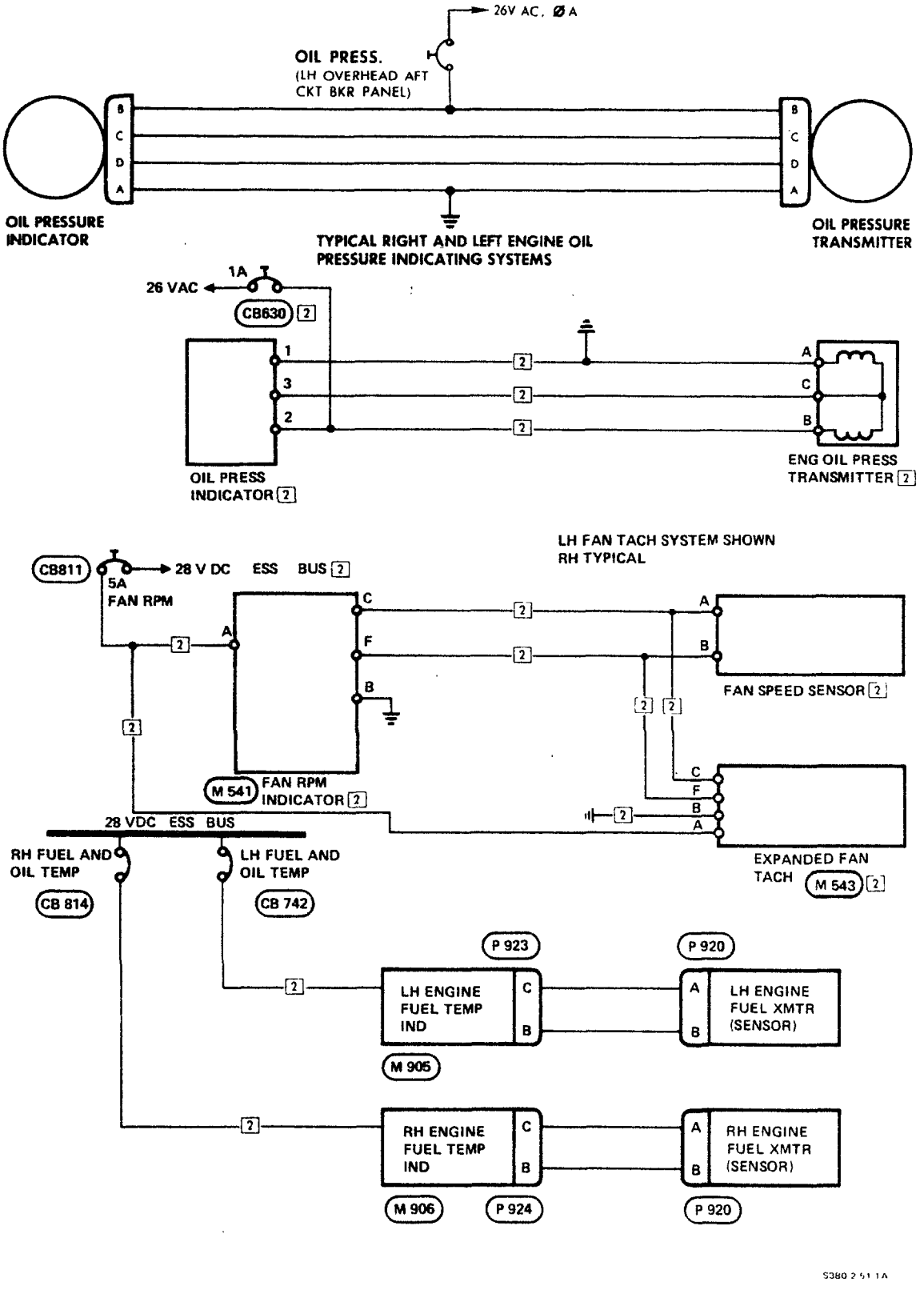
The oil low-pressure warning system has a switch, actuated by a pressure-sensitive diaphragm in a hermetically sealed container, mounted on the engine accessory drive gear case, and a caution light on the caution-warning light panel in the cockpit. During a start, as engine oil pressure increases, contacts within the switch open before 32 psi pressure is reached. If oil pressure should decrease, these contacts will close at 28 ($\pm$1) psi, lighting the "OIL PRESS LOW" caution light. No attempt should be made to repair a faulty unit. The unit must be replaced.

4A. Engine Oil Low-pressure Warning Switch - Aircraft NA-370-1, -7, NA-380-1 and Later

The engine oil low-pressure warning system has two switches, one mounted on each engine and a LH and RH engine oil pressure low caution light on the caution and warning light panel in the cockpit. Each switch is activated by a pressure-sensitive diaphragm in a hermetically sealed container. During starts, as engine oil pressure increases, contacts within the switch open before 8 psi pressure is reached. If oil pressure should decrease, these contacts will close at 5 ($\pm$1/2) psi lighting the appropriate engine oil pressure low caution light. No attempt should be made to repair a faulty unit. The unit must be replaced.


TACHOMETER INDICATOR
TYPICAL RIGHT AND LEFT ENGINE TACHOMETER GENERATOR SYSTEMS
TACHOMETER GENERATOR
ADJUSTABLE RESISTOR SPOOL
EXHAUST TEMP INDICATOR 1
TYPICAL RIGHT AND LEFT ENGINE TAIL-PIPE EXHAUST TEMPERATURE INDICATING SYSTEMS
THERMOCOUPLE LEADS AND PROBES
TEST PLUG 1
(FOR CHECKING INDIVIDUAL THERMOCOUPLE RESISTANCES)
ADJUSTABLE RESISTOR SPOOL
EGT INDICATOR 2
ENGINE TEMP PROTECTOR
THERMOCOUPLE LEAD AND PROBES 2
115V AC, 400 ~
RH ENGINE PRESSURE RATIO
LH ENGINE PRESSURE RATIO
RH ENGINE PRESSURE RATIO IND
LH ENGINE PRESSURE RATIO IND
TRANSDUCER-ENGINE PRESSURE RATIO
TO ENGINE TOTAL DISCHARGE PRESSURE PROBE
TO INLET PRESSURE PROBE
TO ENGINE TOTAL DISCHARGE PRESSURE PROBE
TO INLET PRESSURE PROBE
1 JT12 ENGINES
2 CF700-2D-2 ENGINES
5370-2-51-1A

Engine Indicating Systems - Electrical Schematic
Figure 77-00-1 (Sheet 1 of 2)


26V AC, Ø A
OIL PRESS.
(LH OVERHEAD AFT
CKT BKR PANEL)
B
C
D
A
OIL PRESSURE
INDICATOR
TYPICAL RIGHT AND LEFT ENGINE OIL
PRESSURE INDICATING SYSTEMS
OIL PRESSURE
TRANSMITTER
1A
26 VAC
CB630 2
1
3
2
OIL PRESS
INDICATOR 2
ENG OIL PRESS
TRANSMITTER 2
LH FAN TACH SYSTEM SHOWN
RH TYPICAL
CB811
28 V DC ESS BUS 2
5A
FAN RPM
FAN SPEED SENSOR 2
M 541
FAN RPM
INDICATOR 2
28 VDC ESS BUS
RH FUEL AND
OIL TEMP
CB 814
LH FUEL AND
OIL TEMP
CB 742
EXPANDED FAN
TACH M 543 2
P 923
P 920
LH ENGINE
FUEL TEMP
IND
LH ENGINE
FUEL XMTR
(SENSOR)
M 905
RH ENGINE
FUEL TEMP
IND
RH ENGINE
FUEL XMTR
(SENSOR)
M 906
P 924
P 920

S380 2 51 1A

5. Fuel Flow Indicating System

For information concerning the fuel flow indicating system, refer to Chapter 28.

6. Fuel Temperature Indicators - Aircraft NA-380-1 and Later

On NA-380-1 and later airplanes, fuel temperature indicators for the left and right engines are installed in the center instrument panel. The fuel temperature indicators are hermetically sealed, syncro-type instruments marked to show fuel temperatures in degrees centigrade. The indicators are clamp mounted and are equipped with integral lighting.

7. Fuel Temperature Sensing Units - Aircraft NA-380-1 and Later

On NA-380-1 and later airplanes, a fuel temperature sensing device is installed in a boss, welded to the main fuel inlet line on the upper forward portion of the engine. Access to the fuel temperature sensor is through the engine upper access door.

ENGINE INDICATING - TROUBLE SHOOTING

1. Trouble Shooting Engine Oil Pressure System

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. LOW OR FLUCTUATING OIL PRESSURE.		
(1) Defective oil pressure transmitter and/or indicator.	Replace cockpit instrument with indicator of known accuracy and recheck pressure.	Replace instrument.
B. EXCESSIVE OIL PRESSURE.		
(1) Defective oil pressure transmitter and/or indicator.	Replace cockpit instrument with indicator of known accuracy and recheck pressure.	Replace instrument.
C. EXCESSIVE OIL CONSUMPTION.		
(1) Defective oil pressure and scavenge lines or connections.	Visually inspect all oil lines and connections for evidence of leaks.	Replace tubing as necessary.

2. Trouble Shooting Engine Fuel Temperature System - Aircraft NA-370-1, -7 and NA-380-1 and Later

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. LOW OR NO FUEL TEMPERATURE INDICATION.		
(1) Defective indicator and/or sensor.	Switch instruments in panel. If problem still exists, check resistance of sensor. Resistance is 97 to 101 ohms at 20° to 30° centigrade.	Replace instrument and/or sensor.

ENGINE INDICATING - MAINTENANCE PRACTICES

1. Oil Pressure/Fuel Temperature Indicators

For information on removing or installing oil pressure or fuel temperature indicators, refer to Chapter 31.

2. Removing Oil Pressure Transmitter - Aircraft NA-370-2 Through -6, -8 and -9

A. To remove oil pressure transmitter, proceed as follows:

(1) Remove lockwire and disconnect electrical lead at transmitter.

(2) Disconnect vent line from fitting at transmitter.

(3) Remove lockwire and remove transmitter from accessory drive gear case.

(4) Install plug in gear case.

NOTE: If the transmitter is not to be reinstalled, the filter must be removed for cleaning and installation in a new transmitter.

3. Installing Oil Pressure Transmitter - Aircraft NA-370-2 Through -6, -8 and -9

A. Install oil pressure transmitter as follows:

(1) Clean filter assembly if removed from transmitter.

(2) Install "O" ring on filter assembly and lubricate "O" ring lightly with engine oil.

(3) Fill oil pressure transmitter with engine oil through oil pressure port; then install and tighten filter assembly in oil pressure port of oil pressure transmitter.

(4) Remove gear case plug.

(5) Install "O" ring on transmitter and install transmitter in gear case.

(6) Tighten transmitter with a 1-1/2-inch, open-end, thin-wall, crow's-foot wrench to a torque value of 400 to 500 inch-pounds, and safety with lockwire.

(7) Install "O" ring on union and install union in vent port in transmitter; or install elbow fitting on vent port with "O" ring, nylon split ring, and jam nut (leave jam nut loose).

(8) Connect transmitter vent line to union or elbow. If elbow is used, tighten jam nut on transmitter vent port.

(9) Connect and lockwire electric lead to transmitter.

4. Removing Oil Pressure Transmitter - Aircraft NA-370-1, -7, NA-380-1 and Later

The oil pressure transmitters for NA-370-1, -7, NA-380-1 and later airplanes are located approximately on the upper centerline of the engine just aft of the forward flange of the engine. Access to the oil pressure transmitter is through the engine upper access door.

A. To remove the oil pressure transmitters, proceed as follows:

(1) Open engine upper access door.

(2) Remove lockwire and disconnect electrical plug from transmitter.

(3) Disconnect flexible lines (2) from each end of transmitter. Cap lines and fittings to prevent contamination.

(4) Remove tee fitting from end of transmitter extending through support bracket.

(5) Remove lockwire from nut securing transmitter to support bracket.

(6) Remove nut security transmitter to support bracket; then withdraw transmitter from support bracket.

5. Installing Oil Pressure Transmitter - Aircraft NA-370-1, -7, NA-380-1 and Later

A. To install oil pressure transmitter, proceed as follows:

(1) Install oil pressure transmitter through support bracket on engine and install nut securing transmitter to support bracket. Tighten nut finger tight.

(2) Install tee fitting in end of transmitter extending through support bracket.

(3) Install flexible lines (2 on each end) to fittings on ends of transmitter.

(4) Connect electrical plug to receptacle on transmitter and safety with lockwire.

(5) Tighten nut securing transmitter to support bracket and safety with lockwire.

6. Removing Oil Low-pressure Warning Switch - Aircraft NA-370-2 Through -6, -8 and -9

A. To remove oil low-pressure warning switch, proceed as follows:

(1) Remove lockwire and disconnect electrical lead at low-pressure warning switch.

(2) Remove lockwire and remove low-pressure warning switch from gear case.

(3) Cover or plug open port.

7. Installing Oil Low-pressure Warning Switch - Aircraft NA-370-2 Through -6, -8 and -9

A. To install oil low-pressure warning switch, proceed as follows:

(1) Remove lockwire and gear case plug.

(2) Install "O" ring on oil low-pressure warning switch and install switch in gear case.

(3) Tighten switch with thin double-end open-end wrench, and safety with lockwire.

8. Removing Oil Low-pressure Warning Switch - Aircraft NA-370-1, -7, NA-380-1 and Later

The oil low-pressure warning switch is located on the engine upper centerline adjacent to the forward lifting lug. Access to the oil low-pressure warning switch is through the engine upper access door.

A. To remove oil low-pressure warning switch, proceed as follows:

(1) Open engine upper access door.

(2) Disconnect electrical plug from aft end of switch.

(3) Remove flexible lines (2) from fittings on forward end of switch.

(4) Remove fittings from switch bosses extending through support bracket.

(5) Remove lockwire from four screws securing switch to support bracket.

(6) Remove the four switch attachment screws, then remove switch from engine.

9. Installing Oil Low-pressure Warning Switch - Aircraft NA-370-1, -7, NA-380-1 and Later

A. To install engine oil low-pressure warning switch, proceed as follows:

(1) Position engine oil low-pressure switch in place in support bracket and install the four attachment screws. Safety screws with lockwire.

(2) Install fittings in bosses at forward end of switch.

(3) Connect flexible lines (2) to fittings at forward end of switch.

(4) Connect electrical plug to receptacle on aft end of switch.

(5) Close engine upper door.

POWER

DESCRIPTION

ENGINE PRESSURE RATIO SYSTEM

The engine pressure ratio indicating system gives the pressure ratio between inlet pressure (ambient pressure) and total engine discharge pressure. This provides the pilot with a check on the thrust available for takeoff and allows him to operate the airplane at the proper power setting to obtain the maximum range while cruising. The system consists of pressure ratio gauges, pressure ratio transducers, an engine total discharge pressure probe, and an engine inlet pressure, pitot tube.

ENGINE PRESSURE RATIO TRANSDUCER

The transducer transmits power to the engine pressure ratio gauges to actuate their pointers. In the transducer, ambient ram-air pressure and engine exhaust pressure are applied to two metallic bellows mounted at 90 degrees to each other. When the beam in the transducer is balanced on the traveling fulcrum, no contact is made at either point of the beam displacement sensor. As changes occur in either intake pressure or exhaust pressure, the corresponding bellows transmit the pressure change to unbalance the beam on the fulcrum. The resultant movement of the balance beam closes the related contact of the beam displacement sensor, actuating the motor control relay. The movement of the motor, through gearing, turns the synchro to send this new position to the slave synchro operating the gauge pointer.

ENGINE TOTAL DISCHARGE PRESSURE PROBE

The total discharge pressure of the engine is measured by probes [1] extending into the exhaust gas flow of the engine. Each probe has a series of small calibrated holes through which the exhaust gas enters. The probes are manifolded together to deliver an average pressure of all probes. Since the pressure at each hole is different, should one or more of the holes become clogged or enlarged the average pressure would not be an accurate measure of the total discharge pressure, and the pressure ratio gauges in the cockpit would give a false reading.

ENGINE INLET PRESSURE PITOT TUBE

The ambient pressure is obtained through two pitot tubes, one for each engine. The pitot tubes are on the vertical stabilizer leading edge. Pressure is routed directly from the pitot tube pickup to the engine ratio transducers. Power to heat the pitot pickup is supplied from the 28 VDC essential bus and is controlled by the PITOT & STALL WNG switch on the anti-ice control panel.

[1] Single probe on S/N 380-1 and Subs.

TACHOMETER SYSTEM

Two tachometers are mounted on the pilot's inboard instrument panel. The respective right or left engine speed registered on each tachometer is the percentage of the approximate maximum rpm (16,000 for JT12 engines or 16,700 for CF-700-2D-2 engines) of the compressor rotor speed. Each tachometer is operated by electrical power generated by its respective tachometer generator, geared to the engine accessory section, and driven by the compressor rotor. The AC voltage generated by the tachometer generators is transmitted to the tachometer indicators, at a frequency proportional to engine speed. The frequency-sensitive tachometer indicators respond to this frequency and voltage to indicate engine rpm.

TACHOMETER INDICATORS

The tachometer indicators are magnetic drag-type motors. Two dials and pointers on each instrument show engine rpm as a percent of maximum rpm. The large dial and pointer show from zero to 100% rpm in increments of 2%. The small dial is numbered from 1 to 10; the pointer makes one revolution around the dial for each 10% rpm change shown by the main pointer. Read together, therefore, the two dials show from zero to 110% rpm in increments of 1%. The tachometer indicators are frequency-sensitive, and are secured to the instrument panel by clamp-type mountings. The indicators are equipped with integral lighting.

TACHOMETER GENERATORS

The tachometer generators are small, two-pole AC generators and are installed on the engines. The frequency output of the generators is proportional to engine speed. The tachometer generators may be reached by removing the respective engine lower access doors.

FAN SPEED SENSOR (S/N 370-1, -7, S/N 380-1 AND SUBS)

The fan speed sensing unit is mounted to the aft fan rear frame in the bearing housing. The sensing unit is made up of a coil-wound permanent magnet and a pole-piece mounted perpendicular to a sixty-tooth gear type ring attached to fan rear rotor shaft. The sensing unit is positioned between 0.005 and 0.008 from the peaks of the ring gear. When the fan rotates, the gap between the ring gear and the sensing unit varies as the crests and valleys of the ring gear alternately pass across the end of

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DESCRIPTION (Continued)

FAN SPEED SENSOR (S/N-370-1, -7, S/N 380-1 AND SUBS)

the pole-piece creating an alternating current. The alternating current is trans-mitted to the cockpit indicators through electrical leads from the sensing unit, through the bottom strut of the fan rear frame.

FAN SPEED INDICATORS

Four engine fan rpm indicators, two for each engine, are mounted on the pilot's inboard instrument panel. One indicator for each engine is calibrated in 5% increments from 90 to 115%. To provide easier reading of the fan rpm at the upper end of the scale, the other indicators are calibrated in 1% increments from 90 to 115%. The engine fan rpm indicators are powered by the 28 VDC essential bus.

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POWER - TROUBLE SHOOTING

1. Trouble Shooting Tachometer System

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. INCORRECT OR NO INDICATIONS ON TACHOMETER SYSTEMS.		
(1) Wiring between generators and tachometers faulty or improperly connected.	Make continuity check. Check for proper connections.	Replace faulty wiring or make proper connections.
(2) Faulty tachometer generator.	Make continuity check.	Replace faulty generator.
(3) Faulty tachometer indicator.	Make continuity check, or interchange right and left engine indicators.	Replace faulty indicator.

2. Trouble Shooting Engine Pressure Ratio System

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. NO READING ON GAGE.		
(1) Open PRESS RATIO circuit breaker or damaged circuit breaker.	Check that PRESS RATIO circuit breaker (LH or RH) is pushed in. Check for 115 volts ac at both terminals of circuit breaker.	If circuit breaker is pushed in and voltage is present at terminal connected to power source but not at the other terminal, replace damaged circuit breaker.
(2) Disconnected pressure ratio indicator or broken electrical leads.	Check connections and wires between transducer and pressure ratio indicator.	Connect indicator to transducer; or restore connections and wire leads, as required.
(3) Defective transducer unit.	Check for 26 volts ac at terminal F or J (depending on subsystem being checked).	If there is no voltage present at test point, replace transducer.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
(4) Defective pressure ratio indicator.	Check continuity between terminals A and B, A and C, and A and D of indicator.	Replace indicator if electrical continuity check reveals zero or infinite values of resistance.
(5) Loose or broken connection on pressure or static lines.	Perform leak test on pressure and static lines. (Refer to 77-10-3.)	Tighten connections or replace broken lines.
(6) Clogged EPR probe.	Inspect probe for accumulated deposits and dirt.	Clean probe, being careful not to enlarge calibrated holes.
(7) Ice in lines or pressure pitot tubes.	Ensure that lines are free from accumulated moisture and that pitot heating system is operating.	Drain accumulated moisture from lines by removing water drain traps.
B. ERRATIC OR INCORRECT READINGS ON GAGE.		
(1) Loose or broken connections on static or pressure lines.	Check system for leakage. (Refer to 77-10-3.)	Tighten connections or replace broken lines.
(2) Clogged EPR probe.	Inspect EPR probe for obstructions in holes.	Remove obstructions, being careful not to enlarge calibrated holes.
(3) Ice or other obstruction in pitot tubes and/or lines.	Check pitot tubes and lines for ice, accumulated moisture, and other obstructions.	Remove ice or other obstructions from pitot tubes. Drain lines by removing water drain traps.
(4) Defective transducer transmitter and/or pressure ratio indicator.	Check transducer and indicator, and perform scale and friction error test detailed in 77-10-3.	Replace defective transducer and/or pressure ratio indicator.

POWER - MAINTENANCE PRACTICES

1. Removing and Installing Tachometer Indicators

For information on removing or installing tachometer indicators, refer to Chapter 31.

2. Removing Tachometer Generator - JT-12 Engines

A. The tachometer generators are installed on the upper left side of each engine accessory drive gear case and may be reached through the engine access doors.

(1) Disconnect electrical plug from tachometer generator.

(2) Remove nuts from mount base and lift tachometer generator from engine.

3. Installing Tachometer Generator - JT-12 Engines

A. To install tachometer generator, proceed as follows:

(1) Align tachometer generator shaft drive with engine mounting pad. Install and torque mounting nuts to 70-90 inch-pounds; then safety with lockwire.

(2) Connect electrical plug; then safety with lockwire.

4. Removing Tachometer Generator - CF700-2D-2 Engines

A. The tachometer generators are installed on the rear of the lubrication pump that extends through the oil tank. Access to the generators are reached through the engine access doors.

(1) Disconnect electrical plug from tachometer generator.

(2) Remove nuts from mount base and remove generator from engine.

5. Installing Tachometer Generator - CF700-2D-2 Engines

A. To install tachometer generator, proceed as follows:

(1) Align tachometer generator drive shaft with engine spline. Install and torque mounting nuts to 70-90 inch-pounds.

(2) Connect electrical plug to generator and safety with lockwire.

(3) Close engine access doors. Perform operational check-out of tachometer generator system.

1. Tachometer Indicating System Check

A. To check tachometer indicating system, proceed as follows:

(1) Check indicators for security of mounting, cleanness and tightness of glass, and general condition of dial markings.

(2) Check tachometer generator for security of mounting, external evidence of damage, and tightness of electrical mountings.

(3) Check that zero indication is within ±1.25% rpm. Tap indicators lightly before making this check.

(4) Check that indicators operate smoothly when engines are operating.

2. Tachometer Generator Resistance and Continuity Check

A. To test resistance and continuity of tachometer generators, proceed as follows:

(1) Open lower engine access doors and disconnect electrical plugs from tachometer generators.

(2) With multimeter, check resistance and continuity between pins, and between pins and case of tachometer generator. There should be continuity, and resistance values should be within one ohm of each other.

3. Checking Engine Pressure Ratio System for Leakage

A. To perform a leakage test on engine pressure ratio system lines, obtain a pitot-static test set and proceed as follows:

(1) Connect test set pressure source to engine exhaust pressure line with line connected to pressure ratio transducer.

(2) With static pressure at pitot tube, apply pressure equivalent to approximately 450 knots, or a pressure ratio reading of 1.4; then seal off pressure. The pressure in the line should not decrease at a rate exceeding 5 knots, or a pressure ratio reading of 0.01 per minute.

(3) Connect test set vacuum source to pitot tube line at vertical stabilizer, with line connected to pressure ratio transducer.

(4) With static pressure at EPR line, apply a vacuum equivalent to an altitude of approximately 10,000 feet, or a pressure ratio reading of 1.4; then seal off vacuum. The vacuum in the system should not decrease at a rate exceeding 400 feet, or a pressure ratio reading of 0.01 per minute.

4. EPR Transducer Scale Error and Friction Error Test

A. Check engine pressure ratio system for transducer scale and friction errors as follows:

(1) Check transducer for damaged cable adapter, cable assembly, and tube unions. Examine plug connector for cracks and damaged pins. Check EPR system cables and connections for any sign of damage.

(2) Check pressure ratio gage rotor winding for a resistance value of 15 (±1.5) ohms (resistance measured between terminals A and B of each pressure ratio indicator).

(3) Using a resistance bridge, check for 6.5 (±0.65) ohms across stator windings of pressure ratio indicators. The resistance value, as measured between terminals A and C, then A and D, of each indicator, should be 13.0 (±1.3) ohms.

(4) Check each pressure ratio indicator for an accuracy of 0.50 circular degrees, maximum, against a primary transmitting synchro with an electrical zero of 0.10 circular degrees, maximum.

(5) Connect engine pressure ratio transducer to a power supply capable of delivering 115 (±2) volts ac at 400 (±8) cps with less and 2 percent harmonics. Apply electrical power (at terminals B and K) to transducer.

(6) Measure transducer power output at terminals F and J; it should be 26 (±0.5) volts ac.

(7) Connect pressure ratio indicators to transducer and check for scale error and friction error as follows:

NOTE: Check each subsystem (for left and right engines) separately.

(a) Connect a test pressure source to transducer "ENGINE INLET" port, to apply ram-air pressure to engine pressure ratio system. Applied pressures between 2.0 and 29.999 inches of mercury, absolute, must be measured to an accuracy of ±0.010 inches of mercury, absolute; applied pressures above 30 inches of mercury, absolute, must be measured to an accuracy of ±0.030 inches of mercury, absolute.

(b) Connect a test pressure source to transducer "EXHAUST" port, to apply engine exhaust pressure to system. [(Refer to step (a) for applied test pressure tolerances.]

(c) Applied electrical power (115 volts ac, 400 cps) to applicable transducer terminal (B or K). Wait 3 minutes before proceeding with next step.

(d) Apply ram air and EPR values to transducer (as given in the following table). Check indicator readings for corresponding

values of ram air and EPR (in pairs) while gently tapping transducer. Indicator readings should match the ram air/EPR ratio, as also shown in the following applicable tables.

CAUTION: UNDER NO CIRCUMSTANCES SHOULD THE EPR VALUE BE LESS THAN THE PAIRED RAM AIR VALUE. ALSO, DO NOT DRIVE TRANSDUCER ABOVE A PRESSURE RATIO OF 3.5.

NOTE: In the following applicable tables, the tolerances listed in the "PRESSURE RATIO SCALE ERROR" column indicate the allowable scale error (observed when the transducer is gently tapped); the tolerances in the "PRESSURE RATIO FRICTION ERROR" column indicate allowable friction error (difference between before and after tapping transducer).

Airplanes NA-370-2 Through -6, -8 and -9

TEST PRESSURE (IN. Hg ABSOLUTE)		PRESSURE RATIO SCALE ERROR	PRESSURE RATIO FRICTION ERROR
RAM AIR	EPR		
6.000	7.200	1.200 (±0.025)	0.010
6.000	10.800	1.800 (±0.020)	0.010
6.000	13.800	2.300 (±0.020)	0.010
6.000	16.200	2.700 (±0.020)	0.010
6.000	20.400	3.400 (±0.025)	0.010
11.000	19.800	1.800 (±0.020)	0.010
11.000	25.300	2.300 (±0.020)	0.010
11.000	29.700	2.700 (±0.020)	0.010
25.000	45.000	1.800 (±0.020)	0.010
25.000	55.000	2.200 (±0.020)	0.010
25.000	67.500	2.700 (±0.020)	0.010
35.000	63.000	1.800 (±0.020)	0.010
35.000	70.000	2.000 (±0.020)	0.010

Airplanes NA-370-1, -7, and NA-380-1 and Later

TEST PRESSURE (IN. Hg ABSOLUTE)		PRESSURE RATIO SCALE ERROR	PRESSURE RATIO FRICTION ERROR
RAM AIR	EPR		
7.500	9.000	1.200 (±0.020)	0.010
7.500	12.000	1.600 (±0.020)	0.010
7.500	15.000	2.000 (±0.020)	0.010
15.000	27.000	1.800 (±0.020)	0.010
15.000	33.000	2.200 (±0.020)	0.010
25.000	40.000	1.600 (±0.015)	0.010
25.000	45.000	1.800 (±0.015)	0.010
25.000	50.000	2.000 (±0.015)	0.010
25.000	55.000	2.200 (±0.015)	0.010
35.000	42.000	1.200 (±0.020)	0.010
35.000	56.000	1.600 (±0.020)	0.010

OIL TEMPERATURE SYSTEM - DESCRIPTION AND OPERATION

1. General

On JT-12 engines, the temperature of the engine lubricating oil supply is maintained and controlled by the use of the fuel oil-cooler. The cooler consists of a housing containing a heat exchanger core and a temperature regulating and pressure relief valve. The fuel oil-cooler is in the return line, between the oil scavenge pumps and the oil tank. Oil temperature is reduced by circulating oil around the heat exchanger core tubes, through which fuel is flowing. The temperature regulating and pressure relief valve position determines the amount of return oil that circulates through the heat exchanger.

On CF700-2D-2 engines, the oil cooler is a liquid-to-liquid heat exchanger mounted on the oil reservoir mounting flange and against the right-hand side of the oil reservoir. It consists of numerous passages arranged in a honeycomb pattern. Both fuel and oil flow simultaneously through adjoining passages and an exchange of heat occurs between hot engine oil and cold fuel. Oil enters and leaves the cooler through ports located in the housing for the pressure bypass valve. When the pressure drop across the valve exceeds 16 to 24 psi due to clogging, or low temperature, high viscosity oil, the valve will open and the oil will bypass the heat exchanging element.

OIL TEMPERATURE SYSTEM - TROUBLE SHOOTING

1. Trouble Shooting Oil Temperature System

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. EXCESSIVE OIL TEMPERATURE.		
(1) Engine bearing failure.	Check oil strainer for indication of bearing failure.	Replace engine.
(2) Clogged oil strainer.	Check for excessive carbon or dirt in strainer.	Clean oil strainer.
(3) On JT-12 engines, faulty oil temperature regulator valve.	Substitute oil temperature regulator valve with serviceable unit.	Replace oil temperature regulator valve.

EXHAUST TEMPERATURE INDICATING SYSTEMS - DESCRIPTION AND OPERATION

1. General

Two engine exhaust temperature indicators are mounted on the engine instrument panel. The indicators are calibrated in 20° increments from 0°C to 1000°C. The indicators provide a visual indication of the respective engine temperature so that accurate engine power settings may be obtained. On airplanes NA-370-2 through -6, -8 and -9, a dual junction thermocouple cable and three dual-junction thermocouple probes are installed on each engine. Two connectors are provided in the thermocouple cable (installed on each engine lower aft section), one for connecting to the system exhaust temperature indicator and the other for individual checking of the thermocouple probes on the ground.

On airplanes NA-370-1, -7, and NA-380-1 and later, exhaust gas temperature is sensed by eight single loop thermocouples arranged in the form of a single integral harness and lead assembly. The harness is mounted on the turbine casing with the probes extending through the inner fan front frame casing. The thermocouple harness is terminated at an electrical connector on top of the engine. An engine temperature protector (ETP) is installed forward of the thermocouple receptacle. The ETP biases the EGT circuit resistance to provide a direct correlation with each engine's true cycle temperature as determined by the engine manufacturer. Every ETP unit is matched for each engine and it must not be removed and installed on a different engine. The ETP contains a 0-10 ohm variable resistor and the unit is pre-set and sealed at the factory or overhaul facility.

Alumel and Chromel thermocouple wiring is used to connect the thermocouple harness to the indicators, and adjustable resistance spools are used to calibrate the systems. During engine operation, the hot engine exhaust gases flowing through the engine expose the thermocouple junctions (inside thermocouple probes evenly spaced around the engine) to the varying temperatures. This generates minute voltages between the dissimilar metals of the thermocouple leads, in turn operating the respective exhaust temperature indicator.

2. Engine Exhaust Temperature Indicators

The engine exhaust temperature indicators are hermetically sealed thermocouple thermometers. They are operated by the currents generated by the thermocouple system installed on each engine tail pipe. The indicator dials are calibrated in degrees centigrade and the terminal studs on the back of the indicators are marked by positive and negative symbols. The positive marking is for the Chromel wire and the negative marking is for the Alumel wire. The indicators may be zero-adjusted by use of the adjustment screw on the back side of the indicators. The indicators are secured to the instrument panel by clamp-type mountings, and the indicators are equipped with integral lighting.

3. Exhaust Temperature Thermocouple Leads, Tail-pipe Probes, and Adjustable-resistance Assemblies

Chromel and Alumel thermocouple leads connect each engine exhaust temperature indicator to the respective engine thermocouple harness and probes. On NA-370-2 through -6, -8 and -9 airplanes, a ground test plug is installed on each engine for testing the tail-pipe probe thermocouples individually. On NA-370-2 through -6, -8 and -9 airplanes, adjustable thermocouple resistance spool assemblies are installed in the aft fuselage equipment compartment, just forward and above the aft lower fuselage equipment access doors. On NA-380 airplanes, the adjustable spools are located above the electrical equipment rack at fuselage station 143 (RH).

EXHAUST TEMPERATURE INDICATING SYSTEMS - TROUBLE SHOOTING

NOTE: When trouble shooting the wiring of indicating systems, pay particular attention to electrical plugs at point of soldering. Corrosion, mishandling, and metal fatigue can result in failure at these points. Use an ohmmeter when making continuity checks. A continuity tester will not indicate high resistance at faulty soldered connections.

1. Trouble Shooting Exhaust Temperature Indicating Systems

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. NO INDICATION.		
(1) Open electrical circuit.	Disconnect plugs from indicator and generator and check continuity.	Repair open circuit.
(2) Faulty indicator.	Interchange right and left engine indicators.	Replace faulty indicator.
(3) Open Engine Temperature Protector (ETP) CF700-2D-2 Engines.	Perform continuity check of ETP (resistance will be 0 to 10 ohms).	Replace ETP.
B. VARIABLE INDICATION.		
(1) Loose connections.		Tighten connections.
(2) Corroded connections.		Remove corrosion.
(3) Intermittent open circuit, or short circuit.	Disconnect indicator and check continuity of wiring.	Repair wiring.
(4) Faulty indicator.	Interchange right and left engine indicators.	Replace faulty indicator.
C. HIGH OR LOW INDICATION.		
(1) Incorrect circuit resistance.	Refer to procedure for adjusting engine exhaust temperature resistance assembly - initial adjustment for new resistance spools only.	Adjust resistance.
(2) Faulty indicator.	Interchange right and left engine indicators.	Replace faulty indicator.
(3) Shorted thermocouple leads.	Refer to procedure for checking for grounded thermocouple leads.	Repair thermocouple leads.
(4) Faulty engine thermocouple harness or tail-pipe probe.	Refer to procedure for adjusting engine exhaust temperature resistance assembly - initial adjustment for new resistance spools only.	Replace faulty engine harness and/or probe.

MAINTENANCE PRACTICES - EXHAUST TEMPERATURE INDICATING SYSTEMS

1. Removing and Installing Engine Exhaust Indicators

For procedures necessary to remove and install engine exhaust indicators, refer to Chapter 31.

2. Removing Exhaust Temperature Harness and Thermocouple Probes - JT-12 Engines

A. The exhaust temperature harness and thermocouple probes are installed on the engines and may be reached through the engine access doors. Connector plugs, one for normal system connection, and one for individual thermocouple test, are installed on the lower side of each engine.

NOTE: When thermocouple harness assemblies are removed from the engines, the assemblies should be handled carefully to prevent sharp bends and kinks in the cables. Otherwise, the insulation will be damaged and result in erroneous indications. Also, the leads should be kept free of oil and other materials. If the engine thermocouple harness is to be stored, it should be hung on a rack which will not distort the normal shape of the harness.

(1) To remove engine harness only, disconnect harness leads from each thermocouple probe around engine exhaust case.

(2) Remove clamps securing harness assemblies to exhaust case, and remove nuts and bolts securing thermocouple cable connectors to brackets on turbine case.

(3) Remove harness assemblies from engine exhaust case and carefully place them on supporting rack.

(4) To remove thermocouple probes only, disconnect engine harness branch leads from each thermocouple probe.

NOTE: When removing the inboard exhaust temperature thermocouple probe, it may be necessary to remove the right engine to obtain clearance for the probe.

(5) Remove bolts and nuts securing thermocouple probes to exhaust case, and remove probes.

3. Installing Exhaust Temperature Harness and Thermocouple Probes - JT-12 Engines

A. To install exhaust temperature harness and thermocouple probes, proceed as follows:

(1) Place exhaust temperature harness around exhaust case and, working from middle toward both ends of harness, secure with clips.

(2) Connect each branch lead to proper thermocouple. Arrange leads with enough radius bends, and tighten terminal nuts.

CAUTION: DO NOT ALLOW LEAD TERMINALS TO ROTATE WHILE TIGHTENING NUTS. OTHERWISE, THE SLACK WILL BE REMOVED FROM THE RADIUS BENDS.

(3) Install harness connectors to bracket on exhaust case, and secure with nuts and bolts.

(4) When installing thermocouple probes only, place probes in mounting bosses and secure with bolts and nuts; then connect thermocouple leads to probes.

NOTE: When installing the inboard thermocouple probe on the right engine, it may be necessary to remove the engine.

4. Removing Exhaust Temperature Harness and Probes - CF700-2D-2 Engines

A. The exhaust temperature harness and thermocouple probes are installed on the engines and may be reached through the engine access doors. A connector plug, for the system connection, is installed at the 11:00 o'clock position of each engine just aft of the top ignitor plug.

NOTE: When thermocouple harness assemblies are removed from the engines, the assemblies should be handled carefully to prevent sharp bends and kinks in the cables. Otherwise, the insulation will be damaged and result in erroneous indications. Also, the leads should be kept free of oil and other materials.

(1) To remove thermocouple harness, disconnect harness connector from bracket.

(2) Remove clamps securing harness assembly to standoff brackets attached to turbine housing forward flange.

(3) Remove bolts from the mounting pads on the fan front frame.

(4) Remove harness assembly from engine and carefully place it on supporting rack.

5. Installing Exhaust Temperature Harness and Probes - CF700-2D-2 Engines

A. To install exhaust temperature harness and probes, proceed as follows:

(1) Install the thermocouples in the mounting pads. Ensure that copper gaskets are installed over each probe.

(2) Lubricate threads of the thermocouple mounting bolts with milk of magnesia. Install bolts and torque to 24-27 inch-pounds. Lockwire bolts.

NOTE: Do not allow anti-seize compound to get on the thermocouple elements. Use extreme care when inserting elements into the mounting pads to prevent damage to elements.

(3) Clamp harness to brackets. Tighten clamp bolts.

(4) Install harness connector to mounting bracket. Tighten screws and lockwire.

(5) Check system for proper resistance. Refer to "Adjusting Engine Exhaust Temperature Resistance Assembly."

(6) Close engine access doors.

(7) Perform operational check of engine.

1. Adjusting Engine Exhaust Temperature Resistance Assembly JT-12 Engines - Initial Adjustment For New Resistance Spools Only

A. The right and left engine variable-resistance assemblies are installed on the right and left sides of the fuselage, and may be reached through the aft fuselage lower access doors.

(1) Disconnect thermocouple leads from right or left engine exhaust temperature indicator, depending on system being adjusted, and connect leads together.

(2) Remove cover from adjustable resistance assemblies.

(3) Attach a Wheatstone bridge between open end of resistance spool and open end of thermocouple lead. Then measure total resistance of complete thermocouple circuit.

(4) Unwind one coil at a time from resistance spool and for each turn removed, measure resistance of thermocouple circuit until the resistance measured is slightly greater than 8 ohms.

CAUTION: DO NOT REMOVE TOO MUCH RESISTANCE. IF TOO MUCH RESISTANCE IS REMOVED, THE THERMOCOUPLE CIRCUIT WILL BE OUT OF CALIBRATION. THE TOTAL RESISTANCE CANNOT BE LESS THAN 8 OHMS.

(5) Gradually remove resistance from spool until total resistance of thermocouple circuit is 8 (±0.05) ohms.

(6) Wrap several turns of resistance wire around nonconduction post (to prevent unwinding); then solder resistance wire to terminal. Secure ends of resistance wire on spare resistance spool, using vinyl tape.

(7) Install cover on adjustable resistance assembly, and lockwire.

(8) Connect Wheatstone bridge between indicator terminals, and check resistance of thermocouple leads. The resistance should be 8 (±0.05) ohms.

(9) Connect thermocouple leads to indicator. Connect green wire to post marked "–," and connect white wire to post marked "+."

(10) With ohmmeter, measure insulation resistance between thermocouple system and airplane structural ground, by attaching one test lead to an indicator post and the other test lead to airplane ground. The resistance should not be less than 10,000 ohms.

2. Adjusting Engine Exhaust Temperature Resistance Assembly CF700-2D-2 Engines - Initial Adjustment For New Resistance Spools Only

A. The right and left engine variable-resistance assemblies are installed at fuselage station 155 RH. The total resistance of the aircraft circuit and engine thermocouple harness with the Engine Temperature Protector (ETP) removed from the circuit is 6.5 ohms.

(1) Open engine cowling and disconnect ETP from engine thermocouple harness and aircraft harness. The ETP is bypassed for the total resistance check.

(2) Disconnect thermocouple leads from right or left engine exhaust temperature indicator, depending on system being adjusted, and connect leads together.

(3) Remove cover from adjustable resistance assemblies.

(4) Attach a Wheatstone bridge between open end of resistance spool and open end of thermocouple lead. Then measure total resistance of complete thermocouple circuit.

(5) Unwind one coil at a time from resistance spool and for each turn removed, measure resistance of thermocouple circuit until the resistance measured is slightly greater than 6.5 ohms.

CAUTION: DO NOT REMOVE TOO MUSH RESISTANCE. IF TOO MUCH RESISTANCE IS REMOVED, THE THERMOCOUPLE CIRCUIT WILL BE OUT OF CALIBRATION. THE TOTAL RESISTANCE CANNOT BE LESS THAN 6.5 OHMS.

(6) Gradually remove resistance from spool until total resistance of thermocouple circuit is 6.5 ($\pm$0.05) ohms.

(7) Wrap several turns of resistance wire around nonconduction post (to prevent unwinding); then solder resistance wire to terminal. Secure ends of resistance wire on spare resistance spool, using vinyl tape.

(8) Install cover on adjustable resistance assembly, and lockwire.

(9) Connect Wheatstone bridge between indicator terminals, and check resistance of thermocouple leads. The resistance should be 6.5 ($\pm$0.05) ohms.

(10) Connect thermocouple leads to indicator. Connect green wire to post marked "-," and connect white wire to post marked "+."

(11) Connect ETP to engine and aircraft harness.

(12) With ohmmeter, measure insulation resistance between thermocouple system and airplane structural ground, by attaching one test lead to an indicator post and the other test lead to airplane ground. The resistance should not be less than 10,000 ohms.

1. Checking Exhaust Temperature Indicating System

A. To check exhaust temperature indicating system, proceed as follows:

(1) Check indicators for security of mounting, cleanness and tightness of glass, and general condition of dial markings. Check electrical connections, and check for freedom from corrosion.

(2) Check thermocouple leads for signs of wear and deterioration.

(3) When engines are started, check that indication increases rapidly until normal operating temperatures are reached.

(4) Check that indicators do not oscillate more than 25°C during flight of airplane.

B. To bench check the exhaust gas temperature indicator, proceed as follows:

In order to perform a bench check of the exhaust gas temperature indicator the indicator shall be subjected to a constant room temperature for a period of not less than one hour. Millivoltage shall be supplied to the indicator as specified, for the ambient room temperature for the test points specified. The millivoltage shall be applied to the indicator through a series resistance of eight ohms. The millivoltage applied shall be measured with a potential measuring instrument having a minimum accuracy of plus or minus 0.2 percent. The indicator shall be tapped lightly before each reading and the difference in reading after tapping shall be considered as friction error. Indicator may be adjusted for minimum error at 700°C.

Test Point Degree C.	Scale Error Tolerance ± °C	Friction Tolerance ± °C.
500	25	30
600	15	30
700	10	30
800	15	30
900	25	30
1000	30	30

CAUTION: ATTACH SHORTING BAR BETWEEN INDICATOR TERMINALS DURING TRANSIT TO AVOID INDICATOR DAMAGE.

Ambient Room Temp	Millivolts for Degrees C Test Points							
Degree C	0	400	500	600	700	800	900	1000
0 (32F)	0	16.40	20.65	24.91	29.14	33.30	37.36	41.31
16 (60.8F)	-0.64	15.75	20.00	24.26	28.50	32.67	36.72	40.67
17 (62.6F)	-0.68	15.71	19.96	24.22	28.46	32.63	36.68	40.63
18 (64.4F)	-.072	15.67	19.92	24.18	28.42	32.59	36.64	40.59
19 (66.2F)	-0.76	15.63	19.88	24.14	28.38	32.55	36.60	40.55
20 (68.0F)	-0.80	15.59	19.84	24.10	28.34	32.51	36.56	40.51
21 (69.8F)	-0.84	15.55	19.80	24.06	28.30	32.47	36.52	40.47
22 (71.6F)	-0.88	15.51	19.76	24.02	28.26	32.43	36.48	40.43
23 (73.4F)	-0.92	15.47	19.72	23.98	28.22	32.39	36.44	40.39
24 (75.2F)	-0.96	15.43	19.68	23.94	28.18	32.35	36.40	40.35
25 (77.0F)	-1.00	15.39	19.64	23.90	28.14	32.31	36.36	40.31
26 (78.8F)	-1.04	15.35	19.60	23.86	28.10	32.27	36.32	40.27
27 (80.6F)	-1.08	15.31	19.56	23.82	28.06	32.23	36.28	40.23
28 (82.4F)	-1.12	15.27	19.52	23.78	28.02	32.19	36.24	40.19
29 (84.2F)	-1.16	15.23	19.48	23.74	27.98	32.15	36.20	40.15
30 (86.0F)	-1.20	15.19	19.44	23.70	27.94	32.11	36.16	40.11

2. Checking for Grounded Thermocouple Leads

A. To check for grounded thermocouple leads, proceed as follows:

(1) Remove exhaust temperature indicators from instrument panel.

(2) Disconnect thermocouple leads from right or left engine indicator.

(3) Using multimeter, check resistance from each thermocouple lead to ground. The resistance should be at least 10,000 ohms.

(4) Reconnect thermocouple leads to indicator, ensuring that the Alumel lead (green) is connected to indicator terminal marked "–," and the Chromel lead (white) is connected to indicator terminal marked "+."

3. Checking Harness and Probe Resistance - JT-12 Engines

A. To check harness and probe resistance, proceed as follows:

(1) Disconnect connector plug (normal system, two-pin plug) on lower side of engine.

(2) Connect lead of Wheatstone bridge to pin "A" of engine harness wiring, and connect other test lead to pin "B." The resistance should be between 0.6 and 1.0 ohm. Also, check that resistance of engine thermocouple probe terminals to thermocouple shell is at least 100,000 ohms.

(3) Disconnect Wheatstone bridge leads and reconnect airplane thermocouple wiring plug, and lockwire.

CHAPTER 78

EXHAUST

NOTICE

The following information applies to NA 265-70 airplanes only.

CHAPTER 78

EXHAUST

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CHAPTER 78

EXHAUST

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EXHAUST - DESCRIPTION

1. General

The exhaust system of the engine consists of the exhaust case, inner cone and strut assembly, tail pipe, and thrust reverser assembly. The exhaust system directs high-velocity gases from the engine turbine section aft and overboard to provide the thrust for the engine, and diverts exhaust gases forward to decelerate the airplane during the landing roll. For information relative to the exhaust case and the inner cone and strut assembly, refer to Pratt & Whitney engine maintenance manual, PWA 435107.

COLLECTOR - DESCRIPTION

1. General

The engine exhaust tail pipe is a welded tapered tube, constructed from high-temperature, corrosion-resistant steel. A mounting flange is welded to the forward end, and a thrust reverser assembly consisting of two doors (targets) with related actuating mechanism are attached to the aft end of the tube.

The tail pipe and thrust reverser assembly is attached to the aft end of the engine exhaust case with 48 mounting bolts.

THRUST REVERSER SYSTEM

DESCRIPTION AND OPERATION

GENERAL

The engine thrust reverser system is used to decelerate the airplane during the landing roll by diverting the jet exhaust gases forward. This is accomplished by two doors (targets) attached to the rear of the tail pipe. When the doors are deployed (reverse thrust position), the exhaust gases are diverted forward above and below the engine pod. The reverser system is electrically controlled and hydraulically actuated, receiving electrical power from the 28 VDC essential bus, and hydraulic pressure from the normal hydraulic power system. Each engine reverser system is an independent system. Each reverser consists of a hydraulic control valve, an actuator, a latch unlock solenoid, two position switches, and the necessary mechanical linkages and electrical wiring to operate the system. A single hydraulic accumulator is installed, to allow actuation of the reverser doors in case of normal hydraulic power system failure. Selection of the reverser doors (stowed or deployed) is accomplished by movement of the thrust reverser control levers, which are mounted on the throttle levers in the cockpit. These levers are mechanically connected to the throttle control system. The reverser control levers are blocked by an unlock solenoid after initial movement. The reverser doors must be fully deployed before further movement of the reverser control levers can be obtained. An adjustable thrust reverser control lever stop assembly is also installed on each reverser control lever. This stop is used to obtain maximum reverse thrust during landings at extremes of field elevation and temperature, and to prevent overboosting the engine. The doors, when stowed, form the aft section of the engine cowling. The doors are operated (stowed or deployed) by a hydraulic actuator. The actuator is connected to a main drive link (bogie), which in turn is connected to each door through driver links. The hydraulic actuator contains an internal lock which provides the primary means for locking the reverser doors in the stowed position. Secondary locking latches are also provided for the doors. During in-flight conditions, normal hydraulic power system pressure is isolated from the actuator by the reverser hydraulic control valve. This is accomplished when the main landing gear downlock switches are deactuated, removing electrical power from the valve, allowing it to assume a neutral position where both the deploy and stow ports of the actuator are opened to the return side of the normal hydraulic power system. Six indicator lights are installed in the cockpit to indicate the mode of the reverser doors during ground and flight operations. Two master caution lights are also included in the indicator light system. The caution lights come on in-flight when either the ARM or the UNLOCK indicator light comes on. A guarded thrust reverser switch is also included in the reverser system. (See Figure 78-30-1).

OPERATION OF THRUST REVERSER SYSTEM

Operation of the thrust reverser system is the same for both left and right engines. During a normal operational sequence of the reverser system, the main landing gear downlock switches must be closed (main gear locked down) and ENGINE MASTER switch positioned at ON. To deploy the reverser doors (targets), the THRUST REVERSER control switch must be positioned at ON, and the power levers must be at IDLE. With the power lever at IDLE, the ARM switch and relay are energized and power from the essential bus is then applied to the ARM indicator light and the stow solenoid of the reverser hydraulic control valve. When the reverser levers are moved up and aft, the reverser control lever switches are actuated, thus removing electrical power and the grounding circuits from the stow solenoid of the hydraulic control valve and completing the power and ground circuits to the secondary latch control relay. This energizes the secondary latch control relay and the secondary latch control unlock solenoid, which unlocks the secondary door locks. At the same time, both secondary door lock position indicator switches close, illuminating the UNLOCK indicator light and energizing the deploy solenoid of the reverser hydraulic control valve, which allows hydraulic pressure to be routed to the deploy port of the hydraulic actuator, which then deploys the reverser doors. When the doors are fully deployed, both internal switches of the deploy position indicator switch close, thus illuminating the DEPLOY indicator light and energizing the reverser control lever unlock solenoid, allowing the reverser control levers to be moved to the full reverse-thrust position.

Stowing of the reverser doors is essentially the reverse of the deployment procedure. To stow the reverser doors, the reverser control levers are placed in the full forward position. The reverser doors will actuate to the stowed position. After the doors are fully stowed, the UNLOCK and DEPLOY indicator lights go out.

If the UNLOCK indicator light comes on in-flight or the reverser doors fail to stow with the normal control system, the guarded thrust reverser control switch in cockpit may be actuated to EMERG STOW after the affected engine throttle is retarded to IDLE. When this switch is actuated to EMERG STOW, it applies electrical power directly from the essential bus to the stow solenoid of the hydraulic control valve. Hydraulic pressure is then routed to the stow port of the actuator, thus holding the reverser doors in the stowed position. (See Figure 78-30-2.)

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DESCRIPTION AND OPERATION (Continued)

OPERATION OF THRUST REVERSER SYSTEM (Cont'd)

If the reverser actuating mechanism fails when the doors are fully deployed, the engine exhaust gases (at thrust settings above approximately 80% rpm) will hold the doors in the deployed position.

When the thrust reverser control switch is in the normal or on position, in case of an engine fire, the reverser system is automatically de-energized (isolated) when the FIRE PULL "T" handle of the affected engine is pulled. The reverser hydraulic control valve will then assume a neutral position where both the stow and deploy ports of the hydraulic actuator are opened to the return side of the normal hydraulic power system, thus minimizing the fire hazard in case of hydraulic line failure. Line check valves are also installed to prevent fluid from backflowing from the pressurized reservoir into the engine pod.

On S/N 370-2 thru -6, -8 and -9 airplanes changed by S/B 74-8 an automatic stow function is incorporated in the thrust reverser system. The system illuminates a caution light and automatically energizes the stow solenoid of thrust reverser hydraulic control valve whenever the reverser doors leave the stowed position switch in flight (gear not down and locked). See Figure 78-30-2. The system consists of an amber thrust reverser unsafe caution light, auto stow relay, thrust reverser unlock monitor relay, and interconnecting wiring. The circuitry is energized through action of the stowed position switch (S1225) whenever the landing gear is released from its downlocks. With gear up and locked, reversers stowed, and thrust reverser control switch in OFF position, should the reverser doors move off of the door stops (or actuation point of stowed position switch), power from the thrust reverser indicator lights circuit breaker is routed to the UNLOCK light and to the thrust reverser unlock monitor relay (K1274 or K1273). The relay energizes, routing power from the thrust reverser emergency stow circuit breaker (through a fire shutoff relay) to the coil of auto stow relay (K1276 or K1275). When the auto stow relay is energize, the power and ground circuits to the deploy solenoid of the hydraulic control valve are broken and both power and ground circuits are connected to the stow solenoid, which in turn allows hydraulic pressure to be applied to the reverser actuating cylinder to restow the doors. At the same time a circuit is completed to the thrust reverser unsafe light. The auto stow relay and the thrust reverser unsafe light will remain energized (even though the doors are restowed and the UNLOCK light goes off) until the thrust reverser emergency stow circuit breaker is pulled (or essential bus power is interrupted). The dimming function (resistor R1016 or R1015) for the UNLOCK indicator light is inoperative when the thrust reverser system is not armed due to a shorting action of contacts in the number two arm relay (K1258 or K1259).

A further safeguard controlling the auto stow feature is accomplished by S/B 29. Airplane S/N 370-1 and -7 and 380-1 thru -4, -6 thru -9 and -11, changed by S/B 17 and 380-5, -10 and -12 thru -58 may be modified per S/B 29 to feature an auto stow mode in the event of a failure of the MLG downlock switch. With the wiring modification described in S/B 29 accomplished, a RH downlock failure will cause both LH and RH thrust reversers to auto stow. If the LH downlock should fail the thrust reverser will not go into auto stow.

THROTTLE LEVER ARM SWITCH AND INDICATOR LIGHT

The throttle lever arm switches are mounted in the engine control quadrant. These switches are actuated when the throttle lever is at IDLE. When these switches are actuated, power and a ground circuit are provided for the coil of the arm relay, thus energizing the arm relay. Power is then applied through the arm relay to the ARM indicator light. Power and a ground circuit are also applied through the arm relay to the reverser control lever switches. This circuit is normally energized only after actuation of the main landing gear downlock switches and the throttle lever arm switches.

REVERSE LEVER LOCKOUT AND SECONDARY LATCH MONITOR RELAYS

The reverse lever lockout and secondary latch monitor relays are installed in the thrust reverser system to provide an additional safety feature during thrust reverser deployment. The secondary latch monitor relay is energized by the secondary latch control unlock solenoid. The reverse lever lockout relay is energized only if there is a failure of the secondary latch control unlock solenoid.

REVERSER CONTROL LEVER SWITCHES

The reverser control lever switches are mounted in the engine control quadrant and are actuated by the reverser control levers. Electrical power is applied to the stow solenoid of the hydraulic control valve through these switches when the reverser control levers are in the stowed position. Lifting the reverser control levers actuates these switches and applies electrical power to the deploy solenoid of the hydraulic control valve for deployment of the reverser doors. When the reverser control levers are returned to the stowed position, electrical power is again applied to the stow solenoid of the hydraulic control valve.

DEPLOY POSITION SWITCH

The deploy position switch (with two internal switches) is mounted on the inboard side of each thrust reverser assembly. This switch is actuated by a cam attached to the top reverser door pivot point. When the doors are in the fully deployed position, one switch applies electrical power to release the reverser control lever unlock solenoid. The other switch applies electrical power to illuminate the DEPLOY indicator light in the cockpit.

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DESCRIPTION AND OPERATION (Continued)

STOW POSITION SWITCH

A stow position switch is mounted on the inboard side of each thrust reverser. This switch is actuated by a cam attached to the forward inboard side of the top reverser door. When the reverser doors are in the fully stowed position and the secondary latch control unlock solenoid is de-energized, electrical power is removed from the UNLOCK indicator light in the cockpit. This switch provides a holding circuit for the secondary latch monitor relay and, on airplanes changed by S/B 74-8, also provides the initiation signal for the automatic stow circuit.

THRUST REVERSER CONTROL SWITCH

A guarded thrust reverser control switch is mounted in the cockpit. Actuation of this switch applies electrical power from the 28 VDC essential bus directly to the stow solenoid of the hydraulic control valve. This energizes the solenoid, allowing hydraulic pressure to be routed to the stow port of the hydraulic actuator. Actuation of this switch to ON or EMER STOW position provides the grounding circuit for four of the relays and all of the solenoids in the thrust reverser system.

SECONDARY LATCH CONTROL UNLOCK SOLENOID

A secondary latch control unlock solenoid (with two internal switches) is mounted on the inboard side of each reverser. The solenoid releases the secondary latch, which would hold the doors in the stowed position should the primary lock fail. When the solenoid is energized, the latch disengages from the doors, allowing them to deploy. When the solenoid is de-energized, the latch provides a secondary means of retaining the doors in the stowed position. The two internal switches are used to apply electrical power to the deploy solenoid of the hydraulic control valve and the UNLOCK indicator light.

REVERSER CONTROL LEVER UNLOCK SOLENOIDS

The reverser control lever unlock solenoids are in the engine control quadrant. When the reverser doors are fully deployed, one set of contacts in the deploy position switch closes, routing power to the unlock solenoids; the energized solenoids unlock the reverser control levers, allowing them to be moved to the full reverse-thrust position. When the reverser control levers are returned to the minimum thrust position, the unlock solenoids are de-energized and the levers are relocked.

REVERSER HYDRAULIC CONTROL VALVE

The reverser hydraulic control valve is a conventional spool-type reversing valve. Two valves are in the aft fuselage equipment compartment. These valves are operated by two electrically actuated solenoids attached to each end of the valve. The solenoids are controlled by the reverser control lever switch and deploy position switch. The stow solenoid may also be operated by the thrust reverser control switch. When both solenoids are de-energized, the valve spool returns to a neutral position, allowing the stow and deploy ports of the hydraulic actuator to be opened to the return side of the normal hydraulic power system. When the stow solenoid of the hydraulic control valve is actuated, hydraulic pressure is applied to the stow port of the hydraulic actuator. When the deploy solenoid of the hydraulic control valve is actuated, hydraulic pressure is applied to the deploy port of the hydraulic actuator. Line check valves are also installed in the system return lines to prevent flow from the hydraulic reservoir into the engine pod.

REVERSER HYDRAULIC ACCUMULATOR

The reverser hydraulic accumulator is mounted in the aft fuselage equipment compartment. The accumulator is a piston type (50-cubic-inch capacity) with an air filler valve and an air pressure gauge. The accumulator is precharged to 900 (±50) psi. The accumulator (when charged) is sufficient to operate the reverser doors through one complete cycle without the aid of the normal hydraulic power system pump. Hydraulic pressure in the accumulator is protected by check valves against loss due to normal hydraulic power system failure.

REVERSER HYDRAULIC ACTUATOR

A single hydraulically operated actuator is used to drive each reverser and is installed on the outboard side of each reverser assembly. The actuator contains an internal lock which is the primary means for locking the doors in the stowed position. The actuator (when locked) requires 500 to 900 psi hydraulic pressure to unlock the piston, allowing the actuator to extend, which then deploys the reverser doors. Flow restrictor valves, located beneath the thrust reverser assembly, control the rate of operation of the actuator.

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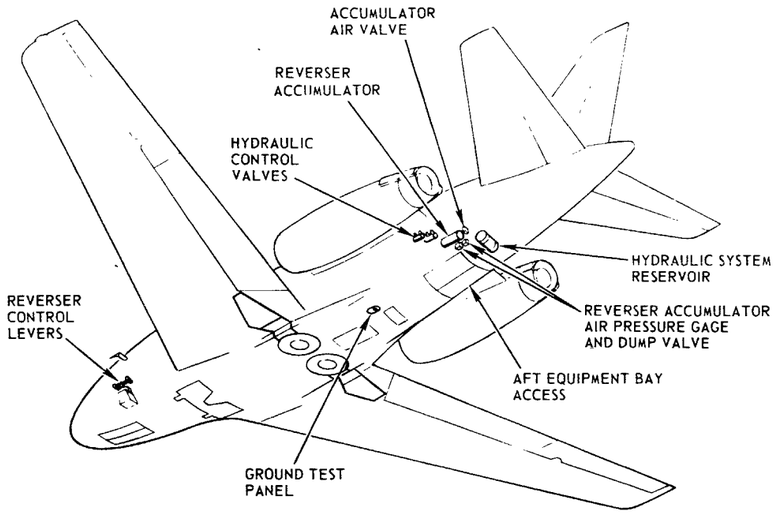
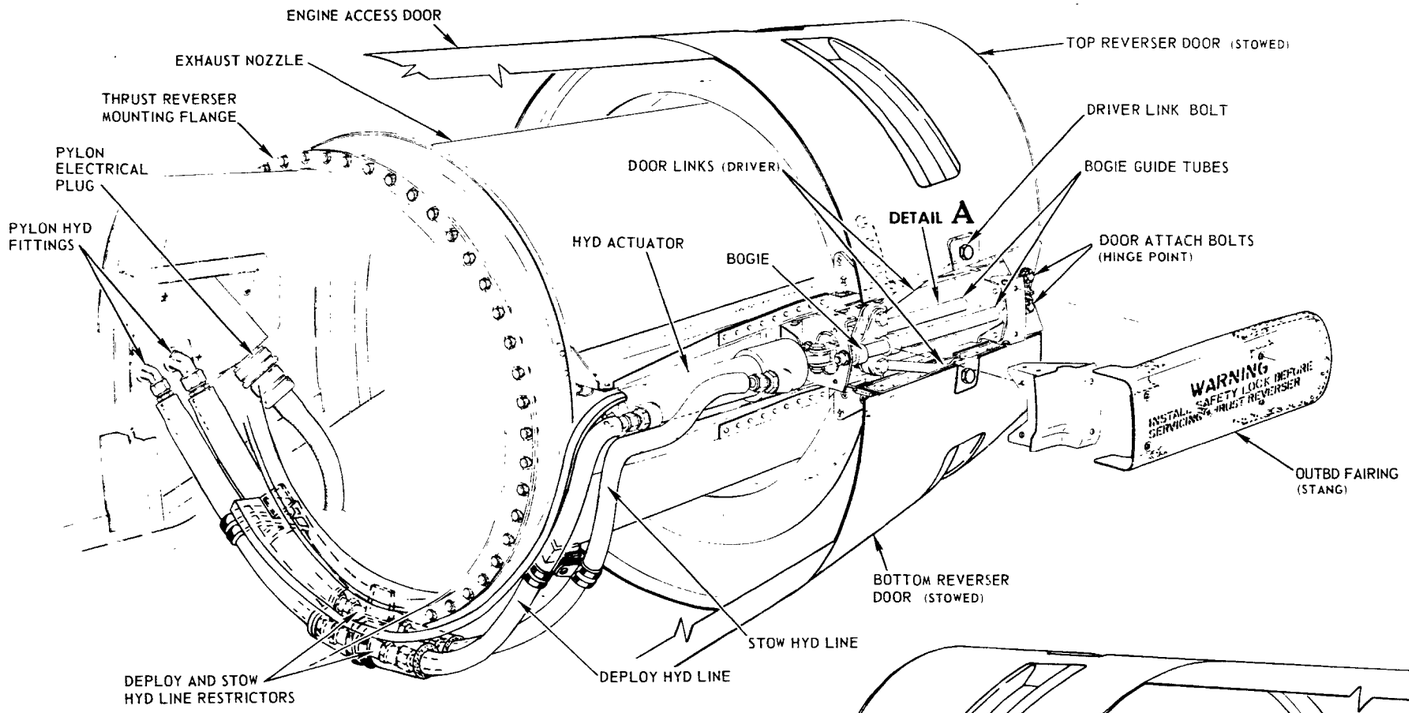
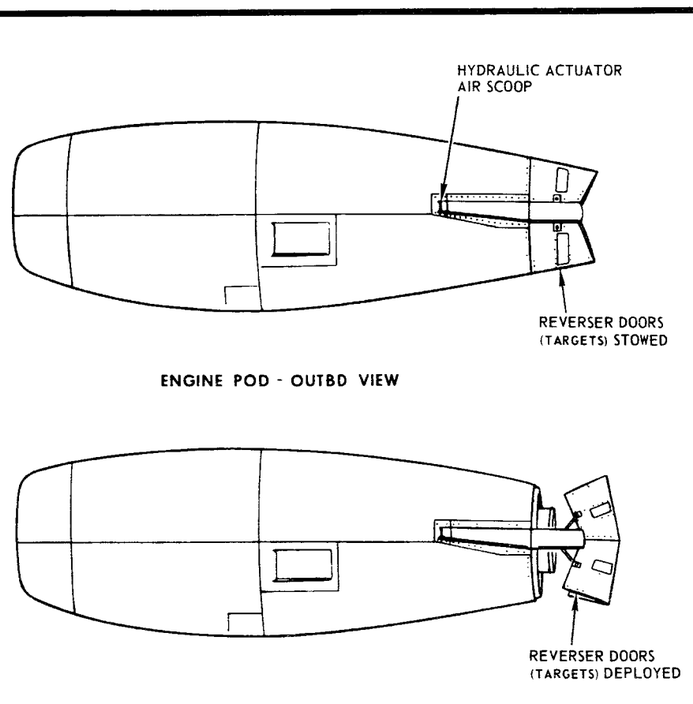
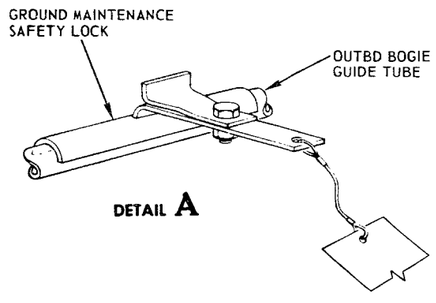


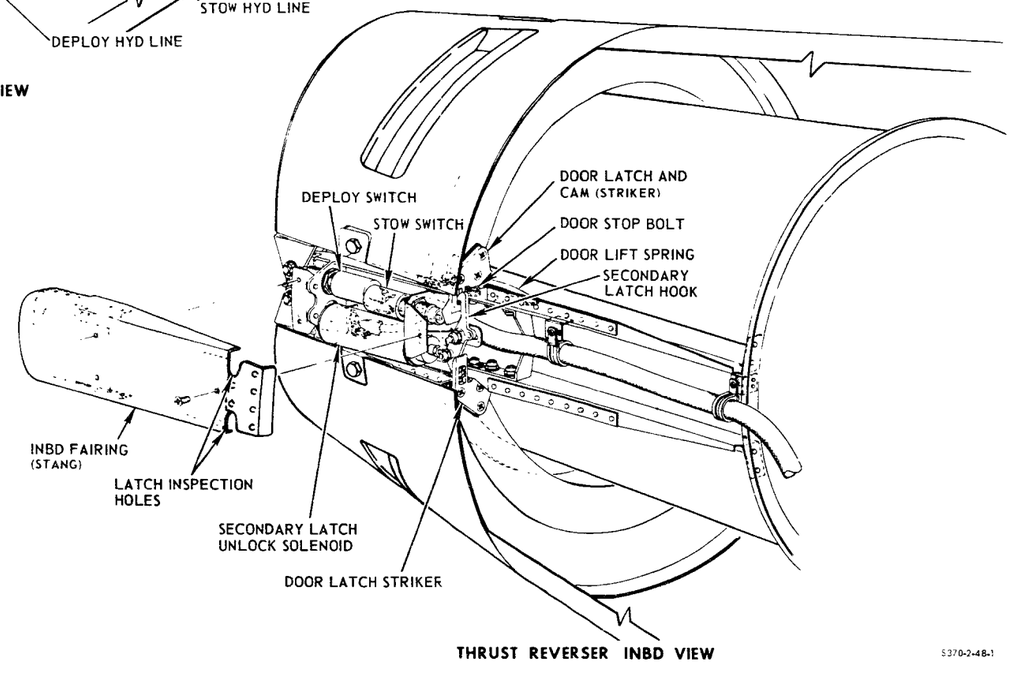


THRUST REVERSER - OUTBD VIEW



ENGINE POD - OUTBD VIEW





THRUST REVERSER INBD VIEW

5370-2-48-1

Figure 78-30-1 — Thrust Reverser Component Locator

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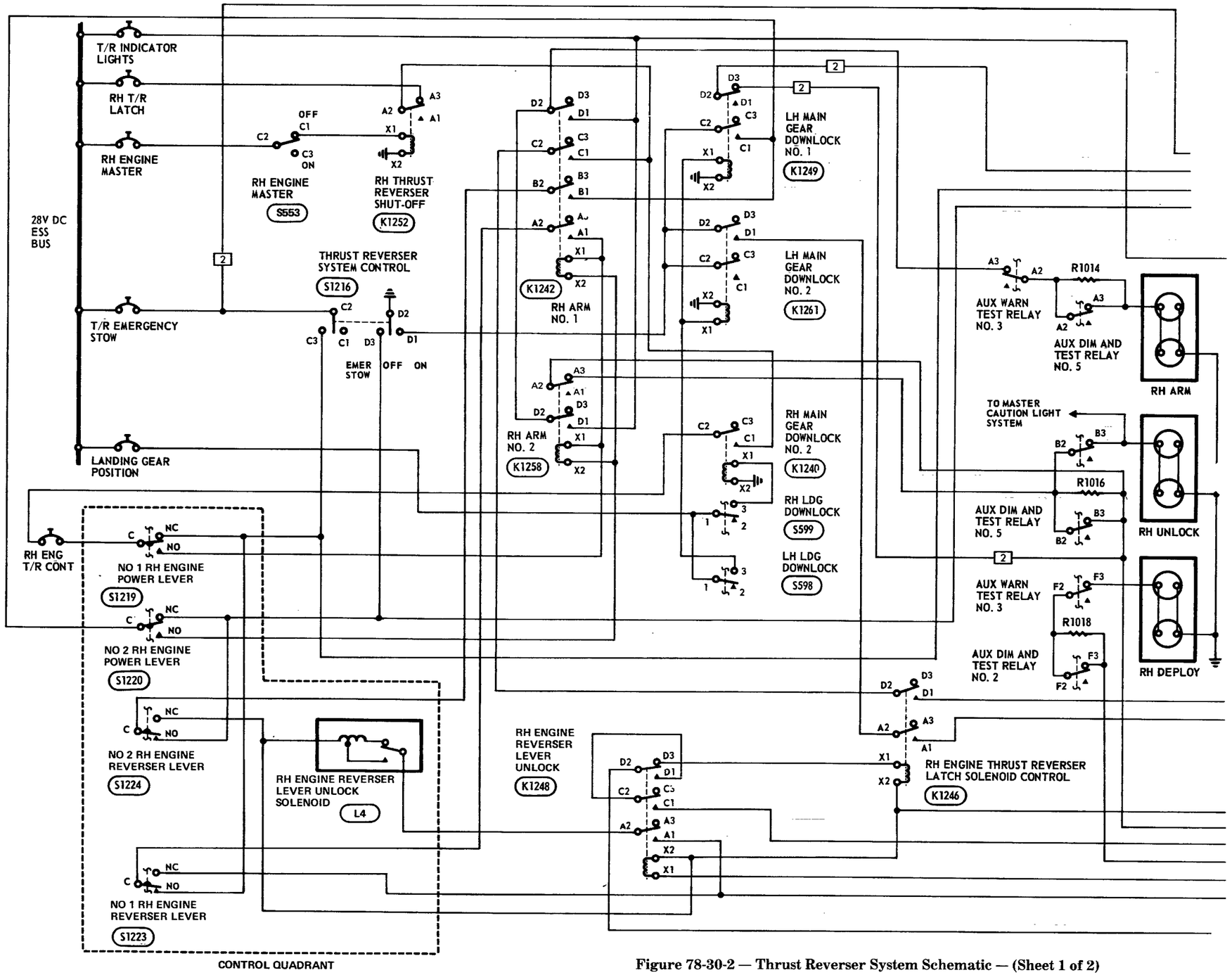

Figure 78-30-2 — Thrust Reverser System Schematic — (Sheet 1 of 2)

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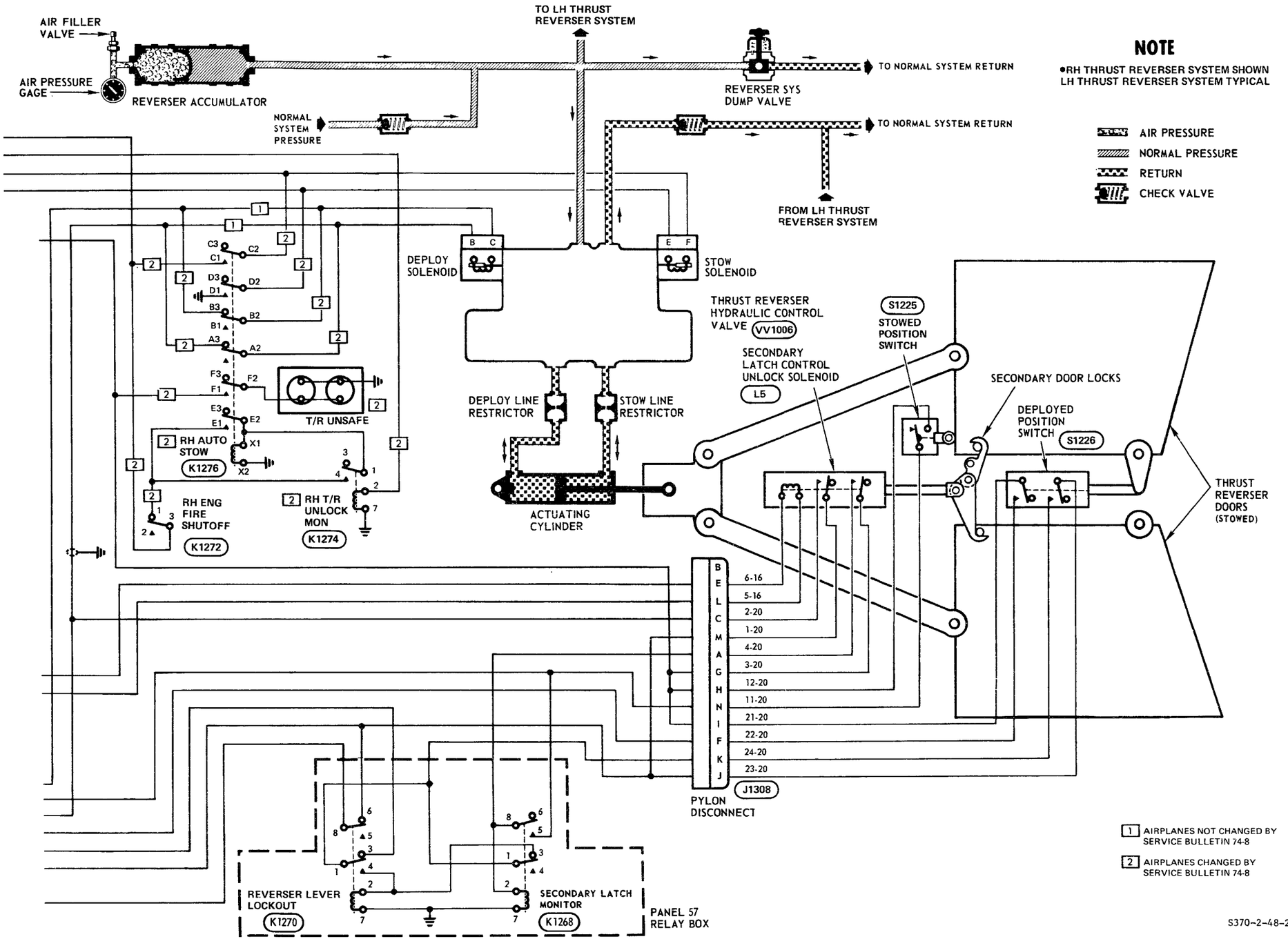

Figure 78-30-2 — Thrust Reverser System Schematic — (Sheet 2 of 2)

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THRUST REVERSER SYSTEM - TROUBLE SHOOTING

1. Trouble Shooting Thrust Reverser System

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. THRUST REVERSER WILL NOT DEPLOY.		
(1) No hydraulic power.	Check normal hydraulic power system for pressure.	If low or no pressure is indicated, trouble shoot hydraulic system.
(2) Deploy solenoid in hydraulic control valve defective.	Check for 28-volt dc between pins "B" and "C" of hydraulic control valve plug.	If correct voltage is present, replace valve.
(3) Main landing gear downlock switches defective.	If arm indicator light is out, check main landing gear downlock switches for continuity.	Replace switches as necessary.
(4) Arm relay defective.	Check for 28 volts at terminals X1 and X2 of arm relay coil.	If correct voltage is present, replace relay.
(5) Defective actuator locking unit.	Check that actuator unlocks with a minimum hydraulic pressure of 500 to 900 psi.	Replace actuator.
(6) Defective latch control unlock solenoid.	Check pins "E" and "L" of plug J1308 or J1309 for continuity.	If no continuity exists, replace solenoid.
(7) Throttle lever control switches defective and/or misadjusted.	Check switches for proper operation and/or adjustment.	Replace or adjust switches as necessary.
(8) Automatic stow feature actuated (airplanes changed by Service Bulletin 74-8.	Check LH main gear downlock switch and stow position switch (S1225).	Trouble shoot landing gear indication system; replace or adjust switch S1225.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
B. THRUST REVERSER WILL NOT STOW.		
(1) No hydraulic power.	Check normal hydraulic power system for pressure.	If low or no pressure is indicated, trouble shoot normal hydraulic system.
(2) Stow solenoid in hydraulic control valve defective.	Check for 28-volt dc between pins "E" and "F" of hydraulic control valve plug.	If correct voltage is present, replace valve.
(3) Hydraulic control valve spool stuck in deploy position.		Replace hydraulic control valve.
(4) Deploy position switch defective.	Check switch for continuity.	Replace or adjust switch as necessary.
(5) Throttle lever control switches defective and/or misadjusted.	Check switches for proper operation and/or adjustment.	Replace or adjust switches as necessary.
(6) Main landing gear downlock switches defective.	Check landing gear downlock switches for proper operation and/or adjustment.	Replace and/or adjust switches as necessary.
C. REVERSER CONTROL LEVER BLOCKED.		
(1) Deploy position switch defective.	Check for proper operation and continuity.	Adjust or replace switch as necessary.
(2) Reverser control lever unlock solenoid defective.	Check solenoid for continuity.	Replace solenoid.
D. "ARM" INDICATOR LIGHT DOES NOT COME ON.		
(1) Main landing gear downlock switches defective.	Check switches for proper operation.	Adjust or replace switches as necessary.
(2) Thrust reverser arming switches defective.	Check switches for continuity.	Adjust or replace switches as necessary.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
(3) Arming relay defective.	Check relay coil for 28-volt dc.	Replace relay.
E. "UNLOCK" INDICATOR LIGHT DOES NOT GO OUT WITH DOORS FULLY STOWED.		
(1) Stowed position indicator switch defective.	Check for continuity (switch open) at pins "H" and "N" at plug J1309 or J1308.	Replace or adjust switch as necessary.
(2) Latch solenoid control relay defective.	Check for continuity at pins "C" and "M" of J1309 or J1308.	Replace relay if necessary.

34-L NEW 12-62

THRUST REVERSER SYSTEM - MAINTENANCE PRACTICES

1. Servicing Thrust Reverser System

A. Refer to Chapter 12 for required servicing of thrust reversers.

1. Replacement of Thrust Reverser Assembly

The thrust reverser assembly is attached to the aft end of the engine exhaust cone with 48 mounting bolts. It may be removed without removing engine from airplane. The following procedures are applicable to either left or right thrust reverser assembly. See figure 78-30-2B.

WARNING: TO PREVENT POSSIBLE INJURY TO PERSONNEL, ON AIRPLANES CHANGED BY SB74-8 THE THRUST REVERSER AUTOMATIC STOW SYSTEM IS ACTIVATED ANY TIME THE MAIN LANDING GEAR IS RELEASED FROM ITS DOWNLOCK OR WHENEVER THE LANDING GEAR POSITION CIRCUIT BREAKER IS OPENED. TO PREVENT ACTIVATION OF THIS SYSTEM, THE EMERGENCY STOW CIRCUIT BREAKER SHOULD BE OPENED.

A. Removing Thrust Reverser

(1) Open lower engine access door and remove upper aft engine access door.

(2) Remove outboard fairing (stang) and install maintenance ground safety lock (Rohr 140-5444).

(3) Disconnect electrical plug from aft end of pylon.

CAUTION: DO NOT DISCONNECT HYDRAULIC HOSES AT PYLON.

- TOO RAPID DISCHARGE OF HYDRAULIC PRESSURE MAY CAUSE DAMAGE IF RESERVOIR RELIEF VALVE FLOW RATE IS EXCEEDED.

(4) Make sure all hydraulic pressures have been discharged from the three hydraulic accumulators, then disconnect hydraulic lines from reverser actuator. Cap openings.

(5) Remove bolts securing heat shield to standoff brackets, then swing hydraulic lines free of reverser assembly.

(6) Remove mounting bolts from cable brackets and forward ends of inboard and outboard stang braces.

(7) Support thrust reverser assembly and remove remaining mounting bolts. Remove assembly from engine using care to avoid striking flange against turbine tail cone.

B. Installing Thrust Reverser

NOTE: An index pin (3/16 inch diameter) is installed on forward flange of reverser assembly (at approximate six o-clock position) to assure proper alignment of thrust reverser assembly to engine case.

(1) Coat threads of mounting bolts lightly with anti-seize compound (Fel Pro C-5).

(2) Install thrust reverser assembly to engine case, aligning index pin to index hole (in engine case flange). Except at brace and bracket locations, install mounting bolts (washer and nuts) with bolt heads aft.

(3) Install mounting bolts (washers and nuts) at forward ends of inboard and outboard stang braces, bolt heads forward.

(4) Secure electrical cable support brackets to aft side of flanges with bolts (washers and nuts), bolt heads forward.

(5) Torque mounting bolts 50 to 70 inch-pounds.

(6) Connect cable plug to pylon electrical receptacle and safetywire. Check that cable is not stressed.

(7) Uncap and connect hydraulic lines to actuator fittings. Tighten "B" nuts, then secure heat shield (and hydraulic line clamps) to standoff brackets. Check that lines are not stressed.

(8) Remove maintenance ground safety lock, then provide electrical and hydraulic power and operate (deploy and stow) thrust reverser system several cycles to remove any trapped air.

(9) Referring to paragraph 78-30-4, perform adjustments as required.

(10) Install outboard fairing (stang) and upper engine access door. Close lower engine access door.

(11) Perform operational checkout of thrust reverser system (refer to paragraph 78-30-5).

2. Replacement of Thrust Reverser Door Hydraulic Actuator

WARNING: MAKE SURE THAT ALL PERSONNEL ARE CLEAR OF THRUST REVERSER DOORS BEFORE OPERATING SYSTEM.

- DO NOT PERFORM MAINTENANCE ON THRUST REVERSER SYSTEM UNLESS MAINTENANCE GROUND SAFETY LOCK IS INSTALLED.

- TO PREVENT POSSIBLE INJURY TO PERSONNEL, ON AIRPLANES CHANGED BY SB74-8 THE THRUST REVERSER AUTOMATIC STOW SYSTEM IS ACTIVATED ANY TIME THE MAIN LANDING GEAR IS RELEASED FROM ITS DOWNLOCK OR WHENEVER THE LANDING GEAR POSITION CIRCUIT BREAKER IS OPENED. TO PREVENT ACTIVATION OF THIS SYSTEM, THE EMERGENCY STOW CIRCUIT BREAKER SHOULD BE OPENED.

NOTE: These procedures are applicable to either left or right actuator (see figure 78-30-1).

A. Removing Thrust Reverser Door Hydraulic Actuator

(1) Remove outboard fairing (stang) from thrust reverser.

(2) Provide electrical and hydraulic power, deploy thrust reverser doors, and install maintenance ground safety lock (Rohr 140-5444). Place lock on bogie guide tube forward of bogie to prevent accidental closing of doors.

CAUTION: TOO RAPID DISCHARGE OF HYDRAULIC PRESSURE MAY CAUSE DAMAGE IF RESERVOIR FLOW RATE IS EXCEEDED.

(3) Remove electrical and hydraulic power, then discharge pressure from normal, auxiliary, and thrust reverser hydraulic accumulators.

(4) Open lower and remove upper engine access doors (refer to 71-10-1).

(5) Disconnect (stow and deploy) hydraulic lines from actuator fittings and install caps. Disconnect heat shield (and hydraulic line clamps) from standoff brackets as necessary to move hydraulic lines clear of actuator.

(6) Remove bolt (cotter pin, nut, and washer) that connects actuator piston rod end to bogie.

(7) Remove bolt (and washer) from forward end of actuator and remove actuator from engine.

(8) Remove fittings from actuator ports (for installation in replacement actuator) and cap ports. Discard packings.

B. Installing Thrust Reverser Door Hydraulic Actuator

NOTE: Doors must be in deployed position and maintenance ground safety lock must be installed prior to installing actuator.

(1) Using new packings (NAS1593-012) install fittings on actuator. Lubricate packings lightly with hydraulic fluid.

(2) Remove end (with jam nut and anti-rotation washer) from actuator piston shaft.

(3) Install actuator on engine with cylinder end in bracket and with hydraulic fittings outboard. Coat threads of forward bolt with anti-seize compound (Fel Pro C-5). Secure actuator to bracket with bolt and washer. Torque bolt 60 to 85 inch-pounds.

(4) Uncap and connect hydraulic lines to fittings. Tighten "B" nuts and secure heat shield to standoff brackets.

(5) Support piston end of actuator so that piston shaft may be extended and retracted under hydraulic pressure without striking any structure.

(6) Remove maintenance ground safety lock, then provide electrical and hydraulic power and deploy (extend shaft) and stow (retract) actuator several cycles to eliminate trapped air.

(7) Place control switch to EMERG STOW. Make sure piston locks into actuator, then remove electrical and hydraulic power.

(8) Coat threads of rod end (removed in step (2)) with anti-seize compound (Fel Pro C-5). Install rod end (with jam nut and anti-rotation washer) into actuator piston shaft.

(9) Referring to paragraph 78-30-4, adjust length of rod end, then install bolt (with two washers and nut) to connect rod end to bogie. Torque 60 to 85 inch-pounds and install cotter pin.

(10) Install outboard fairing (stang) and upper engine access door. Close lower engine access door.

(11) Perform operational checkout of thrust reverser system (refer to paragraph 78-30-5).

3. Replacement of Thrust Reverser Doors

A. Removing Thrust Reverser Doors

Remove reverser doors (upper and lower) as follows:

(1) Deploy thrust reverser.

(2) Remove inboard and outboard fairing and install ground maintenance safety lock.

(3) Discharge all remaining hydraulic pressure.

(4) Turn off all electrical power.

WARNING: DO NOT PERFORM FURTHER MAINTENANCE UNLESS GROUND MAINTENANCE SAFETY LOCK IS INSTALLED AND ALL HYDRAULIC PRESSURE IS DISCHARGED FROM SYSTEM.

(5) Remove cotter pins, nuts, and bolts connecting driver links to doors.

(6) Remove cotter pins, nuts, and bolts attaching reverser doors to support fittings.

(7) Remove doors.

CAUTION: TO PREVENT POSSIBLE DAMAGE TO DOOR LINKAGE, DO NOT OPERATE REVERSER SYSTEM WHEN DOORS ARE REMOVED.

B. Installing Thrust Reverser Doors

When installing a replacement door, ensure that latches, cam strikers, door stops, springs, and latch strikers are serviceable before installing new door. Both upper doors are interchangeable (left to right), and both lower doors are interchangeable. To install reverser doors, proceed as follows:

(1) Lightly coat drive link and longeron bearings and door attach bolt shanks with PT-605* lubricant.

(2) Apply a thin coat of Fel Pro C-5 antiseize compound to threads of all bolts.

(3) Remove the following items from defective door and install if serviceable on new door.

(a) Upper inboard latch and cam striker.

(b) Lower inboard latch striker.

*Procure from:
Product Techniques
1153 North Stanford
Los Angeles, California 90059

(4) Position doors in reverser assembly and install mounting bolts (heads toward center of engine). Install ground maintenance safety lock.

(5) Install washer under each nut. Tighten nut and torque to 10-15 inch-pounds. Install cotter pin.

(6) Install attach bolts (heads outboard) in driver links.

(7) Install washer under each nut. Tighten nut and torque to 60-85 inch-pounds. Install cotter pin.

(8) Remove ground maintenance safety lock and stow reverser doors.

(9) Check door contour with engine access doors.

CAUTION: Do not attempt to fair reverser doors with engine access doors by moving the reverser doors. The relationship between the door latch strikers and the latches is affected by changes in door position. Reverser door position should be established only by using rigging tool 140-SK388-1 in accordance with thrust reverser rigging instructions.

(10) Deploy doors. Check gap between upper and lower doors. Rig doors if necessary. (Refer to 78-30-4.)

4. Replacement of Thrust Reverser Stow Switch

A. Removing Thrust Reverser Stow Switch

(1) Remove inboard fairing (stang) from aft end of thrust reverser assembly, for access to stow switch. (See figure 78-30-1.)

(2) Remove jam nut holding stow switch to mounting bracket, and remove switch from bracket. (See figure 78-30-3.)

(3) Disconnect pylon electrical disconnect plug. (See figure 78-30-1.)

(4) Disassemble pylon electrical disconnect plug (MS3106K20-27P Cannon plug) as follows:

(a) Disconnect conduit coupling nut.

(b) Remove conduit adapter back shell and slide back on wires.

(c) Remove rear insulating retaining ring.

(d) Slide rear insulator back over wires to expose contact pins.

(e) Remove applicable contact pins (with horseshoe retainer) from forward insulator.

(f) Remove horseshoe retainer from contact pins and cut attaching wires.

(g) Remove stow switch, pulling attached wires through rear insulator, conduit adapter back shell, and wire harness conduit.

B. Installing Thrust Reverser Stow Switch

(1) Position stow switch on mounting bracket and adjust jam nuts to approximate final switch position.

(2) Install switch wires through wire harness conduit, conduit adapter back shell, and rear insulator.

(3) Cut stow switch wires to proper length, strip insulation, and crimp on new contact pins.

(4) Assemble pylon electrical disconnect plug as follows:

(a) Install horseshoe retainer on contact pin and insert pin in forward insulator.

(b) Slide rear insulator forward on wires, ensuring that locating keyway aligns with key in barrel of connector.

(c) Install rear insulator retaining ring in slot provided.

(d) Install conduit adapter back shell and conduit coupling nut.

(5) Connect pylon electrical disconnect plug and adjust stow position switch. (Refer to 78-30-4.)

(6) Install inboard fairing (stang) on thrust reverser assembly.

5. Thrust Reverser Lower Door Link Replacement

A. To remove lower door link(s), proceed as follows:

(1) Deploy thrust reverser and dump system hydraulic pressure.

(2) Open all thrust reverser circuit breakers on right hand console panel.

(3) Remove outboard fairing (stang).

(4) Install ground maintenance safety lock.

(5) Turn off all electrical power.

(6) Remove cotter pins, nuts, and bolts connecting lower door links to door and bogie.

(7) Remove door link from thrust reverser.

B. Install lower door link(s), as follows:

(1) Position door link and install bolts, washers, nuts, and safety with cotter pins.

(2) Remove ground maintenance safety lock.

(3) Install outboard fairing.

(4) Close all thrust reverser circuit breakers on right hand console panel.

(5) Operate thrust reverser to ensure that no binding occurs on door link.

6. Replacement of Spherical Bearings in Reverser Door Link or Reverser Door Support Fittings

A. When removing the spherical bearing (Rohr P/N 140-5410-1 or -3) from the reverser door support fittings, use a phenolic block, hardwood or any non-metal tool for initial rotation of the bearings. Metal tools will damage the bearing inner race.

(1) Remove reverser doors or drive links as required.

(2) Rotate the spherical bearing until the flat side of the bearing is parallel with the flat portion of the inner race or the flat portion of the link.

(3) Remove bearing from link or inner race.

(4) Clean bearing and link or inner race as necessary. Inspect inner race for corrosion.

(5) Inspect bearing for evidence of wear. If wear is prevalent, check bearing in accordance with figure 78-30-2A. Replace bearing as required.

(6) Apply a film of PT 605* lubricant or equivalent to bearing and inner bore of race or link.

(7) Position bearing so that flat side is parallel with flat portion of the inner race or flat portion of the link. Install bearing in link or door support fitting and rotate 90 degrees.

(8) Replace reverser doors or drive link assembly. (Refer to applicable procedures)

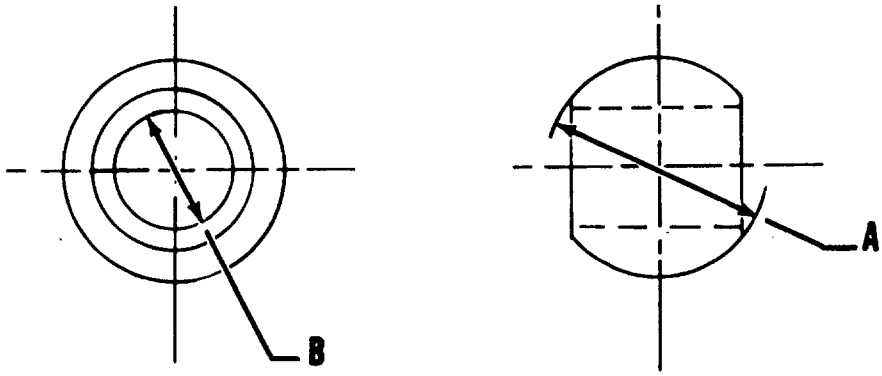
7. Replacement of Secondary Latch Solenoid

A. To remove secondary latch solenoid, proceed as follows:

(1) Remove stang; then remove the forward mounting screw that attaches the solenoid to the weld bracket.

(2) Remove the screw and mounting clip from the aft end of the solenoid. Disconnect clevis fitting from solenoid to "S" hook. Remove clevis fitting from solenoid.

*Procure from:
Product Techniques
1153 North Stanford
Los Angeles, California 90059



	MIN	MAX
Dimension A		
140-5410-1 BRG	0.5920	0.5930
140-5410-3 BRG	0.5915	0.5920
Dimension B	0.3140	0.3150

NOTE: 140-5410-3 bearing replaces 140-5410-1 bearing.

Removing Spherical Bearings
(Rohr P/N 140-5410-1 or -3)
Figure 78-30-2A

(3) Pull ferrule with outer braided conduit and inner teflon sleeving back from the electrical connector approximately 2 inches to expose the electrical wires.

NOTE: Some wire harness assemblies may have spirap tubing installed in place of the teflon sleeving.

(4) Unscrew coupling nut from the electrical connector.

(5) Disassemble thrust reverser electrical connector. (Refer to "Thrust Reverser Conduit Electrical Plug Assembly.")

(6) Cut contacts from solenoid wires at the electrical connector to eliminate the possibility of damage to other wiring insulation when pulling wires out of the conduit assembly.

(7) Cut one of the six electrical wires at the solenoid, leaving the wire in the braided conduit to be used as a "fish wire" to aid in installing the wiring for the new solenoid.

(8) Remove the solenoid by pulling the remaining five wires from the conduit assembly.

B. To install secondary latch solenoid, proceed as follows:

(1) Thread wire through electrical conduit using "fish wire" left during the removal procedures.

(2) Mount solenoid in bracket and install mounting clip and screws on aft end of solenoid.

(3) Install clevis fitting in solenoid. Do not connect clevis to "S" hook at this time. This will be accomplished during the rigging procedure.

(4) Assemble electrical plug. (Refer to "Thrust Reverser Conduit Electrical Plug Assembly.")

(5) Connect and safety electrical plug to pylon.

(6) Rig secondary latch solenoid.

(7) After rigging, check complete installation for proper safeties and security.

8. Thrust Reverser Conduit Electrical Plug Assembly (Disassembly and Assembly Procedures)

A. To disassemble electrical plug assembly, proceed as follows:

(1) Open engine cowling.

(2) Disconnect electrical plug from pylon.

(3) Disconnect conduit coupling nut.

(4) Remove conduit adapter backshell and slide back on wire.

(5) Remove rear insulator retaining ring.

(6) Slide rear insulator back over wires to expose contacts.

NOTE: Ensure that horseshoe retainer does not drop out.

(7) Remove applicable contacts (with horseshoe retainer) from forward insulator.

(8) Remove horseshoe retainer from contact.

(9) Cut off applicable contact and remove wire(s) through rear insulator and conduit adapter backshell.

B. To assemble electrical plug assembly, proceed as follows:

(1) Install electrical wires through conduit adapter backshell and rear insulator.

(2) Cut electrical wire to proper length strip and crimp on new contact (contact #330-0187-000).

(3) Install horseshoe retainer on contact and insert in forward insulator.

(4) Slide rear insulator forward on wires. Ensure that locating keyway aligns with key in barrel of connector.

(5) Install rear insulator retaining ring in slot provided.

(6) Install conduit adapter backshell and coupling nut. Tighten coupling nut.

(7) Install and tighten electrical plug to pylon.

(8) Perform an operational checkout of thrust reverser system.

9. Thrust Reverser Bogie Inspection - Airplanes Serial No. NA370-2 Through -6, -8 and -9 Not Changed by Service Bulletin 74-10

A. To perform thrust reverser bogie inspections, proceed as follows:

(1) Deploy thrust reversers.

(2) Open all thrust reverser circuit breakers on right-hand console and dump thrust reverser accumulator hydraulic pressure.

(3) Remove stang(s).

NOTE: Identify bogie and bogie guide tubes to insure proper position during the installation procedures.

(4) Disconnect door links from bogie.

(5) Disconnect hydraulic actuating cylinder from bogie; then remove bogie guide tubes.

(6) Remove bogie and clean with suitable solvent.

(7) Perform a magnetic particle inspection of bogie. If cracks are found, replace bogie.

(8) Position serviceable bogie and guide tubes between longeron brackets. Tighten guide tube bolts.

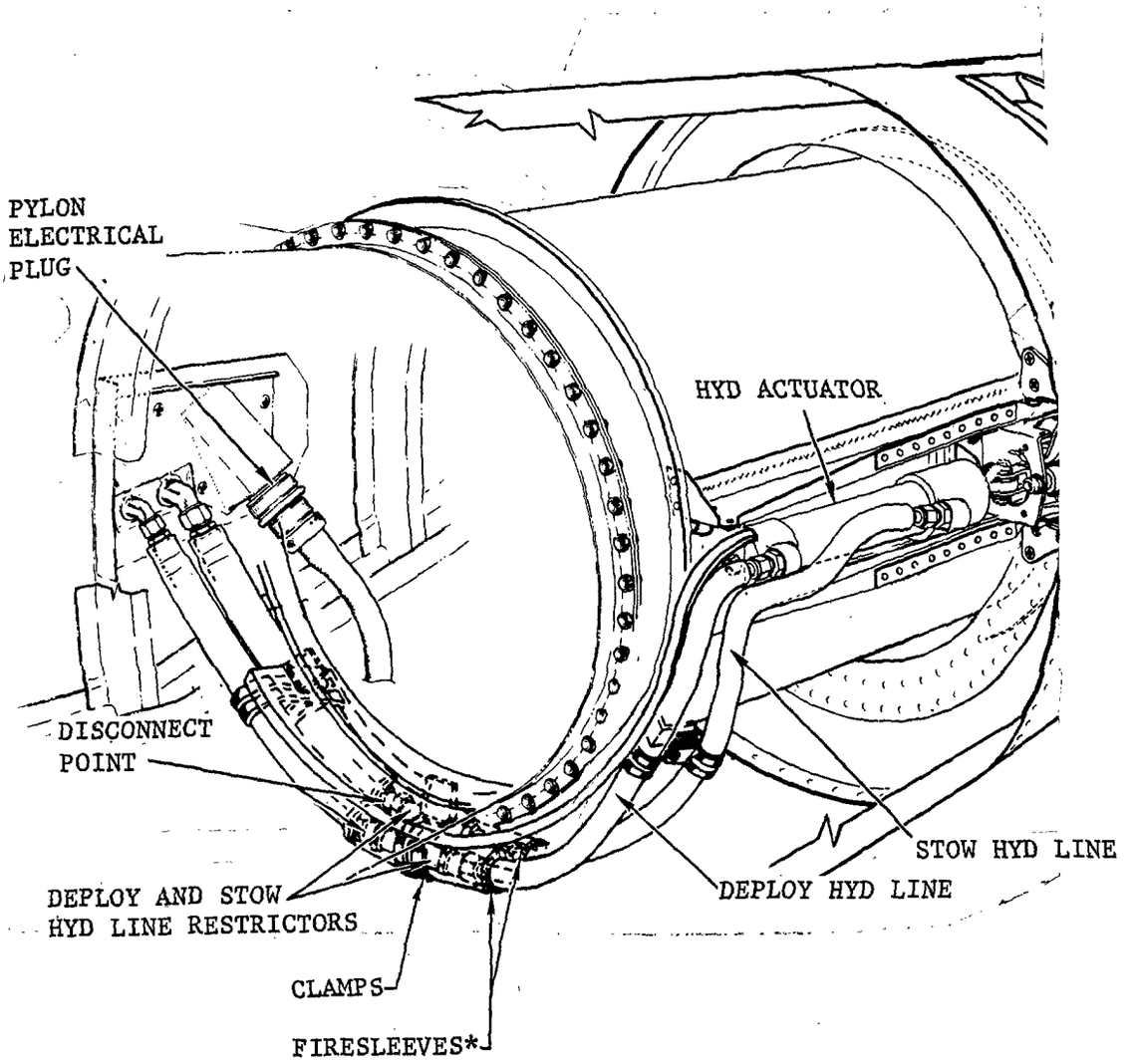
(9) Connect actuating cylinder and door links to bogie. Safety bolts.

(10) Lubricate reverser as required.

(11) Check thrust reverser rigging.

(12) Install stang(s).

(13) Perform operational checkout of thrust reverser system.



Thrust Reverser Hydraulic Line Connections
Figure 78-30-2B

MAINTENANCE PRACTICES (Continued)

RIGGING THRUST REVERSERS

WARNING

Subsequent to cycling the reversers, ensure that the thrust reverser control switch is at OFF or the hydraulic pressure is dumped. If the hydraulic pressure is not dumped, ensure that the thrust reverser control switch is left in the off position until a cycling operation is required.

WARNING

To prevent possible injury to personnel, on airplanes changed by S/B 74-8, the thrust reverser automatic stow system is activated any time the main landing gear is released from its downlock or whenever the landing gear position circuit breaker is opened. To prevent activation of this system, the emergency stow circuit breaker should be opened.

1. Deploy thrust reverser. Position THRUST REVERSER control switch to OFF.
2. Remove inboard and outboard stangs.
3. Remove spacer washers from door stop bolts. Reinstall and torque stop bolts to 30-40 inch-pounds.

CAUTION

Ensure that actuator piston rod is clear of all structure prior to and during retraction.

4. Remove bolt connecting actuator piston rod to bogie. Apply power to system and fully retract actuator piston rod. Position THRUST REVERSER control switch to OFF.
5. Disconnect solenoid clevis fitting from latch hook, then manually stow doors.
6. Install rigging tool (Rohr P/N 140-SK388-1) pins in holes provided in the inboard side of the upper and lower reverser doors. Install tool with jam nut up. Tighten jam nut. (See Figure 78-30-3.)

NOTES

- ***To avoid damage to bolt threads, do not overtorque jam nut.***
- ***If rigging tool is not available, see Figure 78-30-4 for fabrication details and material requirements.***

7. Measure and record gap between upper and lower inboard striker plates and stop bolts.
8. Remove rigging tool and manually deploy doors. Obtain a proper combination of washers (AN960C416L or S140-5482-1 - Laminated) to match recorded gap in step 7; then, install washers under door stop bolts. Torque stop bolts to 30-40 inch-pounds.
9. Install rigging tool and tighten jam nut. Check that stop bolts are in firm contact with striker plates.
10. Hold latch hook against upper striker plate and check clearance between aft face of hook and upper and lower striker plate. Clearance to be as shown in Figure 78-30-3. Add or remove shims between striker plate and door as required to obtain proper clearance.

NOTE

If adjustment is necessary, the rigging tool must be removed prior to removing striker plate. Reinstall rigging tool after adjustment of either striker plate.

11. Check end play of latch hook. Clearance between latch hook and washer is .010 to .050 inch. Use washers (S140W5482-2) as required to obtain clearance. Ensure that hook to striker plate clearance is maintained. (See Figure 78-30-6.)
12. Hold secondary latch hook against upper striker plate and adjust solenoid clevis fitting to permit a slip-fit of clevis bolt, then extend (lengthen) clevis fitting one full turn from the slip fit position. Torque clevis jam nut to 20-25 inch-pounds and bend tab of lock washer over jam nut.
13. Install solenoid clevis bolt, washers and nut. Tighten nut finger tight and install cotter pin.

NOTE

If the clevis fitting is not centered over the latch hook, loosen solenoid rear attach brackets and shift solenoid as required to center clevis. Tighten brackets after adjustment and lockwire screws.

14. Remove rigging tool and manually position doors (toward deploy position) until latch hook engages the striker plates.
15. Tape a 0.020 inch feeler gauge to stow switch actuating cam (lower portion of upper striker plate). Adjust stow switch (using the switch lock nuts) until the unlock light in cockpit just goes out.

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MAINTENANCE PRACTICES (Continued)

RIGGING THRUST REVERSER

NOTES

- ***Keep the stow switch lock nuts tight, subsequent to adjusting the stow switch in step 15.***
- ***Optimum adjustment of the stow switch is .020 inch — UNLOCK light out, .015 inch — UNLOCK light on.***

16. Remove 0.020 inch feeler gauge and insert a 0.015 inch feeler gauge between stow switch roller and cam. Check that unlock light in cockpit remains on. Repeat step 15 and 16 if necessary. Torque stow switch nuts to 34-40 inch-pounds. Lockwire nuts together.
17. Manually position doors to obtain a clearance of 3 to 5 inches from full deploy position. Measure clearance at vertical centerline of doors.
18. Adjust deploy switch as required until deploy light in cockpit illuminates. Tighten and lockwire deploy switch locknuts together.
19. Install rigging tool and tighten jam nut. With actuator fully retracted and locked, adjust actuator rod end to obtain "slip fit" of bogie to actuator attaching bolt. (Install attach bolt only.)
20. Remove rigging tool and partially deploy doors. Position THRUST REVERSER control switch to OFF.
21. Remove actuating cylinder rod end to bogie attach bolt and screw rod end in 1/2 turn and reinstall attach bolt. Torque rod end jam nut to 125-155 inch-pounds. Safety with lockwire.
22. Position THRUST REVERSER control switch to ON and stow thrust reverser. Position THRUST REVERSER control switch to OFF and dump accumulator hydraulic pressure. Check that striker plates are in firm contact with stop bolts.

NOTE

If the strikers are not in firm contact with the stop bolts the actuator rod end may be shortened an additional 1/2 turn (maximum) to obtain striker plates to stop bolt contact.

23. Install washers and nuts on actuator to bogie attach bolt. Torque nut to 60-85 inch-pounds. Install cotter pin.
24. Check complete installation for proper safeties and security.
25. Perform an operational test of thrust reverser system. Ensure that adequate clearance exists between any actuating and fixed component of the reverser mechanisms during operation. (See Figure 78-30-6 for clearances.)

NOTE

Thrust reverser system rigging should be checked and re-rigged as necessary subsequent to the initial engine run utilizing the thrust reversers.

26. Install inboard and outboard fairings.
27. Install upper engine access door and close lower access door (refer to 71-10-01).

THRUST REVERSER FLYAWAY LOCKS

Two flyaway locks (P/N 140-5443) are supplied as part of loose equipment with each airplane. If it is desired that the thrust reverser be mechanically locked out, the locks may be installed. The flyaway locks provide a positive mechanical block against thrust reverser deployment. The locks may be installed as follows:

1. With thrust reverser doors stowed, position battery and electrical master switches at OFF.
2. Remove outboard stang fairing from each engine.
3. Adjust flyaway lock to a short dimension and install between guide tube support bracket and bogie (see Figure 78-30-5).
4. With lock in position, extend threaded end until lock is tight.
5. Tighten jam nut and lock with safetywire.
6. Reinstall the two stang fairings.

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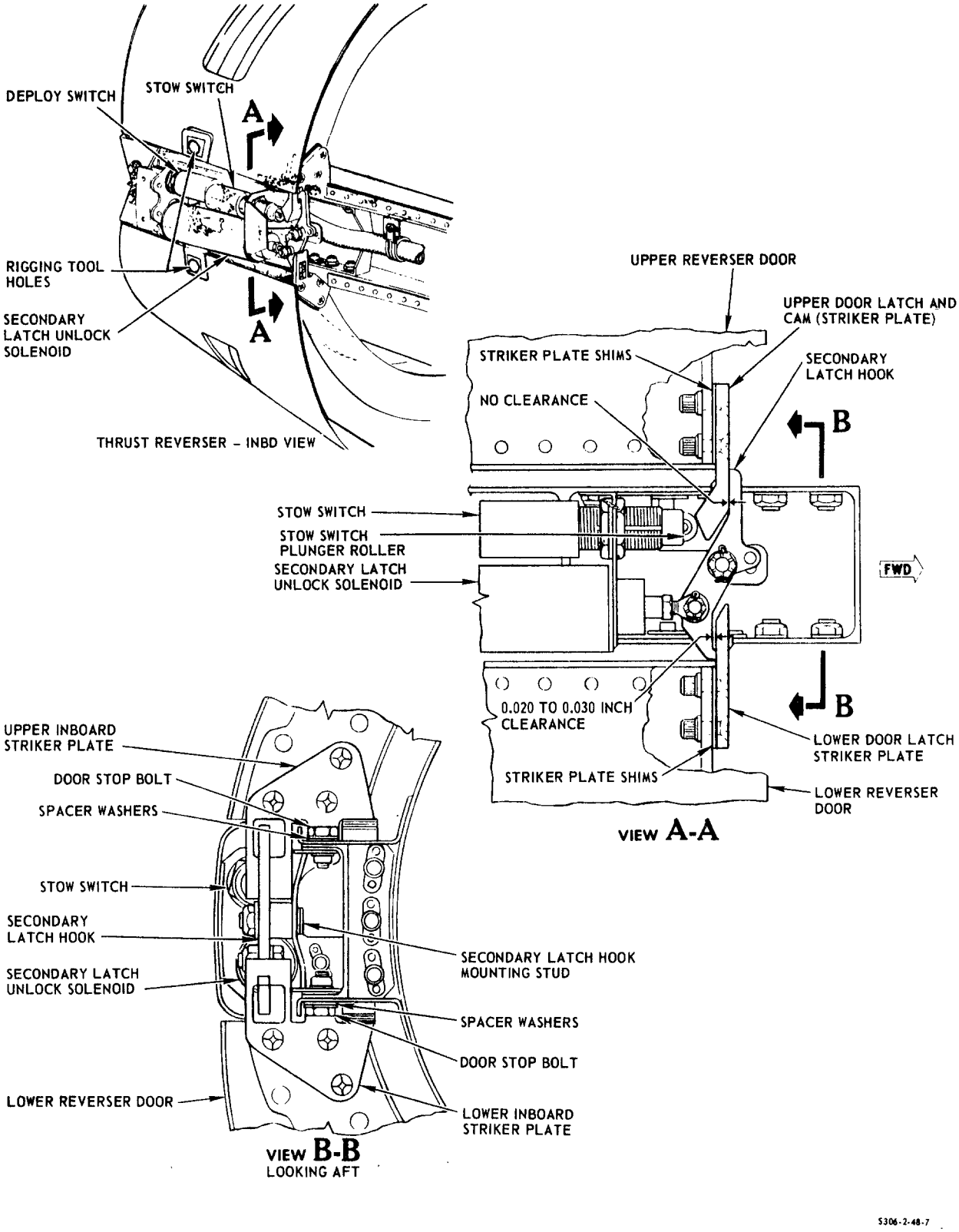

Figure 78-30-3 — Rigging Thrust Reversers

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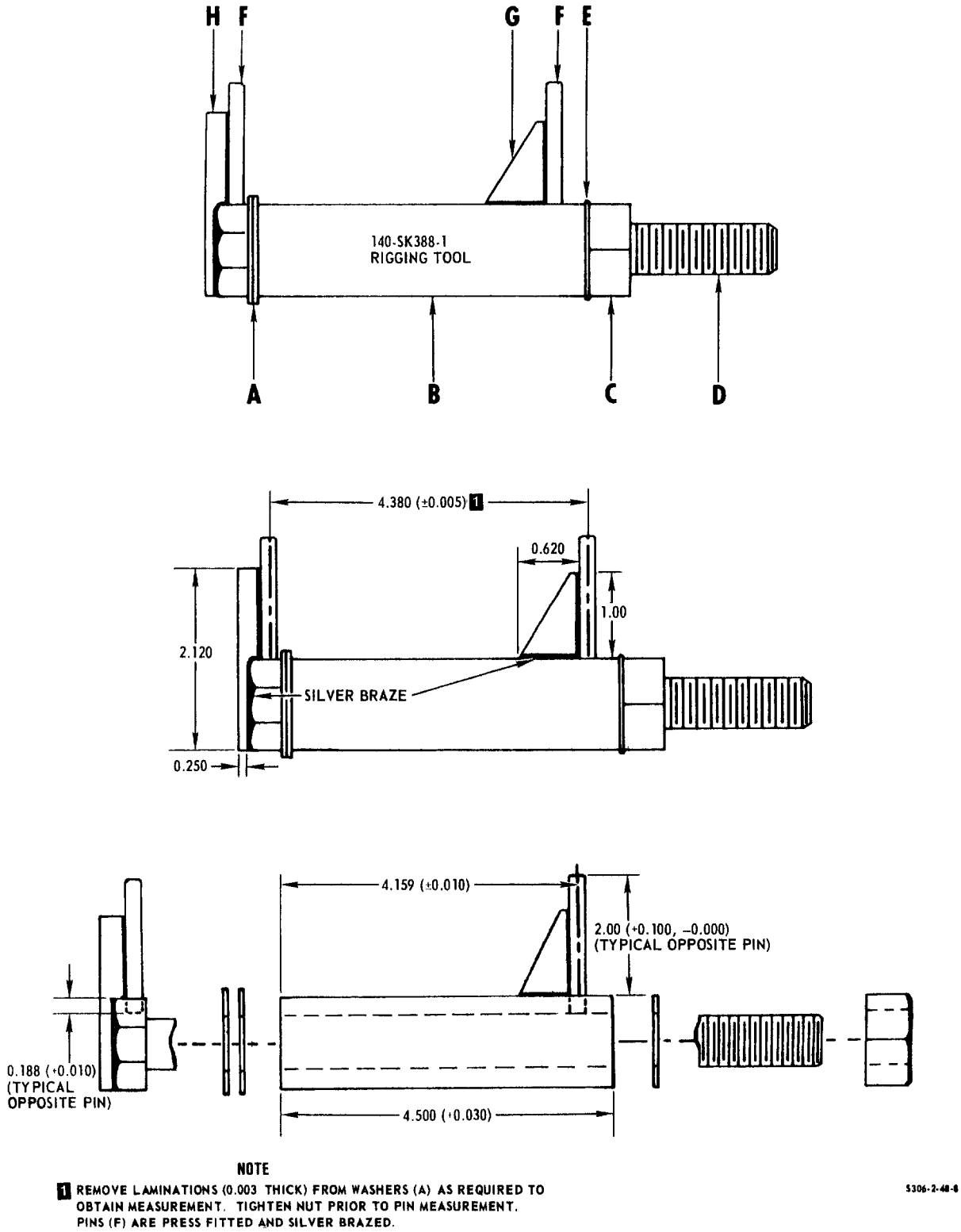

NOTE

1 REMOVE LAMINATIONS (0.003 THICK) FROM WASHERS (A) AS REQUIRED TO OBTAIN MEASUREMENT. TIGHTEN NUT PRIOR TO PIN MEASUREMENT. PINS (F) ARE PRESS FITTED AND SILVER BRAZED.

5306-2-48-8

Figure 78-30-4 — Thrust Reverser Rigging Tool (Sheet 1 of 2)

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LOCALLY MANUFACTURED RIGGING TOOL
MATERIAL REQUIREMENTS

ITEM	NOMENCLATURE	MATERIAL PART NO. (EQUIVALENT MATERIAL MAY BE USED)	NUMBER REQUIRED
A	Washer (Laminated) [1]	P/N 5722-43-5	2
B	Sleeve	Cres Tube Type 304 1¼" OD .188" WL	1
C	Nut	P/N AN 924C12	1
D	Bolt	P/N NAS 1012-88H	1
E	Washer	P/N AN 960C1216L	1
F	Pin	Make from .3120" Dia. Std. Dowel Pin	2
G	Gusset	Cres Bar 1/4 × 5/8 × 1 inch	1
H	Support	Cres Bar 1/4 × 5/8 × 2-1/8 inch	1

[1] May be procured from Laminated Shim Co. Inc. 1691 California Street, Corona, CA 91719

Figure 78-30-4 — Thrust Reverser Rigging Tool (Sheet 2 of 2)

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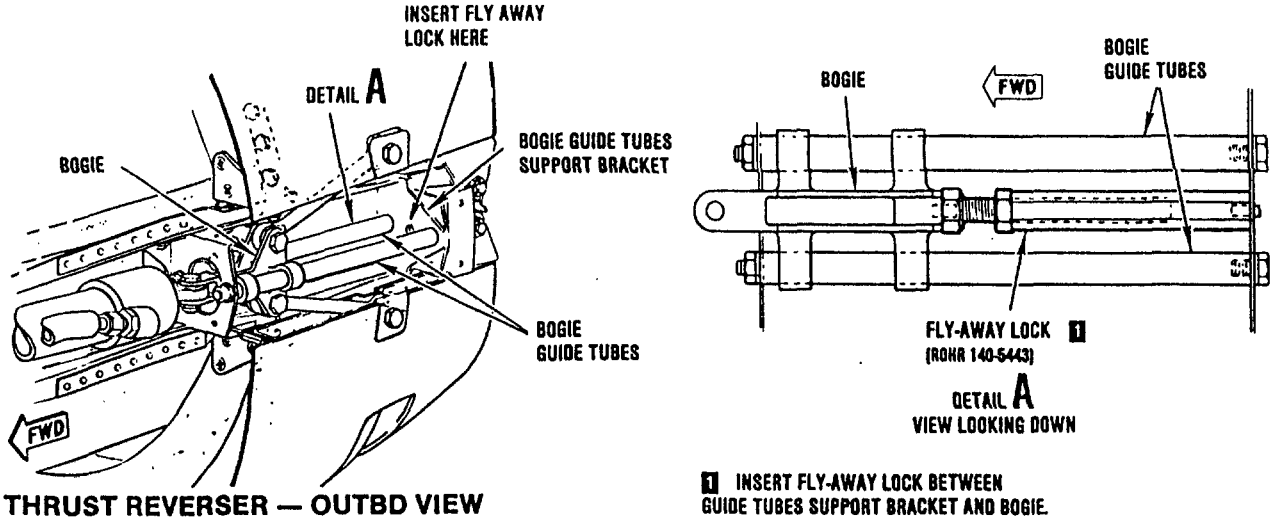

Figure 78-30-5 — Thrust Reverser Fly-Away Locks

NEW

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EXHAUST

MAINTENANCE PRACTICES

OPERATIONAL CHECK-OUT OF THRUST REVERSER SYSTEM

Before an operational check-out of the reverser system, the normal hydraulic power system must be properly serviced. If any hydraulic units and/or lines have been disconnected, the reverser system should be operated through five complete cycles to bleed air from the hydraulic actuator and lines. To perform an operational check-out of the reverser system proceed as follows:

NOTE

- ***An operational check-out of the reverser system must be made if a unit has been replaced, if hydraulic lines have been disconnected, or if an adjustment has been made on any unit. The check-out should be thorough enough to ensure that the system is free of hydraulic leaks and that no malfunction could occur because of improper maintenance.***
- ***The following procedures may be performed with the normal hydraulic power system electric pump; however, if several consecutive cycles are required for check-out, an external hydraulic test stand should be connected to the airplane.***

1. Check that reverser accumulator is properly serviced.
2. Connect 28 VDC external electrical power to airplane.

WARNING

Make sure that all personnel are clear of reverser doors before operating system.

3. Push in the following circuit breakers:
   A. HYD PUMP CONT
   B. THRUST REVERSERS — LATCH (two)
   C. THRUST REVERSERS — EMERG STOW
   D. THRUST REVERSERS — CONTROL (two)
   E. THRUST REVERSERS — IND LTS
4. Place THRUST REVERSER control switch at ON.
5. Place engine master switch at ON.
6. Place hydraulic pump switch at ON.
7. Set adjustable reverse lever stop to field elevation and temperature.
8. Advance both throttle levers above IDLE.
9. Attempt to move reverser control levers. The levers should not move.
10. Retard throttle levers to IDLE.
11. Check that both ARM (amber) indicator lights are on.
12. Move reverser control levers up and aft until lever movement is stopped.

NOTE

Reverser control lever movement will be stopped by the reverser control lever unlock solenoid. further movement of the levers can be accomplished only after the reverser doors are fully deployed.

13. Check that both UNLOCK (red) indicator lights are on.
14. Check that reverser doors are fully deployed. This is indicated by illumination of both DEPLOY (blue) indicator lights.
15. Check that reverser control lever can be moved to full aft position.
16. With reverser control lever in full aft position, make sure that throttle levers cannot be moved forward out of IDLE position.
17. Place reverser control levers full downward. Reverser doors should stow.
18. Check that DEPLOY and UNLOCK indicator lights go off.
19. Move throttle levers to OFF position. check that ARM indicator light goes out.
20. Return all switches to their normal or OFF position.

OPERATIONAL CHECK-OUT OF THRUST REVERSER FAIL SAFE SYSTEM

1. Check that HYDRAULIC PUMP switch is OFF.

CAUTION

Too rapid dumping of the accumulators may exceed the reservoir pressure relief valve rate and cause possible damage.

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MAINTENANCE PRACTICES (Continued)

OPERATIONAL CHECK-OUT OF THRUST REVERSER FAIL SAFE SYSTEM (Cont'd)

2. Relieve thrust reverser system hydraulic accumulator pressure.
3. Remove outboard and inboard fairings. (stangs) on left and right thrust reversers.
4. Install ground maintenance safety locks on left and right thrust reversers (Rohr P/N 140-5444).
5. Apply external 28 VDC electrical power to airplane.
6. Open the LH FUEL-BOOST PUMP, RH FUEL-BOOST PUMP, LH ENGINE-IGN & BOOST CONT, and RH ENGINE-IGN & BOOST CONT circuit breakers.
7. Check that LDG GEAR-POS and all thrust reverser circuit breakers are engaged (pushed in).
8. Position LH-ENGINE MASTER and RH-ENGINE MASTER switchs to ON.
9. Position left and right engine throttle levers to the IDLE position.
10. Position THRUST REVERSER switch to ON.
11. Manually actuate left thrust reverser deploy switch (see Figure 78-30-3). Check that LH THRUST REVERSER DEPLOY idicating light is illuminated. Check that LH THRUST REVERSER auxiliary throttle cannot be moved past the first detent (full RPM position).
12. Release left thrust reverser deploy switch. Return LH THRUST REVERSER auxiliary throttle to the stow position.
13. Manually actuate right thrust reverser deploy switch. Check that RH THRUST REVERSER DEPLOY blue indicating light is illuminated. Check that RH THRUST REVERSER auxiliary throttle cannot be moved past the first detent (full RPM position).
14. Release right thrust reverser deploy switch. Return RH THRUST REVERSER auxiliary throttle to the stow position.
15. Position THRUST REVERSER switch to OFF.
16. Position left and right engine throttle levers to the cut-off position.
17. Position LH-ENGINE MASTER and RH-ENGINE MASTER switches to off.
18. Engage (push in) LH ENGINE-IGN & BOOST CONT, RH ENGINE-IGN & BOOST CONT, LH FUEL-BOOST PUMP and RH FUEL-BOOST PUMP circuit breakers.
19. Remove external electrical power from airplane.
20. Remove ground maintenance safety locks from left and right thrust reversers.
21. Install outboard fairings (stangs) on left and right thrust reversers.

PREFLIGHT CHECK (PRIOR TO FIRST FLIGHT OF DAY) OF THRUST REVERSER SYSTEM — S/N 370-2 THRU -6, -8 and -9

1. Disengage LH and RH FUEL BOOST PUMP and LH and RH ENGINE-IGNITION and BOOST CONTROL circuit breakers.
2. Ensure that hydraulic system is properly serviced and that the thrust reverser ground maintenance safety locks are removed. Engage all thrust reverser system circuit breakers.
3. Check THRUST REVERSER control switch for freedom of operation between switch position. Position switch at OFF.
4. Apply electrical power to airplane.

WARNING

Make sure that all personnel are clear of reverser doors before actuating thrust reverser system.

5. Position HYD PUMP control switch to AUTO.
6. Position THRUST REVERSER control and ENGINE MASTER switchs to ON.
7. Position both throttle levers to IDLE. Check that both ARM lights (two) are illuminated.
8. Move thrust reverser levers up to interlock. Check that UNLOCK and DEPLOY lights (four) are on. The two unlock lights will come on first, immediately followed by the deploy lights. With a slight aft pressure on the thrust reverser levers, check that interlocks release when the deploy light comes on.
9. Position THRUST REVERSER control switch to EMERG STOW. Check that ARM, UNLOCK and DEPLOY Lights (six) are out. The two arm lights will go out first, immediately followed by the two DEPLOY and two UNLOCK lights.
10. Position THRUST REVERSER control switch to ON. Check that two ARM lights come on first, immediately followed by the two UNLOCK lights and the DEPLOY lights.
11. Position thrust reverser control levers to full forward and down position (stowed position). Check that DEPLOY and UNLOCK lights (four) go out in proper sequence.

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MAINTENANCE PRACTICES (Continued)

PREFLIGHT CHECK (PRIOR TO FIRST FLIGHT OF DAY) OF THRUST REVERSER SYSTEM — S/N 370-2 THRU -6, -8 AND -9

12. Position throttle control levers forward out of IDLE position. Check that ARM lights (two) go out. Retard throttle control levers to IDLE and check that the two ARM lights come on.
13. Position THRUST REVERSER control switch to OFF. Check to ensure that the two ARM lights go out.
14. Position ENGINE MASTER and HYDRAULIC PUMP switches to OFF. Position throttle control levers to cut-off.
15. Engage (push in) LH and RH FUEL BOOST PUMP and LH and RH ENGINE-IGNITION and BOOST CONTROL circuit breakers.
16. Visually inspect secondary latch hook for proper engagemnt in the striker plates. Ensure that secondary latch hook and striker plates are not bent.

NOTE

After accomplishing this preflight check, the thrust reverser system should not be operated prior to the first flight.

THRUST REVERSER 150 HOUR INSPECTION — S/N 370-2 THRU -6, -8 AND -9

Inspect the following thrust reverser components for security, cracks, hot spots, warping, loose fasteners and applicable safeties. This inspection is to be performed with the inboard and outboard fairings (stangs) removed.

1. Upper and lower doors.
2. Reverser door drive links and mounting bolts.
3. Bogies and bogie guide rods.
4. Door striker plates, stop bolts, latch hooks and springs.
5. Interior and exterior surface of exhaust nozzle and reverser doors.
6. Reverser door attach fittings and bolts.
7. Inboard and outboard stang (fairing) brackets.
8. Exhaust nozzle ring segments (mice).
9. Forward inboard stang bracket assembly (Rohr P/N 140-5421-17 and -18).
10. Visible portion of upper and lower reverser door longerons (Rohr P/N 140-5437-1, -2 and 5438-1 and -2).
11. Outboard reverser door support fitting (Rohr P/N 140-5439-1).
12. Inboard reverser door support fitting (Rohr P/N 140-5440-1).
13. Secondary latch solenoid, stow and deploy switches and electrical wiring for security, chafing, cleanliness and safeties.
14. Check for interference or binding between the following items: (See Figure 78-30-5.)
    A. Striker plates and secondary latch hook.
    B. Striker plates and longeron.
    C. Reverser fairings (stangs) and reverser doors. Stangs must be installed to perform this check.

OPERATIONAL CHECK OF THRUST REVERSER INTERLOCK RELAYS — S/N 370-2 THRU -6, -8 AND -9

NOTE

This check may be accomplished subsequent to the landing gear operational check.

1. With the airplane on jacks, dump the main and thrust reverser accumulators. On airplanes changed by SOK 11, dump auxiliary system accumulator.
2. Open LH and RH FUEL BOOST PUMP and LH and RH IGNITION and BOOST CONTROL circuit breakers. Open STALL WARNING — INBD, OUTBD VANE HEATER circuit breakers.
3. Apply 28 VDC electrical power to airplane. Position throttles to IDLE. Position LH and RH ENGINE MASTER switches to ON.
4. Position THRUST REVERSER control switch to ON. Check that both ARM lights are ON.
5. Position LDG GEAR control handle to UP. Position HYDRAULIC PUMP control switch to AUTO. As soon as the RH and LH main gear are relased from the downlocks, reposition HYDRAULIC PUMP control switch to OFF. The thrust reverser ARM indicating lights (two) should be out.
6. Open (pull out) LDG GEAR — CONT circuit breaker.
7. Manually lock the LH main landing gear. Arm lights should be out.
8. Manually lock the RH main gear. Both ARM indicating lights should illuminate.
9. Close (push in) LDG GEAR-CONT circuit breaker.
10. Repeat step 5 to check the opposite interlock relay.
11. Open (pull out) LDG GEAR-CONT circuit breaker.
12. Manually lock the RH main landing gear. ARM lights should be out.

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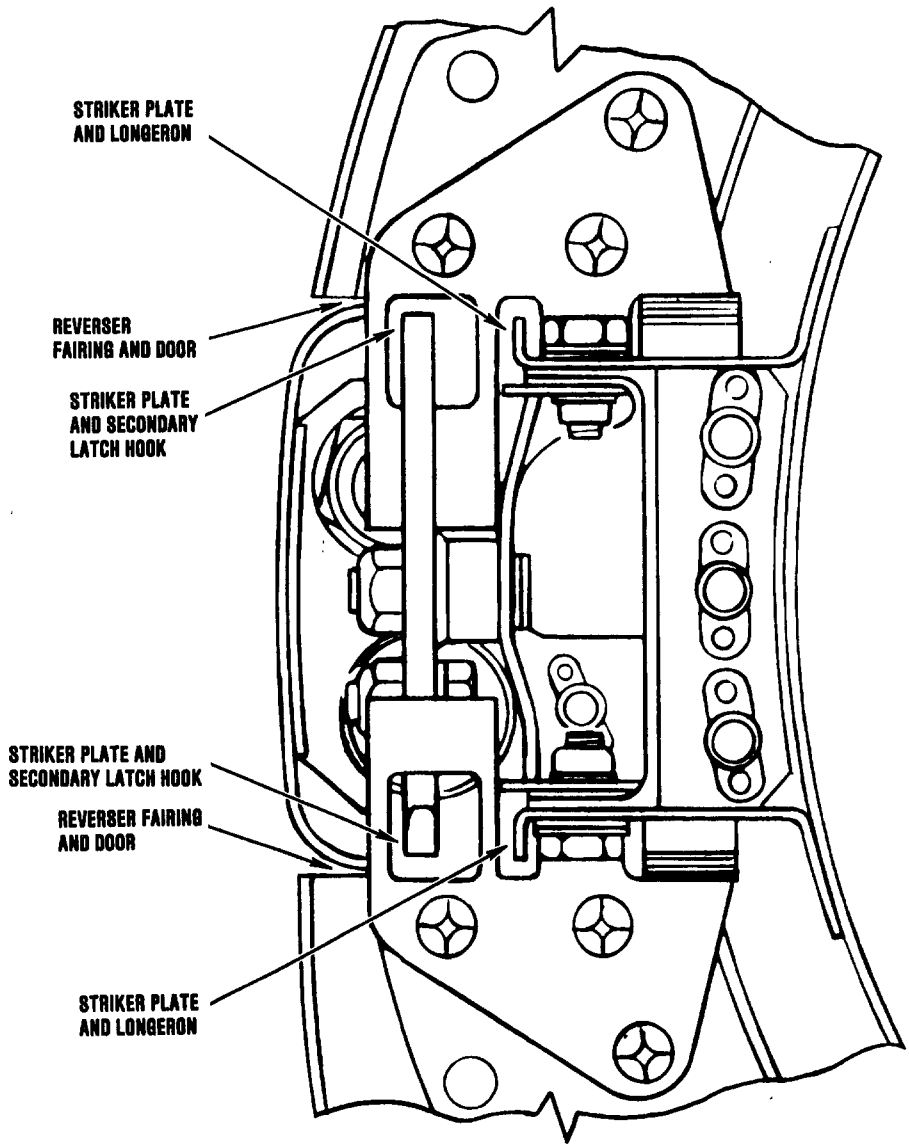
STRIKER PLATE AND LONGERON
REVERSER FAIRING AND DOOR
STRIKER PLATE AND SECONDARY LATCH HOOK
STRIKER PLATE AND SECONDARY LATCH HOOK
REVERSER FAIRING AND DOOR
STRIKER PLATE AND LONGERON

Figure 78-30-5 — Thrust Reverser (S/N 370-2 thru -6, -8, and -9)

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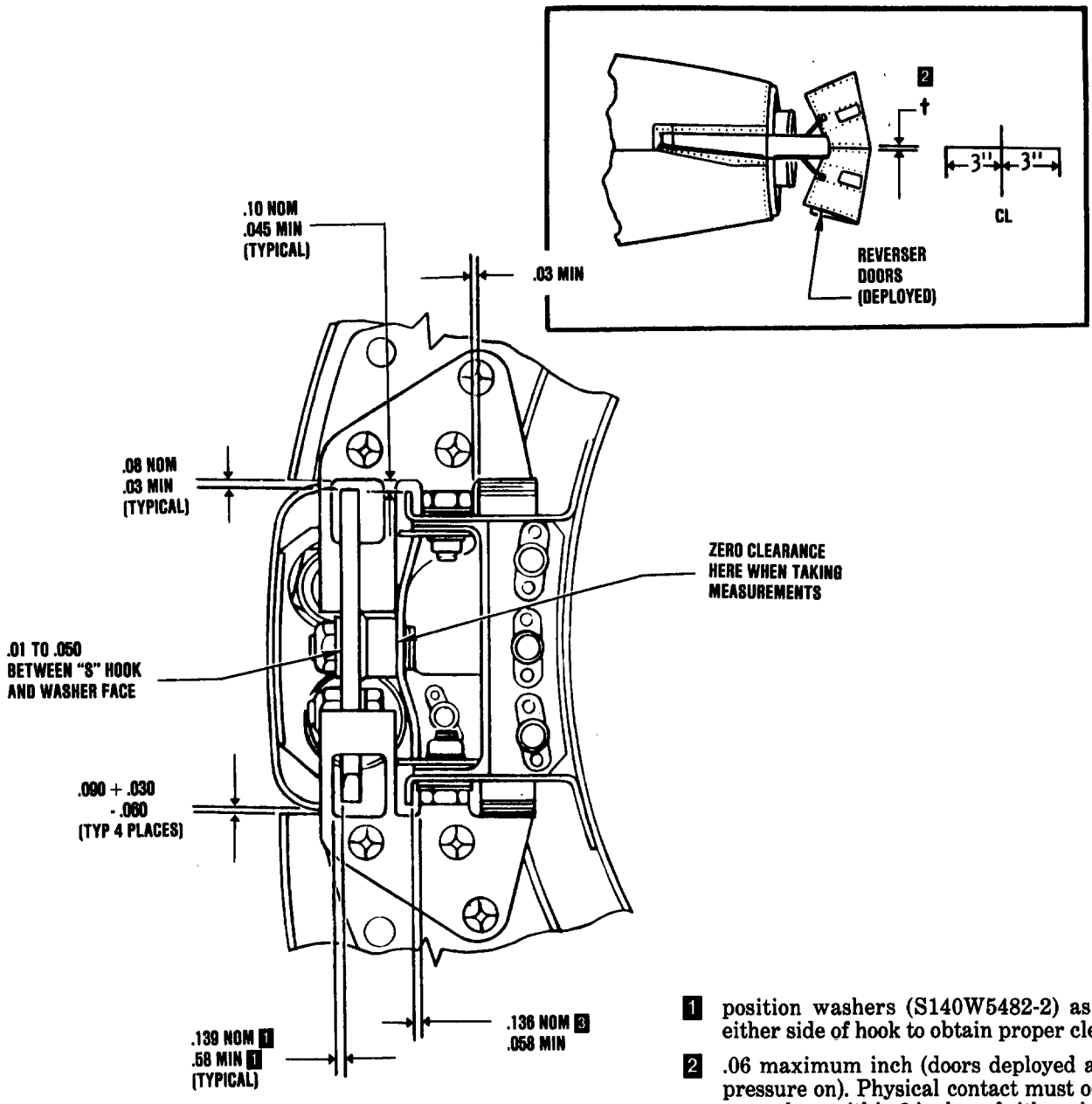

1 position washers (S140W5482-2) as required on either side of hook to obtain proper clearance.

2 .06 maximum inch (doors deployed and hydraulic pressure on). Physical contact must occur at CL or someplace within 3 inches of either side of CL. Minimum gap of .02 inch must be maintained for a distance of 3 inches along trailing edge from outer ends of doors. Rework channel (140-5402-37 and 140-5403-25) as required to maintain the .02 inch clearance.

3 Reverser doors to be positioned outboard as far as possible prior to taking this measurement.

Figure 78-30-6 — Thrust Reverser Clearance

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MAINTENANCE PRACTICES (Continued)

OPERATIONAL CHECK OF THRUST REVERSER INTERLOCK REPLAYS — S/N 370-2 THRU -6, -8 and -9 (Cont'd)

13. Manually lock the LH main gear. Both ARM lights should illuminate.
14. .Position LDG GEAR control handle to DOWN. Close LDG GEAR-CONT circuit breaker. Position HYDRAULIC PUMP control switch to AUTO. Check that the landing gear position indicator lights are ON.
15. Position power levers to cut-off. Position THRUST REVERSER control switch to OFF and ENGINE MASTER switches to OFF. Check that all circuit breakers are engaged.
16. Lower airplane and remove jacks.

THRUST REVERSER RIGGING CHECK

WARNING

Electrical and hydraulic power are required during the inspection procedures. Ensure that all personnel are clear of reverser doors before performing a cycling operation.

WARNING

To prevent possible injury to personnel, on airplanes changed by SB #74-8, the thrust reverser automatic stow system is activated any time the main landing gear is released from its downlock or whenever the landing gear position circuit breaker is opened. To prevent activation of this system, the emergency stow circuit breaker should be opened.

1. Disengage LH and RH ENGINE — IGNITION and BOOST CONTROL and LH and RH FUEL BOOST PUMP circuit breakers.
2. Engage all (six) thrust reverser circuit breakers and on airplanes changed by SB #74-8 engage the landing gear position circuit breaker.
3. Position LH power lever to IDLE.
4. Dump thrust reverser accumulator hydraulic pressure.
5. Remove inboard and outboard stangs (fairings) of each reverser.

NOTE

- ***If any rigging discrepancies and/or door stop bolt adjustments are required during the following rigging check, the complete thrust reverser assembly shall be rigged in accordance with the rigging procedures contained in this chapter.***
- ***The thrust reverser system should be deactivated (fly-away lock installed) in interference or binding exists between any actuating and fixed components of the thrust reverser mechanism.***

6. Install rigging tool (Rohr P/N 140-SK388-1) pins in holes provided on the inboard side of the upper and lower reverser doors. Tighten jam nut. Check that inboard striker plates are in firm contact with inboard stop bolts.
7. Check clearance between secondary latch hook and striker plates (see Figure 78-30-6 and 78-30-3).
8. Check secondary latch hook and striker plates for distortion, binding and cracks. Check for proper alignment of secondary latch hook to striker plates.
9. Apply electrical power to airplane. Position LH ENGINE MASTER switch and THRUST REVERSER CONTROL switch to ON. Check that ARM light is illuminated.
10. Remove rigging tool and apply hydraulic power to airplane and deploy thrust reverser. Check that all indicator lights are illuminated. Position HYDRAULIC PUMP control switch of OFF and remove electrical power from airplane.
11. Dump thrust reverser accumulator pressure.
12. Disengage THRUST REVERSER LATCH circuit breaker and on airplanes changed by SB #74-8, open EMERG STOW circuit breaker.
13. Manually position reverser doors (toward stow position) until the door striker plates engage the secondary latch hook. Apply elctrical power to airplane and check that the UNLOCK light in cockpit is illuminated.
14. Insert 0.015 inch feeler gauge between the stow switch roller and upper reverser door striker cam. Check that the UNLOCK light remains ON. Remove the 0.015 inch feeler gauge and replace with a 0.020 inch gauge. Check that the UNLOCK light goes out.
15. Engage THRUST REVERSER LATCH circuit breaker.
16. While monitoring DEPLOY light in cockpit, manually disengage secondary latch hook from striker plates and slowly move reverser doors toward the deploy position until the DEPLOY light in cockpit comes on. Measure clearance form vertical centerline of reverser doors. Clearance should be 3 to 5 inches.

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MAINTENANCE PRACTICES (Continued)

THRUST REVERSER RIGGING CHECK (Cont'd)

17. Manually depress stow switch. Check that UNLOCK light goes out. While depressing the stow switch, manually actuate the secondary latch hook to the unlock position. Check that the UNLOCK light illuminates. Release switches.
18. Check that striker plate shims do not protrude into the secondary latch hook openings of the striker plate. If shims protrude remove excess shim material as required. (See Figure 78-30-3.) Check secondary latch hook for binding and striker plates for cracks.
19. Position LH thrust reverser control lever to stowed position.
20. Apply hydraulic power to airplane. Thrust reverser should stow. Check that DEPLOY and UNLOCK lights are extinguished. ARM light should be illuminated.
21. Deploy thrust reversers. During the deploy cycle, ensure that secondary latch hook does not catch in the upper and/or lower plates. On airplanes changed by SB #74-8 close EMERG STOW circuit breakers.
22. Position THRUST REVERSER control switch to ON and LH power control lever to cut off. Check that all thrust reverser indicator lights are out.
24. Position THRUST REVERSER and HYDRAULIC PUMP control switch to OFF and remove electrical and hydraulic power form airplane. Discharge reverser accumulator hydraulic pressure.
25. Install ground maintenance safety locks on left and right thrust reverser (Rohr P/N 140-5444) and on airplanes changed by S/B #74-8, open EMER STOW circuit breakers.
26. Apply electrical power to airplane.
27. Position LH ENGINE MASTER switch to ON.
28. Advance left engine power lever to the idle position.
29. Position THRUST REVERSER control switch to ON (ARM light should be illuminated).
30. Manually actuate left thrust reverser deploy switch. Check the following.

    LH DEPLOY (blue) indicating light illuminated.

    LH auxiliary throttle should not move past the first detent.
31. Release left thrust reverser deploy switch. Return LH thrust reverser control lever to the stowed position.
32. Position THRUST REVERSER control switch to OFF.
33. Position left engine power lever to cut-off position.
34. PositionLH ENGINE MASTER switch to OFF and remove electrical power from airplane. Remove ground maintenance safety lock.
35. Engage LH ENGINE-IGNITION & BOOST CONT, RH ENGINE-IGNITION & BOOST CONT, LH FUEL BOOST PUMP, RH FUEL-BOOST PUMP and EMER STOW (if applicable) circuit breakers.
36. Install outbd and inbd fairings (stangs) on left and right thrust reversers.
37. Repeat procedures on the RH engine thrust reverser.

OPERATIONAL CHECK OF THRUST REVERSER AUTOMATIC STOW SYSTEM — AIRPLANES CHANGE BY S/B #74-8

WARNING

Ensure that all personnel are clear of reverser doors prior to operating system.

1. Open LH and RH fuel boost pump and LH and RH ENGINE IGNITION and BOOST CONTROL and STALL warning — INBD, OUTBD VANE HEATERS circuit breakers.
2. Apply electrical and hydraulic power to airplane.
3. Position power levers IDLE and LH and RH ENGINE MASTER switches to on.
4. Position THRUST REVERSER CONTROL switch to on and deploy thrust reversers.
5. Position THRUST REVERSER CONTROL switch to OFF.
6. Open LDG POS circuit breaker.
7. Check that thrust reversers stow and the THRUST REVERSER UNSAFE light illuminates.
8. Position reverser control levers to stow.
9. Close LDG POS circuit breaker.
10. Momentarily open EMER STOW circuit breaker. Check that reverser unsafe light goes out.
11. Check thrust reverser system for normal operation.
12. Position power levers to cut-off. Position all switches and circuit breakers in their normal or airplane parked position.

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NOTICE

The following information applies to NA 265-80 airplanes only.

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	441	Jan 10/86
	442	Jan 10/86
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EXHAUST - DESCRIPTION

1. General

The exhaust system directs high-velocity gases from the engine turbine section aft and overboard to provide engine thrust. The thrust reversers divert the exhaust gases forward to decelerate the airplane during the land roll. For information relative to the exhaust section of the engine, refer to General Electric Maintenance Manual, SEI-187.

THRUST REVERSER SYSTEM - DESCRIPTION AND OPERATION

1. General

The engine thrust reverser system is used to decelerate the airplane during the landing roll by diverting the jet exhaust gases forward. The reversing elements consist of cascades and blocker doors which are nested between the exhaust nozzle and cowling during normal engine operation, but which are extended to a position aft of the exhaust nozzle to provide reverse thrust. The thrust reverser system is comprised of two independently operated reverser systems; one for each engine. Each system consists primarily of one half of a dual cockpit control assembly, a thrust reverser assembly, a pneumatic actuation system, and the necessary wiring and pneumatic plumbing. The reverser system is electrically controlled and pneumatically operated using engine bleed air. (See figures 78-30-1, 78-30-2 and 78-30-5). The LH and RH thrust reversers are functionally identical, so only the LH thrust reverser is covered in this chapter.

Even though the LH and RH thrust reversers are functionally identical, some physical differences between LH and RH sides do exist. Outboard is outboard on both LH and RH, but the top of the LH thrust reverser is the same as the bottom of the RH thrust reverser except for the pneumatic motor and drive mechanism. The pneumatic motor and drive are mounted on the bottom on both LH and RH reversers.

2. Cockpit Control System

The cockpit control system consists of a reverser lever assembly pivoted on the power lever, above the quadrant. The reverser lever is equipped with an electrical solenoid lockout which prevents the lever from being positioned in the full reverse thrust position at any time the reverser is in any position other than fully deployed. The power lever is mechanically locked in the IDLE position when the reverser lever is raised toward the deployed position. Both the reverser and power levers control thrust reverser operation through actuation of switches mounted in the quadrant. The reverser lever increases engine thrust through mechanical linkage attached to the power lever linkage. The thrust reverser should be fully deployed between 64% to 68% engine RPM (at interlock stop).

3. Thrust Reverser Assembly

The thrust reverser assembly consists of the translating structure and the aft end of the engine nacelle to which the translating structure is attached. (See figure 78-30-1). The translating structure consists of the cascade assembly, two translating beams, two rack assemblies, two beam/rack latch assemblies, two yoke assemblies, two clamshell type blocker door assemblies, and the rear structure assembly with pylon fairing. The cascade assembly consists of a stacked series of twelve concentric vanes attached to the structure. The translating beams are

installed on each side of the cascade assembly and a rack assembly is installed in a guide in each beam. A yoke assembly is attached to the inside of each translating beam and rack assembly and to the two blocker doors. The two blocker doors are installed in the rear structure and are connected to tension links which form the door pivot points. Linkage connected from bellcranks on the doors to the yoke assembly provide leverage to open and close the doors during initial movement of the translating structure to the stowed position or during the final travel toward the deployed position. The rear structure is attached to the aft end of the cascade assembly. When stowed, the rear structure forms the engine exhaust nozzle and when deployed provides the means for diverting engine thrust forward through the cascade vanes.

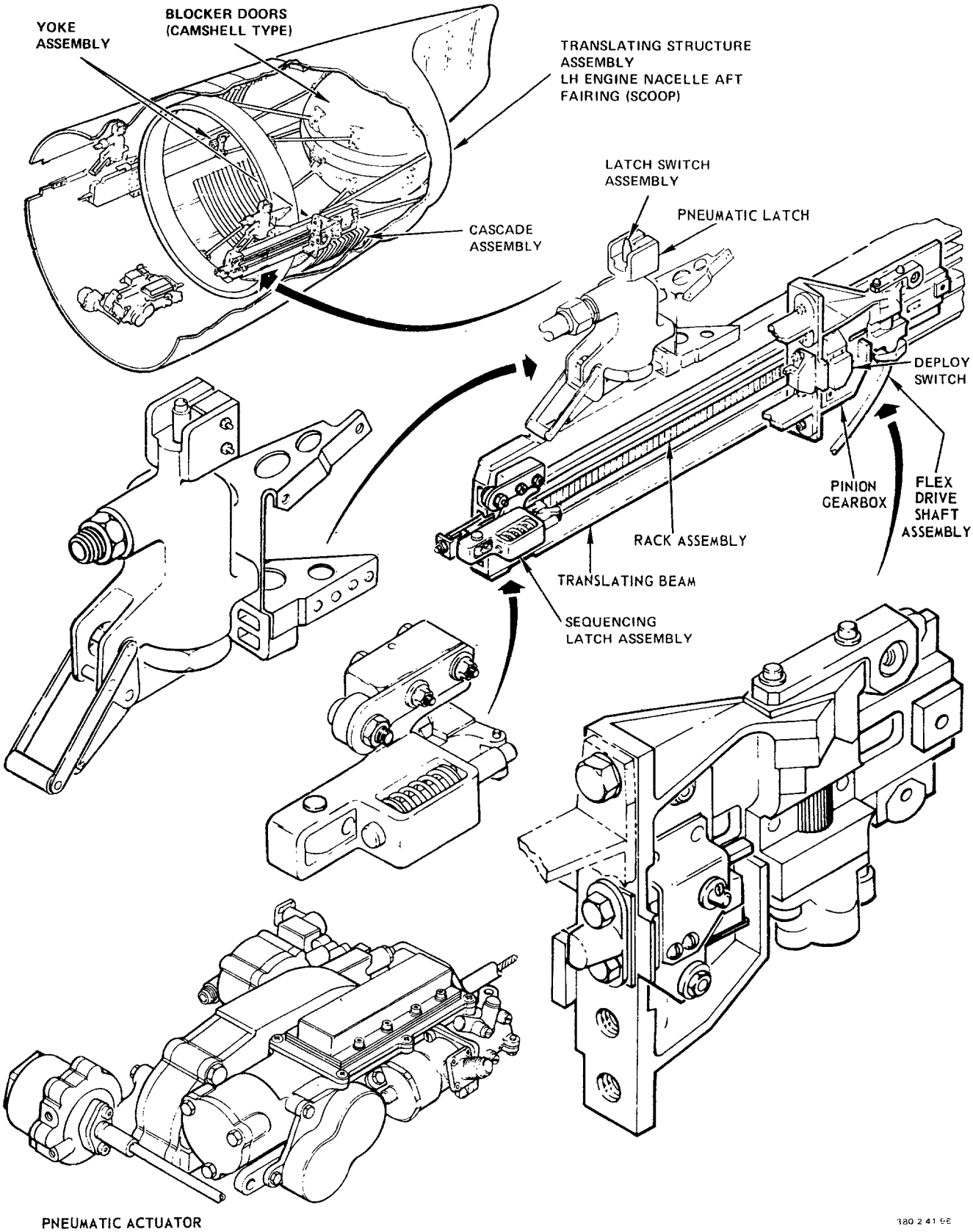
4. Pneumatic Actuation System

The actuation system components are installed mainly in the rear of the engine nacelle forward of the translating structure. The system consists of two pneumatic latches with a switch assembly, an inboard pinion gearbox, an outboard pinion gearbox with a deploy switch assembly, two flex drive shaft assemblies, and a pneumatic drive actuator with a dual output reduction gearbox. The gearbox ratio is 48.08 to 1. The pneumatic actuator assembly provides rotary power input to the gearbox. At manifold pressures of 17 to 19 psig, the motor rotates between 14,000 and 16,000 RPM with a torque of 140 inch pounds. The actuator air line installation provides for bleed air flow from the engine (8th stage bleed air) to the manifold. It also provides bleed air pressure to the pneumatic latches through a solenoid operated latch valve.

The thrust reverser indicating system consists of the following warning lights; LH T/R UNLOCK, LH T/R DEPLOY, LH T/R ARM and a T/R UNSAFE*. The airplane master caution light functions with the LH T/R ARM and LH T/R UNLOCK with the landing gear in the up position.

*NA380-5,-10,-12 and later airplanes.

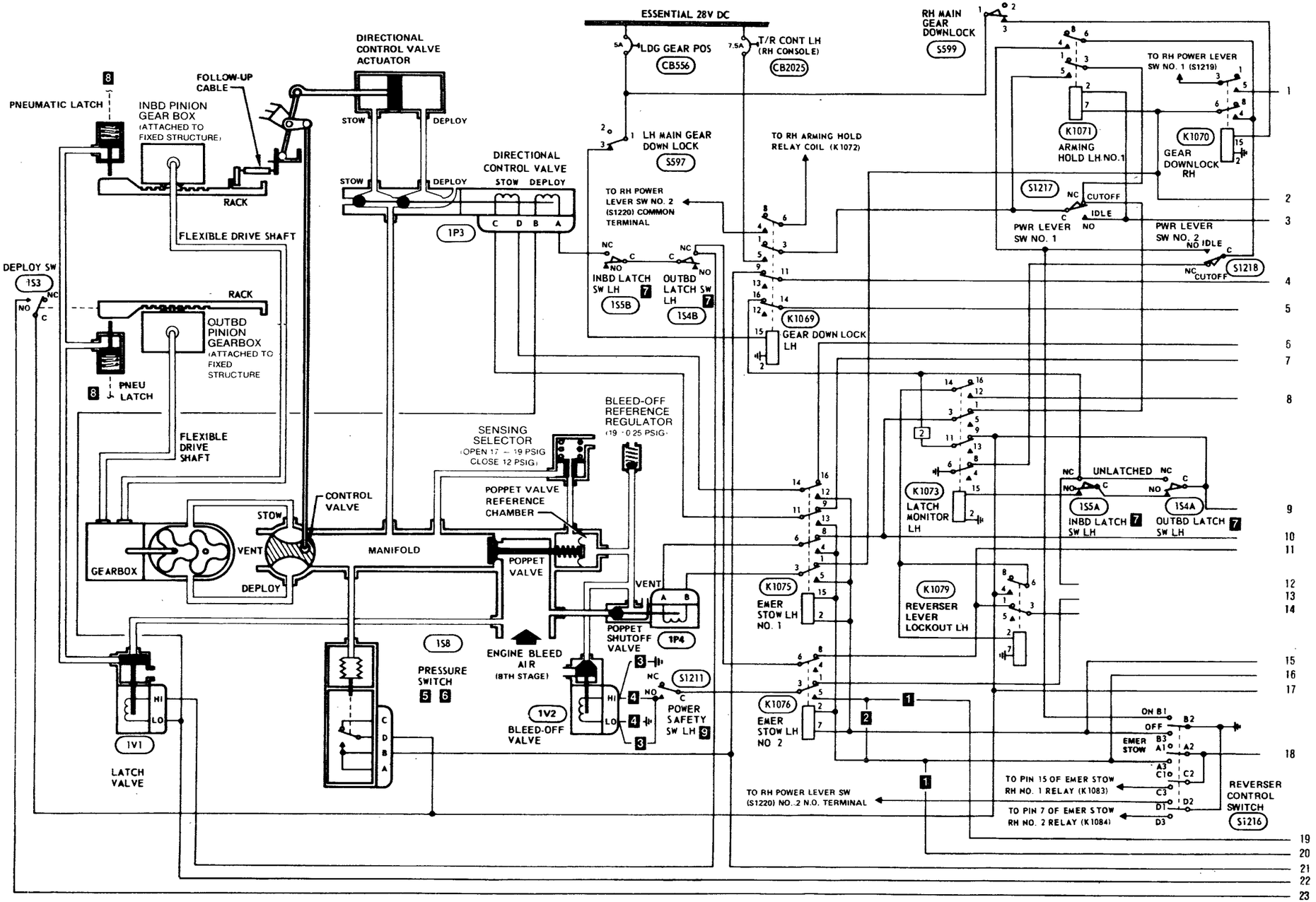
4A. The feedback system shuts off the air supply to the pneumatic motor as the rack approaches the end of travel in both deploy and stow. Feedback components are the follow-up cable, a slider on the inboard pinion gearbox, a slider mechanism in the pneumatic actuator, and a follow-up cable actuator (tang) on each end of the inboard rack. As the rack reaches the last one-half inch of travel, the tang on the end of the rack contacts the slider in the pinion gearbox, moving the slider which in turn moves the follow-up cable. As the follow-up cable moves, it moves the pneumatic motor control valve toward null, shutting off air to the motor and slowing it down. The closer the rack gets to the end of travel the more the motor is slowed down until the motor actually functions as a brake. At end of rack travel, the pneumatic control valve is not quite fully nulled, but there in insufficient air to the motor to cause it to drive further.



Thrust Reverser Translating Components
Figure 78-30-1

NOTES

1. *S/N 380-5, -10, -12 and subs and after S/B 75-17*
2. *S/N 380-1 thru -4, -6 thru -9, and -11, and before S/B 75-17*
3. *S/N 380-1 thru -3*
4. *S/N 380-4 and subs*
5. *Closes 2.5 (±1) psig open 0 psig on S/N 380-1 thru -57*
6. *Closes 4.75 ±1) psig open 2 psig on S/N 380-58 and subs*
7. *Actuated by pneumatic latches*
8. *Actuates respective latch switches*
9. *Switches actuated by power lever linkage at approximately 80% engine rpm.*



Thrust Reverser Electrical Schematic.
Figure 78-30-2 (Sheet 1 of 2)

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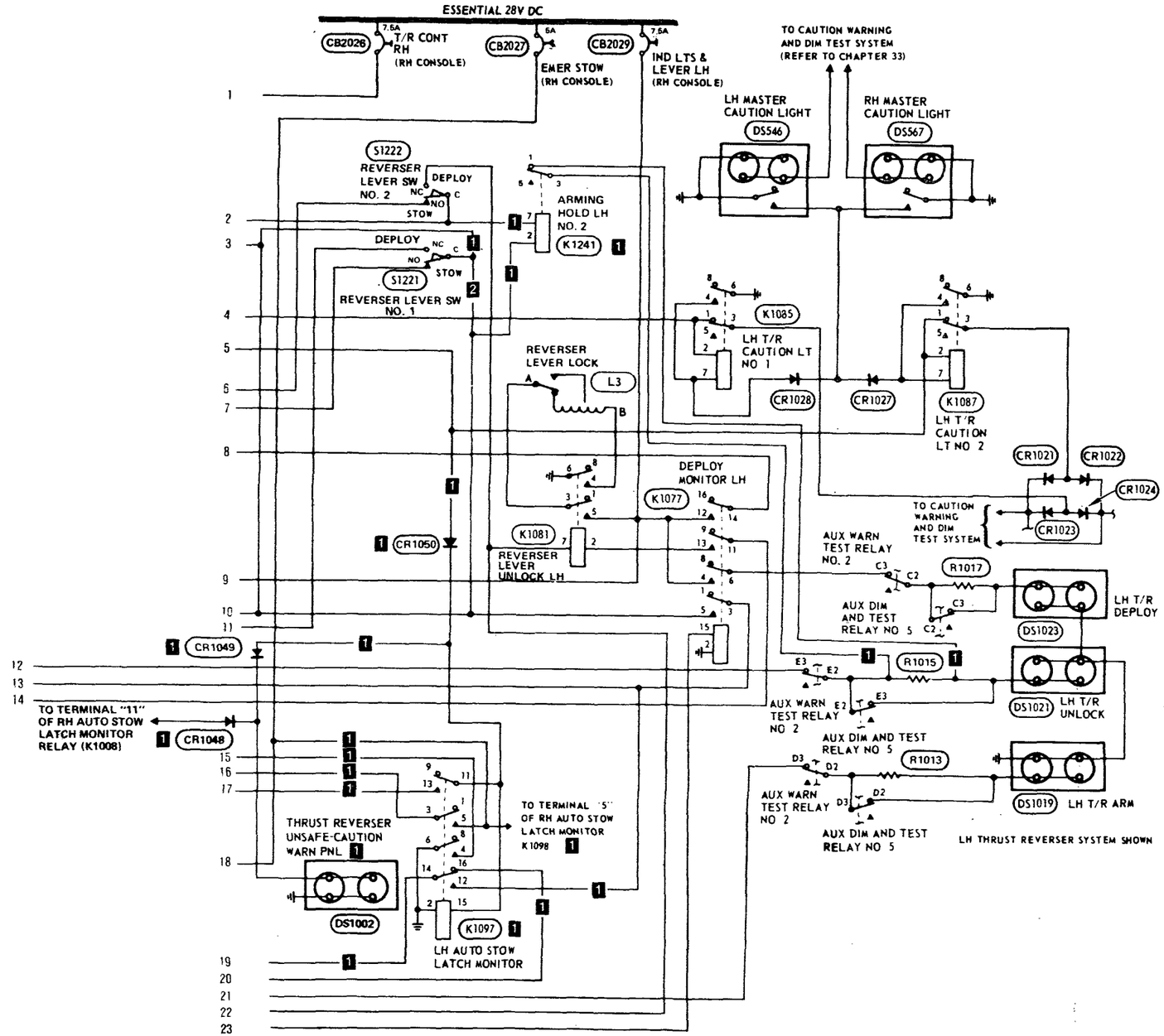

Thrust Reverser Electrical Schematic
Figure 78-30-2 (Sheet 2 of 2)

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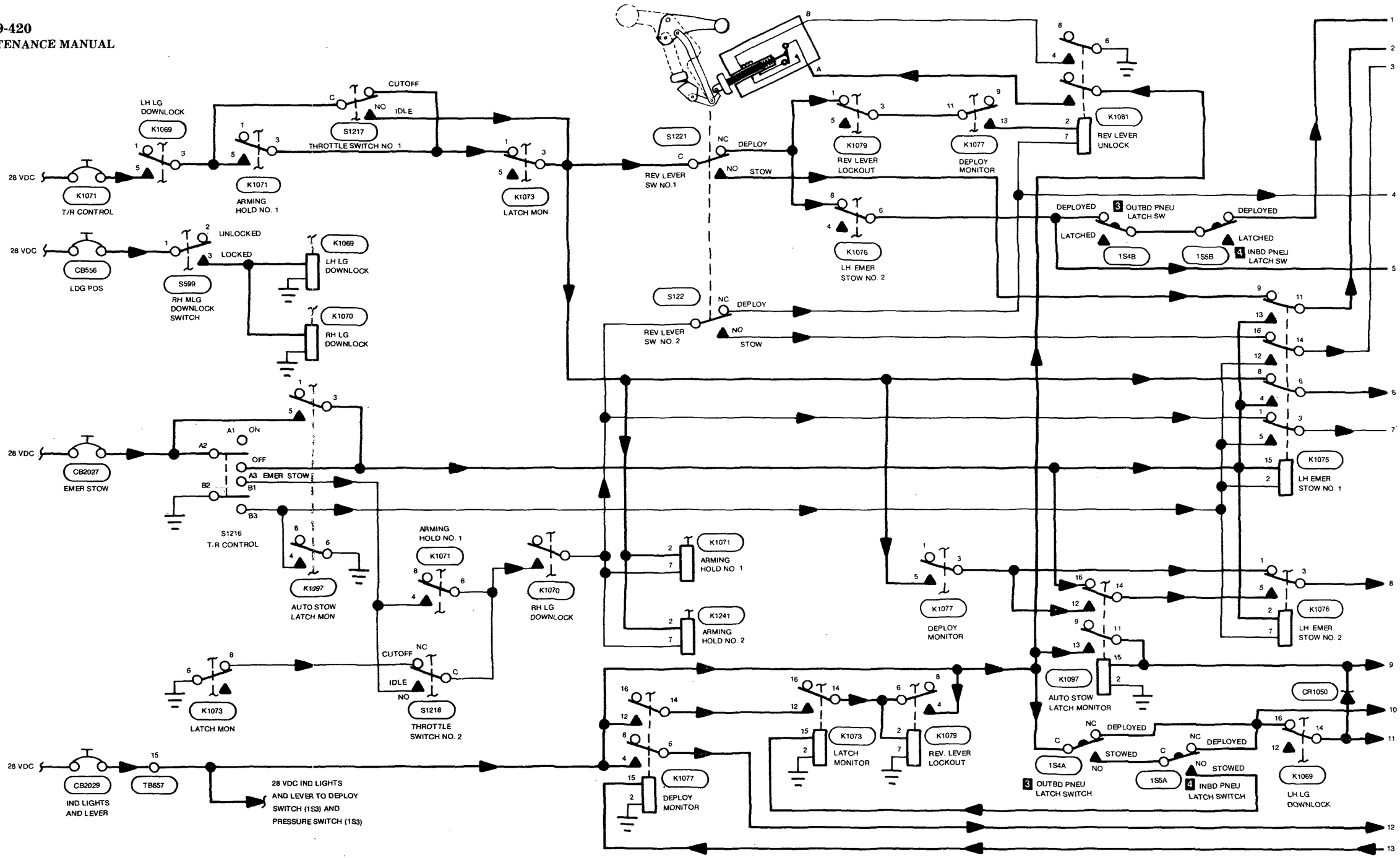

Thrust Reverser System - S/N 380-5, -10, -12 and Subs
and After S/B 75-17 and S/B 29
Figure 78-30-2A (Sheet 1 of 2)

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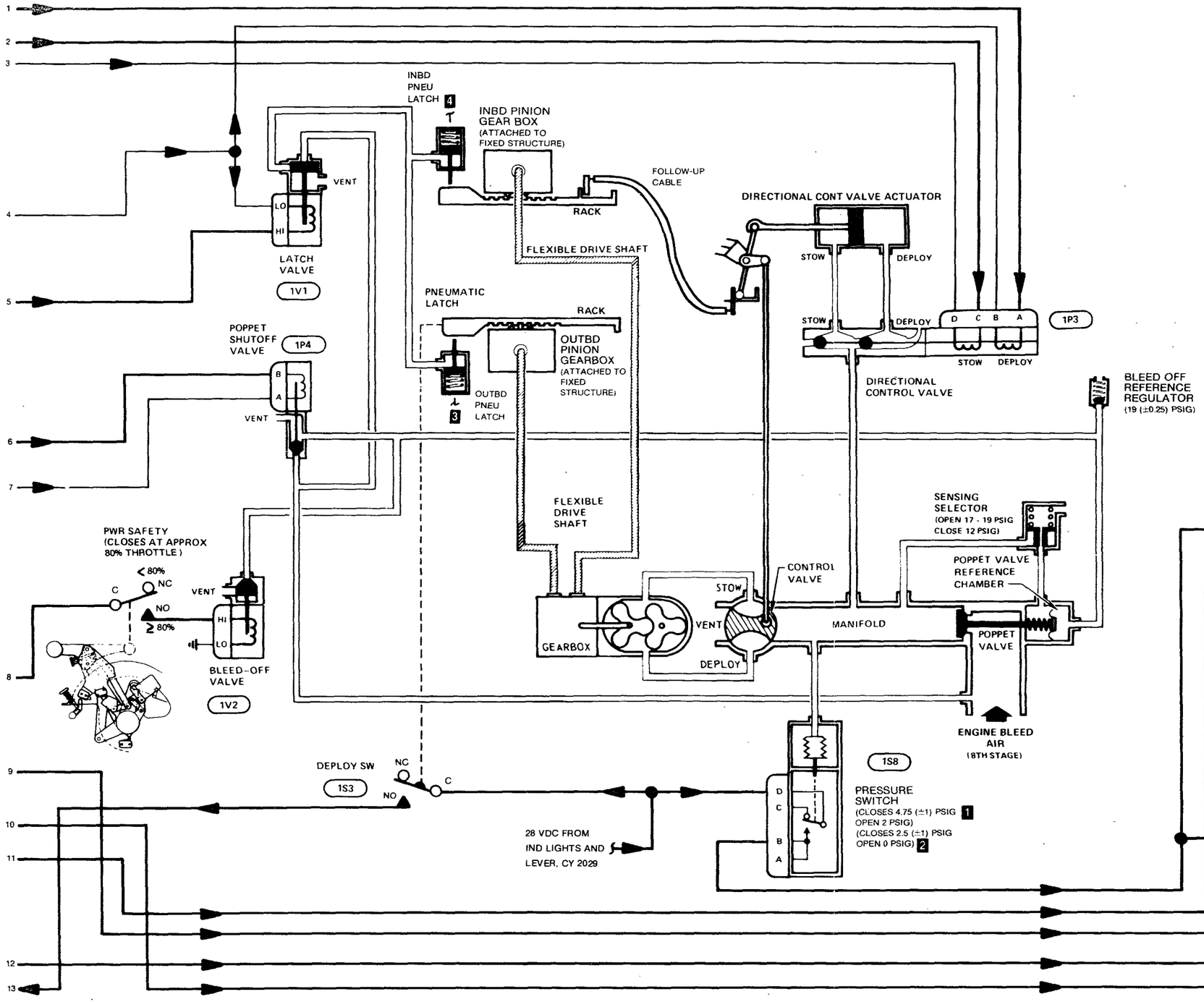

RELAYS		
NAME	RDI	LOCATION
Ldg. Gear Downlock - LH	K1069	Relay Panel Pedestal
Ldg. Gear Downlock - RH	K1070	
Arming Hold No. 1 - LH	K1071	
Latch Monitor - LH	K1073	
Emer. Stow No. 1 - LH	K1075	
Emer. Stow No. 2 - LH	K1076	
Deploy Monitor - LH	K1077	
Reverser Lever Lockout - LH	K1079	
Reverser Lever Unlock - LH	K1081	
Auto Stow Latch Monitor - LH	K1097	
Arming Hold No. 2 - LH	K1241	

NOTES

1. S/N 380-58 AND SUBS
2. S/N 380-1 THRU -57
3. OUTBD PNEU LATCH SWITCHES 1S4A AND 1S4B CHANGED TO OUTBD PNEU LATCH.
4. INBD PNEU LATCH SWITCHES 1S5A AND 1S5B CHANGED TO INBD PNEU LATCH

Thrust Reverser System - S/N 380-5, -10, -12 and Subs and After S/B 75-17 and S/B 29
Figure 78-30-2A (Sheet 2 of 2)

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5. Poppet Shutoff Valve

The poppet shutoff valve, is a solenoid actuated valve which controls the operation of the poppet valve. The poppet shutoff valve is energized open when the power lever is at idle and the thrust reverser control switch is on, and through alternate circuitry when an emergency or automatic stow condition exists.

With the thrust reverser control switch on, and the power lever at idle, essential bus power from the T/R CONT LH circuit breaker is routed through the left gear downlock relay, K1069 (contacts 3 and 5) to the No. 1 power lever switch S1217 common and idle contacts through the emergency stow relay K1075 (contacts 6 and 8) to terminal "A" of the poppet shutoff valve. A ground circuit for the poppet shutoff valve is provided from terminal "B" through the emergency stow relay (contacts 1 and 3), through the right gear downlock relay K1070 (contacts 4 and 6), through the No. 2 power lever switch S1218 (common and idle contacts), through the thrust reverser control switch S1216, (B1 and B2), to ground.

During an emergency stow, the power and ground circuits for the poppet shutoff valve are completed through the emergency stow relay K1075, (contacts 4 to 6, and 3 to 5) and through the thrust reverser control switch 1216.

On aircraft NA380-5,-10,-12 and later airplanes, the auto stow relay (K1097) will actuate the emergency stow system and provide power and ground circuits (contacts 3 to 5, and 4 to 6) substituting for those through the thrust reverser control switch provided during an emergency stow operation.

6. Pressure Switch (1S8)

The pressure switch is pneumatically operated by engine bleed air pressure tapped from the manifold. The contacts of the switch controls electrical power from the IND LTS & LEVER circuit breaker to the ARM indicator light. The contacts are closed when bleed air pressure in the manifold is above (*2.5±1 psig) (†4.75±1 psig). The switch deactuates (contacts B and D opens) when the manifold pressure drops to (*0.0 psig) (†2.0 psig).

7. Latch Valve

The latch valve is an electrically controlled valve. As the reverser lever is moved to the deploy position, electrical power is routed from the T/R CONT CIRCUIT breaker through the No. 1 power lever switch (S1217) through the reverser power lever switch (S1221), through the No. 2 emergency stow relay to the HI terminal of the latch valve. A ground circuit for the latch valve is completed through the No. 2 reverser lever switch, through the right hand gear downlock relay and through two parallel circuits, one through the No. 1 arming hold relay

*Airplanes NA-380-1 thru -57.
†Airplanes NA-380-58 and Later.

K1071, the other through the No. 2 power lever switch S1218 and through the reverser control switch to ground. With the latch valve energized, engine bleed air direct from the engine is routed through the valve to the inboard and outboard pneumatic latches.

During a normal stow operation with the reverser control switch ON, the power lever could be moved to cutoff prior to complete stowage. Under these circumstances the No. 1 arming hold relay would maintain a power and ground circuit for the poppet shutoff valve and the stow solenoid of the Directional Control Valve.

When latch switches 1S4A and 1S5A are deactuated, power is removed from the latch monitor relay K1073, and power is applied to the unlock light.

When latch switches 1S4B and 1S5B are deactuated power is applied to the deploy solenoid of the control valve.

8. Pneumatic Latches

The pneumatic latches are mounted on the forward end of both racks. Each latch consists of an actuator (piston type), two switches, and two latch roller arms. The latch pistons are spring loaded to the latched position. Each latch requires approximately 5 psi air pressure to overcome the spring loaded piston. Both latches are actuated to the unlatched position by 8th stage engine bleed air pressure that is routed through the latch valve. As pneumatic pressure is applied to the pneumatic latches and exceeds 5 $\pm$ 1 psig, the latch pistons extend, retracting the mechanical latch roller arms that were maintained in the locked position by the linkage being overcenter. As the latch roller arms are retracted, the latch switches 1S4A, 1S4B, 1S5A, and 1S5B are deactuated.

9. Bleed-off Valve

The bleed-off valve is a solenoid actuated valve controlled by the power safety switch (S1211) which is actuated by the reverser lever linkage (cam) at approximately 80% engine RPM. When the valve is energized open, it vents the poppet valve reference chamber allowing the poppet to close which shuts off engine bleed air to the manifold. In this condition, engine bleed air is not available to stow the reversers. As long as the valve is energized, the emergency stow and auto stow functions are inoperative. This prevents stowing of the reversers at high engine power settings (above 80% engine RPM).

10. Sensing Selector

The sensing selector maintains a manifold pressure from 12 to 19 psi. When the poppet shutoff valve is energized open, engine bleed air is admitted to the poppet valve reference chamber. This causes the poppet valve to open allowing engine bleed air to pressurize the manifold.

Bleed air from the manifold enters the sensing selector. If the pressure is above 17 to 19 psig the sensing selector opens, admitting bleed air to a chamber opposite the poppet valve reference chamber resulting in the poppet valve being modulated as required to maintain a manifold pressure of 12 to 19 psi. The sensing selector has a 0.03 pounds per minute continuous air bleed flow.

11. Bleed-off Reference Regulator

The bleed-off reference regulator maintains a constant pressure in the poppet valve reference chamber. As the poppet valve reference chamber pressure increases to 19 (±0.25 psig) the regulator will open. The poppet valve reference chamber is then maintained at this pressure providing the engine bleed air is above this pressure.

12. Deploy Switch

The deploy switch is mounted on the outboard pinion gearbox support and is actuated when the reverser is fully deployed. It is actuated by an actuating lever making contact with the deploy pin in the forward end of the outboard rack. When actuated it completes a power circuit from the IND LTS & LEVER LH circuit breaker to the deploy monitor relay K1077.

13. Directional Control Valve and Actuator

The directional control valve and actuator operate in conjunction with one another to select the stow or deploy position of the thrust reverser. The control valve is a latching type, and is positioned in the deploy position by electrical circuits (power and ground) completed through the reverser lever and power lever actuated switches. With the deploy solenoid energized, manifold bleed air is routed to the actuator. The actuator in turn positions the control valve to the deploy position routing engine bleed air to the drive motor. The drive motor is then energized, driving the thrust reversers to the deployed position. When a stow position is selected, the same reverser and power lever switches are activated to stow position which results in the electrical circuits (power and ground) to be completed to the stow solenoid in the control valve. The control valve is then positioned to stow, allowing manifold pressure to position the actuator to the stowed position. The actuator in turn positions the control valve routing engine bleed air to the drive motor. The drive motor is then energized, driving the thrust reversers to the stowed position. The directional control valve stow solenoid may be automatically energized by the auto stow system. (NA380-5, -10, -12 and later airplanes).

In the event the auto stow system is activated, electrical circuits (ground and power) are completed to the stow solenoid through the energized emergency stow relays (K1075 and K1076). This positions the directional control valve and actuator the same as during a normal stow selection.

14. Latch Switches

The latch switches are mounted on the inboard and outboard pneumatic latches. Two latch switches are mounted on the top of each pneumatic latch. (Four switches per reverser). The switches are actuated when the pneumatic latches are in the latched position. When actuated, the switches 1S4A and 1S5A in series provides a power circuit from the IND LTS & LEVER LH circuit breaker to the latch monitor relay K1073. When deactuated, they remove power from the latch monitor relay, and applies power to the unlock light. On NA380-5, -10, -12 and later airplanes, these switches also activate the auto stow system.

The switches 1S4B and 1S5B, when activated, opens the power circuit to the deploy solenoid of the directional control valve. When deactuated, they close the power circuit to the deploy solenoid of the directional control valve.

15. Thrust Reverser Control Switch

A guarded thrust reverser control switch is mounted on the top, left, corner of the power lever pedestal, aft of the power levers. It has three positions, ON, OFF, and EMER STOW.

In the ON position, and dependent on the position of the power and reverser levers, it provides the ground circuits for the latch valve, poppet shutoff valve, and the deploy or stow solenoids of the directional control valve.

In the EMER STOW position, power and ground circuits are provided to the No. 1 emergency stow relay K1075 and the No. 2 emergency stow relay K1076.

When the No. 1 emergency stow relay is energized, direct power and ground circuits are provided to the poppet shutoff valve, and the stow solenoid of the directional control valve.

When the No. 2 emergency stow relay is energized, the power circuit to the latch valve and the deploy solenoid of the directional control valve is opened. A power circuit is also completed to the 80% RPM power safety switch, bypassing the contacts of the deploy monitor relay and preventing stow at engine RPM above 80% with the reverser at any position.

16. Reverser Lever Unlock Solenoids

The reverser lever unlock solenoids are in the engine control quadrant. When the reverser blocker doors are fully deployed, electrical power and a ground circuit is provided through the reverser lever unlock relay, thus energizing the solenoid, allowing the reverser lever to be moved to the full reverse-thrust position. When the reverser levers are returned to the minimum thrust position, the unlock solenoids are de-energized and the levers are relocked.

17. Power Lever Switches (No. 1 and No. 2)

The power lever switches are mounted in the engine control quadrant. These switches are actuated when the power lever is at IDLE. When these switches are actuated, power is provided for the coil of the arming hold relays, thus energizing the relay. The "ARM" indicator light is illuminated by actuation of the reverser pressure switch. The deploy circuit is energized only after actuation of the main landing gear downlock switches and the power lever switches. The No. 1 switch provides the power circuit and the No. 2 switch provides the ground circuit.

18. Reverser Lever Switches (No. 1 and No. 2)

The reverser lever switches are mounted in the engine control quadrant and are actuated by the reverser levers. Electrical power is applied to the stow solenoid of the directional control valve through these switches when the reverser levers are in the stowed position. Lifting the reverser control levers de-actuates these switches and applies electrical power to the deploy solenoid of the directional control and latch valves for deployment of the reverser blocker doors. When the reverser levers are returned to the stowed position, electrical power is again applied to the stow solenoid of the directional control valve. The No. 1 switch provides the power circuit and the No 2 switch provides the ground circuit.

19. Operation of Thrust Reverser System

Operation of the thrust reverser system is the same for both the left and the right engine. Operation of the left engine thrust reverser is covered in the following paragraphs. (See figure 78-30-2 for the schematic diagram of the electrical circuit).

Electrical power for the system is obtained from the 28-volt dc essential bus. Air for the pneumatic system is extracted from each engine. To deploy the thrust reverser, the left and right main gear must be down, the reverser control switch at ON and the power lever on the affected

engine must be at idle and the gust lock disengaged. In this condition, circuits are completed which connect 28-volt dc power to the stow solenoid in the directional control valve and to the poppet shutoff valve solenoid to energize it to the open position. The open poppet shutoff valve allows engine bleed air to pressurize the manifold and actuating the system pressure switch which turns on the "LH T/R ARM" indicator light. When the reverser lever is raised to the deploy position, the circuit is broken to the directional control valve stow solenoid and a circuit is completed to energize the latch valve. The latch valve opens, allowing engine bleed air to actuate the latches to the unlatched position. The latches in turn de-actuate the latch valve switches which complete a circuit to energize the deploy solenoid of the directional control valve and a circuit to turn on the "LH T/R UNLOCK" indicator light. With the deploy solenoid in the directional control valve energized, engine bleed air pressure moves the control valve actuator which, through mechanical linkage, positions the control valve to the deploy position. This allows engine bleed air to drive the pneumatic drive motor in the deploy direction. The drive motor gearbox drives the reverser pinion gearboxes through flexible shafts. The gearboxes drive the respective translating beam and rack assembly on which the reverser translating structure reaches a point 5.59 inches from fully deployed, a stop is contacted by the beam assemblies. At this time, the rack/beam assembly hammer disengage and the rack continues rearward for another 5.59 inches. This final movement causes mechanical linkage to actuate the blocker doors to the closed position, deflecting the engine thrust outward through the cascade assembly which directs the thrust forward. The rearward moving inboard rack actuates the push-pull follow-up cable which moves the control valve actuator to the neutral position shutting off the pneumatic drive motor. When fully deployed, a pin on the reverser rack actuates the reverser deploy switch. The deploy switch completes a circuit which causes the "LH T/R DEPLOY" indicator light to come on (engine RPM should be within 64 to 68%) and unlocks the reverser lever allowing it to be moved aft to increase reverse thrust. The deploy switch also completes a circuit which, when the power lever is above the 80% RPM position, the bleed-off valve is energized. The bleed-off valve releases air pressure in the poppet valve reference chamber. This shuts off engine bleed air to the pneumatic drive unit. This is a safety circuit which prevents inadvertent stow of the thrust reverser at high power settings.

With power lever settings below 80% RPM, emergency stow capability is provided so that the reverser can be stowed at any time, bypassing normal stow circuitry by positioning the thrust reverser selector switch to EMER STOW. This completes circuits which energize the two emergency stow relays. One of the emergency stow relays completes circuits directly to the poppet shutoff valve and the stow solenoid of the directional control valve to initiate the reverser stow cycle. The other emergency stow relay breaks the circuit to the latch valve and closes part of the circuit to the reverser bleed-off valve. If the reverser lever is positioned at the 80% RPM setting or higher, the

circuit to the bleed-off valve is completed and the valve opens. This prevents operation of the pneumatic drive unit so that the reverser cannot be stowed at high power settings.

On NA380-5, -10, -12 and later airplanes, an automatic stow function is added to the thrust reverser system. The system consists of a thrust reverser UNSAFE caution light (amber) and an auto stow latch monitor relay. The system is energized through actuation of the inboard or outboard latch switches (1S5A or 1S4A) whenever the landing gear is released from its downlocks. With the landing gear up and locked, engine RPM below 80%, reversers stowed, and the thrust reverser control switch in the OFF position, should either the inboard or outboard latch switch (1S5A or 1S4A) open, electrical power is removed from the latch monitor relay (K1073) which is then deenergized. With relay K1073 deenergized electrical power is routed from the indicator lights and lever circuit breaker, through the gear down relay (K1069) to the coil of the auto stow latch monitor relay (K1097) and the thrust reverser unsafe caution light. Relay K1097 is then energized and the caution light is illuminated. With relay K1097 energized, a ground and power circuit is provided to the emergency stow relay (K1076 and K1075). With relays K1076 and K1075 energized, electrical power and ground circuits are provided to the poppet shutoff valve and the stow solenoid in the directional control valve. This action allows engine bleed air to be routed to the actuator, which in turn stows the reversers. The unsafe light will remain on until the indicator lights and lever circuit breaker is opened or when electrical power to the aircraft essential bus is interrupted. This is due to the electrical holding circuit provided by contacts "11 and 13" of the auto stow latch monitor relay (K1097).

In the normal stow cycle, the reverser lever is moved forward and down to the stowed position. This completes circuits through the reverser and power lever switches to the poppet shutoff valve and to the stow solenoid of the directional control valve. Engine bleed air is routed to the stow side of the pneumatic drive motor. As the racks start to move forward, a circuit is opened by deactuation of the deploy switch which causes the "LH T/R DEPLOY" indicator light to go out and the reverser lever lock to deenergize and block the reverser lever. The initial movement of the rack assemblies cause mechanical linkage to open the blocker doors in the exhaust nozzle.

As the rack moves forward 5.59 inches, the blocker doors are stowed and cam surfaces on the forward end of the racks lifts the hammer blocking arm, subsequently releasing the hammer. At the same time the hammer roller is forced (by forward movement of the beams) out of the catches (catches are attached to the forward side of the pinnion gearboxes) allowing the hammer (finger end) to engage the rack detents. This locks the rack to the beam. As the beam moves forward the hammer lock pin extends past the outboard side of the hammer roller. This prevents the hammer from disengaging from the rack during the remaining portion of the stow cycle. As the translating structure approaches

the full stowed position, the push-pull follow-up cable is moved, repositioning the directional control valve to neutral. As the beam/rack assys reach the full forward position, the ramps on the top forward end of the beams contact the pneumatic latch roller arms moving them forward and releasing the latch roller arms. The rollers of the pneumatic latch arms drop on the ramps where they are prevented from extending far enough to deactuate the latch switches. As the beams reach their most forward position, the latch roller arms pass the aft end of the ramps and extend into the beams and latch. This deactuates the pneumatic latch switches and energizes the latch monitor relay (K1079). With the latch monitor relay energized, the No. 2* arming hold relays (K1214 and K1071 are broken. On NA380-5, -10, -12 and later the dimming function for the UNLOCK light has been removed when the thrust reverser system is not armed. The dimming resistor is shorted-out by contacts (#3 and #5) of the No. 2 arming hold relay.

DEPLOY SEQUENCE

A. Controls:

(1) Thrust reverser circuit breakers in.

(2) T/R switch at ON.

(3) Power lever at IDLE

(4) Reverser levers stowed.

B. Relays Energized:

K1069 (LH gear down lock relay), K1070 (RH gear down lock relay), K1073 (latch monitor).

C. Circuit Events:

(1) 28-volt dc power from the T/R control circuit breaker is routed through normally open (N.O.) contacts in the LH gear down lock relay K1069, through the power lever switch No. 1 (closed with power lever at IDLE), through normally closed (N.C.) contacts in the emergency stow relay, K1075 to pin "A" of the poppet shutoff valve. Pin "B" of the poppet shutoff valve is grounded through a set of N.C. contacts in relay K1075, through a set of N.O. contacts in the RH gear downlock relay, K1070, through a parallel circuit, one a set of N.O. contacts in the arming hold relay, K1071, and the other through the power lever switch No. 2, and then through the reverser control switch in the ON position. This energizes the valve to the open position allowing engine bleed air to pressurize the manifold. This air pressure actuates the pressure switch which completes a circuit from the

*NA380-5, -10, -12 and later

"IND LTS & LEVER" circuit breaker to the "T/R ARM" indicator light.

D. Controls: Reverser lever at deploy position.

(1) 28-volt dc power from the T/R control circuit breaker is applied to the "HI" pin on the latch valve through N.C. contacts in emergency stow relay, K1076, through reverser lever switch No. 1 (closed with reverser levers at deploy), through power lever switch No. 1 (closed with power lever at IDLE), and through N.O. contacts in LH gear downlock relay K1069. The valve coil is grounded at pin "LO" through reverser lever switch No. 2 (closed with reverser lever deployed), through N.O. contacts in RH gear downlock relay, K1070, through power lever switch No. 2 (closed with power lever at IDLE), and through the reverser control switch at the ON position. This energizes the latch valve to the open position, allowing engine bleed air to pressurize the latch actuators which move to the unlatched position. Movement of the latches to the unlatched position deactuates the latch switches, 1S4A, 1S4B, 1S5A, and 1S5B. When either 1S4A or 1S5A is deactuated the circuit to the coil of the latch monitor relay K1073 is broken, causing the relay to deenergize and a circuit is completed through either switch to the "T/R UNLOCK" indicator light. A backup circuit to the "T/R UNLOCK" light is completed through N.C. contacts of the latch monitor relay (K1073) to ensure of an indication whenever the thrust reverser is unlatched.

(2) 28-volt dc power from the T/R control circuit breaker is routed to the directional control valve deploy solenoid, through the N.O. contacts of the left hand gear downlock relay K1069, through the power lever switch No. 1 in the IDLE position, through the reverser lever switch No. 1 in the deploy position, through N.C. contacts of the No. 2 emergency stow relay K1076 and through the outboard and inboard latch switches 1S4B and 1S5B.

The actuator positions the control valve in the deploy position allowing engine bleed air to drive the pneumatic motor in the deploy direction. The pneumatic motor drives a gearbox which transmits torque through flexible drive shafts to pinnion gearboxes on the thrust reverser structure. The pinnion gearboxes drive their respective racks which are fixed to the thrust reverser translating structure. The guide, translating feedback, inside the pneumatic actuator assembly, is connected to a push-pull follow-up cable. The guide is connected to the directional control valve actuator. The stop on the forward end of the inboard rack when fully extended contacts the follow-up cable slider assembly and the follow-up cable repositions the guide and directional control valve actuator

within the pneumatic actuator assembly. As the control valve actuator is repositioned by the follow-up cable, the control valve is rotated to the off position slowing down the air motor and when fully deployed shutting it off. The deploy pin on the outboard rack, in the deployed position, actuates the deploy switch, 1S3, completing a circuit from the "IND LTS & LEVER" circuit breaker to energize the deploy monitor relay K1077. When the deploy monitor relay energizes, three circuits in the deploy sequence are completed. In one, 28-volt dc power is connected from the "IND LTS & LEVER" circuit breaker to the "T/R DEPLOY" indicator light. The second circuit completed is a circuit from the "T/R CONT" circuit breaker through N.O. contacts in the LH GEAR DOWNLOCK relay through the power lever switch No. 1 (power lever at IDLE), through the reverser lever switch No. 1 (lever at deploy), through N.C. contacts in the power lever lockout relay, K1079, through N.O. contacts of the deploy monitor relay (K1077), to the coil of the reverser lever unlock relay, K1081. The coil of the reverser lever unlock relay is grounded through the reverser lever switch No. 2 (lever at deploy), through N.O. contacts in the RH gear downlock relay, through a parallel circuit, one the power lever switch No. 2 (power lever at IDLE), the other through the No. contacts of the No. 1 arming hold relay K1071 and then through the T/R control switch (at ON). The reverser lever unlock relay energizes and completes a power and ground circuit to energize the reverser lever lock so that engine thrust may be increased for braking. The third circuit completed when the deploy monitor relay is energized is from the T/R CONT circuit breaker through N.O. contacts in the LH gear downlock relay through power lever swith No. 1 (power lever at IDLE), through N.O. contacts in relay deploy monitor relay and through the power safety switch (closed with engine RPM above 80%) to the bleed-off valve. The bleed-off valve exhausts engine bleed-air pressure from the poppet valve reference chamber which allows the poppet valve to close, shutting off engine bleed air to the manifold and directional control valve.

With air pressure shutoff, the pressure switch (1S8), opens and the "T/R ARM" indicator light goes off.

STOW SEQUENCE

A. Controls:

(1) T/R switch at ON.

(2) Power lever at IDLE.

(3) Reverser lever at deploy position.

B. Relays Energized:

LH landing gear downlock (K1069), RH landing gear downlock (K1070), Deploy monitor (K1077).

C. Circuit Events:

(1) When the reverser lever is placed in the stowed position, the power lever unlock relay (K1081) deenergizes, breaking power and ground circuits to the reverser lever lock. At the same time, the power and ground circuits for the latch valve solenoid and the directional control valve deploy solenoid are broken. The latch valve closes off engine bleed air to the pneumatic latches which in turn allows the latches to move to the latched position after the thrust reverser is fully stowed. Latch switches (1S4B and 1S5B) open the power circuit to the deploy solenoid of the directional control valve so that the reversers cannot be pressurized on the deploy side without latches being released. With the reversers stowed (latched), 28 vdc to the UNLOCK light is interrupted. (The UNLOCK light is on whenever either inbd or outbd latch is not latched.) The ground circuit to the poppet shutoff valve is completed through the No. 1 emergency stow relay (K1075), LH gear down lock (K1070), and through the parallel circuits - the power lever switch No. 2 and through the N.O. contacts of the No. 1 arming hold relay (K1071) and the reverser control switch (switch at ON). At the same time a power circuit is completed through the LH No. 1 emergency stow relay (K1075), through parallel circuits - the power lever switch No. 1 and through the latch monitor relay (K1073), LH No. 1 arming hold relay (K1071) and then through the LH gear downlock relay (K1069) to the T/R control circuit breaker. The poppet shutoff valve is then energized open allowing engine bleed air to enter the poppet valve reference chamber. The poppet valve is then opened, allowing bleed air to pressurize the manifold. At this time, the pressure switch is actuated and the T/R ARM indicator light comes on.

The reverser lever switches also complete the power and ground circuits for the stow solenoid of the directional control valve causing the valve to move to the stow position. The stow sequence within the pneumatic drive mechanism is the reverse of the deploy sequence. As the thrust reverser racks move toward the stow position, the deploy switch is deactuated, deenergizing the deploy monitor relay (K1077), breaking the power circuit to the reverser lever unlock relay (K1081) and the circuit to the T/R DEPLOY indicator light causing the light to go off. When the reverser racks reach the full stowed position the latches are latched and the follow-up push-pull cable repositions the directional control valve actuator to neutral. This shuts off air supply to the pneumatic drive motor.

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The power safety switch is actuated closed whenever the reverser levers are advanced to 80% RPM or above. The switch closes the circuit to the bleed-off valve which in turn is actuated and allows the poppet valve to close by venting the poppet valve reference chamber. This prevents the thrust reversers from being stowed at high (80% RPM or higher) reverser power settings.

EMERGENCY STOW SEQUENCE

A. Controls:

(1) Circuit breakers in.

(2) T/R control switch at OFF.

(3) Power lever below 80% engine RPM position.

(4) Reverser levers in deploy position.

B. Relays Energized:

LH landing gear downlock K1069, RH landing gear downlock K1070, Deploy monitor K1077, Power lever unlock K1081.

C. Circuit Events:

With the T/R control switch positioned to EMER STOW, power and ground circuits are completed to emergency stow relays No. 1 and No. 2 (K1075 and K1076). Relay K1075 completes ground and power circuits to the stow coil of the directional control valve moving the valve to the stow position and to the coil of the poppet shutoff valve starting the pneumatic stow cycle, the indicator light circuits function the same as in a normal stow cycle.

THRUST REVERSER SYSTEM

FAULT ISOLATION

Table 101. Thrust Reverser System Fault Isolation

SYMPTOM/ PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
T/R WILL NOT DEPLOY		
1. Deploy solenoid in directional control valve defective.	Check for 28 vdc between pins A and B of directional control valve plug.	Replace pneumatic actuator assembly if correct voltage is present.
2. Main landing gear downlock switches defective.	Check main landing gear downlock switches for continuity if arm indicator light is out.	Replace switches as necessary.
3. Pressure switch defective.	Check for 28 vdc at terminals Band D of electrical plug.	Replace switch if correct voltage is present.
4. Defective poppet shut-off valve.	Check pins A and B of valve plug for continuity.	Replace pneumatic actuator assembly if no continuity exists.
5. Power lever control switches defective and/or misadjusted.	Check switches for proper operation and/or adjustment.	Replace or adjust switches as necessary.
6. Defective pneumatic actuator assembly.	Check actuator for operation and/or adjustment.	Replace pneumatic actuator assembly.
7. Defective pneumatic latch.	Check latch for operation and/or adjustment.	Replace pneumatic latch.
8. Defective flex drive assembly.	Check flex drive shaft inner cable.	Replace flex drive shaft assembly.
9. Open bleed air supply line to pneumatic actuator assembly.	Check bleed air supply line for cracked or loose sections.	Tighten connections or replace damaged sections.
10. Bleed-off valve stuck open or power safety switch failure.	Raise reverser lever to interlock stop (power lever in idle and T/R switch on) if arm light does not come on. Remove pneumatic line from bleed off valve if unlock light comes on but reverser fails to deploy. Plug line and attempt deployment. Defective bleed-off valve or power safety switch if deployment occurs and arm light on.	Check power safety switch circuit, if closed, adjust or replace switch, if switch circuit is open, replace bleed-off valve.
BLOCKER DOORS FAIL TO DEPLOY		
1. Defective sequencing latch assembly on fwd end of translating beam.	Check operation of sequencing latch assembly.	Replace defective sequencing latch assembly.
2. Defective or misadjusted linkage on translating structure.	Check linkage for proper adjustment.	Replace or adjust linkage as required.
3. Loose or defective catch on pinion gearbox.	Check catch for proper operation and/or attachment.	Tighten catch and recheck pin/catch relationship.

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FAULT ISOLATION (Cont'd)

Table 101. Thrust Reverser System Fault Isolation (Cont'd)

SYMPTOM/ PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
PNEUMATIC ACTUATOR ASSEMBLY CONTINUES WITH DEPLOY MODE AFTER T/R IS FULLY DEPLOYED		
1. Push-pull followup cable improperly adjusted.	Check adjustment of push-pull followup cable.	Adjust push-pull followup cable as required.
2. Defective push-pull followup cable.	Check for damaged or broken push-pull followup cable.	Replace push-pull followup cable.
3. Defective pneumatic actuator assembly.	Check operation of pneumatic actuator assembly.	Replace pneumatic actuator assembly.
UNLOCK LIGHT FAILS TO ILLUMINATE		
1. Defective lamp.	Check lamp.	Replace lamp.
2. Misadjusted or defective pneumatic latch switch(es).	Check adjustment and operation of switch(es).	Adjust or replace switch(es) as required.
3. Defective wiring.	Perform continuity check of UNLOCK light wiring.	Replace defective wiring.
DEPLOY INDICATOR LIGHT FAILS TO ILLUMINATE WHEN T/R IS DEPLOYED		
1. Defective lamp.	Check lamp.	Replace lamp.
2. Misadjusted or defective deploy switch.	Check adjustment and operation of deploy switch.	Adjust or replace deploy switch as required.
3. Defective wiring.	Perform continuity check of deploy light wiring.	Replace defective wiring.
REVERSER LEVER CANNOT BE OPERATED FROM REVERSE IDLE TO REVERSE THRUST		
1. Defective lockout solenoid.	Check operation of lockout solenoid.	Replace lockout solenoid.
2. Defective wiring.	Perform continuity check of lockout solenoid wiring.	Replace defective wiring.
3. Defective deploy switch.	Check operation and adjustment of deploy switch.	Adjust or replace deploy switch as required.
T/R WILL NOT STOW		
1. Stow solenoid in directional control valve solenoid defective.	Check for 28 vdc between pins C and D of directional control valve solenoid plug.	Replace pneumatic actuator assembly if correct voltage is present.
2. Directional control valve spool stuck in deploy position.		Replace pneumatic actuator assembly.
3. Pneumatic latch switch(es) defective.	Check switch for continuity.	Replace or adjust switch as necessary.

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FAULT ISOLATION (Cont'd)

Table 101. Thrust Reverser System Fault Isolation (Cont'd)

SYMPTOM/ PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
T/R WILL NOT STOW (Cont'd)		
4. Power lever control switches defective and/or misadjusted.	Check switches for proper operation and/or adjustment.	Replace or adjust switches as necessary.
5. Main landing gear downlock switches defective.	Check landing gear downlock switches for proper operation and/or adjustment.	Replace and/or adjust switches as necessary.
6. Defective switch(es) on power lever.	Check switch(es) for proper operation and/or adjustment.	Replace defective switch(es) assembly.
7. Defective pneumatic actuator assembly.	Check pneumatic actuator operation and/or adjustment.	Replace pneumatic actuator assembly.
8. Defective flex drive shaft assembly.	Check flex drive shaft assembly inner cable.	Replace flex drive shaft assembly.
9. Open bleed air supply line to pneumatic actuator assembly or bleed valve stuck open.	Check bleed air supply line for cracked, loose or missing sections. Bleed valve should not vent during stow mode.	Tighten connections, replace missing or damaged sections, or replaced bleed valve.
10. Defective push-pull followup cable.	Check for damaged or broken push-pull followup cable.	Replace push-pull followup cable.
11. Push-pull followup cable improperly adjusted.	Check adjustment of push-pull followup cable.	Adjust push-pull followup cable as required.
AFTER BLOCKER DOORS HAVE OPENED T/R FAILS TO MOVE TO STOWED POSITION		
1. Defective sequencing latch or catch assemblies.	Check operation and/or adjustment of latch sequencing and catch assemblies.	Adjust or replace sequencing latch or catch assembly as required.
T/R FAILS TO LOCK IN STOWED POSITION		
1. Defective or misadjusted pneumatic latch assembly.	Check operation and/or adjustment of pneumatic latch assembly.	Adjust or replace pneumatic latch assembly.
2. Push-pull followup cable improperly adjusted.	Check adjustment of push-pull followup cable.	Adjust push-pull followup cable as required.
PNEUMATIC ACTUATOR CONTINUES WITH STOWING MODE OPERATION AFTER T/R IS FULLY STOWED.		
1. Pneumatic latch switch(es) not properly adjusted.	Check adjustment of switch(es).	Readjust switch(es) as required.
2. Push-pull followup cable improperly adjusted.	Check adjustment of push-pull followup cable.	Adjust push-pull followup cable as required.
3. Defective push-pull followup cable.	Check for damaged or broken push-pull followup cable.	Replace pneumatic actuator assembly.
4. Defective pneumatic actuator assembly.	Check operation of pneumatic actuator assembly.	Replace defective pneumatic actuator assembly.

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FAULT ISOLATION (Cont'd)

Table 101. Thrust Reverser System Fault Isolation (Cont'd)

SYMPTOM/ PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
UNLOCK INDICATOR LIGHT DOES NOT GO OUT WITH T/R FULLY STOWED		
1. Pneumatic latch switch(es) defective.	Check switch(es) for continuity (switch open).	Replace or adjust switch(es) as necessary.
2. Latch monitor relay defective.	Check for continuity at pins 15 and 2 of relay K1073.	Replace relay if necessary.

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THRUST REVERSER

MAINTENANCE PRACTICES

NOTE

This section is presented in a manner as to allow the mechanic to remove and replace an item and to completely rig the system in sequence. The replacement of thrust reverser components are presented first with the rigging procedures appearing last. Portions (replacement information) of the rigging procedures are redundant to the procedures as depicted in Chapter 71; therefore, should a malfunction of the reverser occur, the removal of the aft fairing scoop and thrust reverser assembly as depicted in this section shall be used in lieu of those procedures appearing in Chapter 71.

Anytime the thrust reverser requires major maintenance and/or replacement of a major component part, it is recommended that the complete rigging procedures in this section be performed in sequence as depicted. This method will ensure that the thrust reverser components are serviceable and are properly rigged.

When performing the rigging procedures, it is imperative that all rigging dimensions are obtained prior to proceeding to the next step. This action eliminates the possibility of bending of push-pull rods, lever arms, and etc.

1. Servicing Thrust Reverser System (Lubrication and Cleaning)

   A. Lubricate thrust reversers every *150 hours or 150 cycles*, whichever occurs first. To lubricate thrust reversers, proceed as follows:

      (1) Stow thrust reversers and remove the two upper access doors on nacelle aft fairing.

CAUTION

On RH thrust reverser, pneumatic latches are oriented such that any cleaning solvent spilled on or applied to upper end of the pneumatic latches is trapped. Vent holes in the upper surface allow the cleaning solvent to enter the pneumatic latches, possibly damaging the pneumatic latches and infiltrating into other parts of the pneumatic system, damaging other components as well. Positively avoid getting cleaning solvent into the cupped area on top of the RH thrust reverser pneumatic latches.

      (2) Using a soft bristle brush, clean latch pivot point and roller with methyl ethly ketone (MEK) or a suitable electrical contact cleaner. (See figure 78-30-3.) (Procure through any local electronic supplier.)

CAUTION

Do not use RAM GS-3 on pneumatic latch stem, seals, or near air inlet port. RAM GS-3 is incompatible with lubricant applied to stem, seal, and internal parts during manufacture. When the two lubricants are mixed, a sticky gummy residue is formed that binds pneumatic latch piston and stem and damages seals.

The proper remedy for a binding pneumatic latch is to replace the faulty pneumatic latch with a servicable unit. As a temporary measure until a serviceable unit is obtained, a sticking pneumatic latch can be cleaned then lubricated by spraying Permatex No. 116DA lubricant (Woodhill Chemical Co., 18731 Greenwood Parkway, Cleveland, OH 41128) into the air inlet port and around the stem of the piston.

      (3) Allow latch and roller to dry, then apply an adequate amount of dry film lubricant (Ram GS-3 dry lubricant, Ram Chemical, 210 E. Alondra Blvd., Gardena, California, or equivalent) to latch pivot point and roller.

      (4) Deploy thrust reversers.

      (5) Lubricate rack and beam assemblies of the translating structure with dry film lubricant (Ram GS-3 or equivalent).

      (6) Lubricate blocker door operating rods and tension rods end bearings (16 end bearings) with dry film lubricant (Ram GS-3 dry lubricant or equivalent). (See figure 78-30-22.)

      (7) Stow thrust reversers and install access doors.

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MAINTENANCE PRACTICES (Continued)

2. Cleaning Thrust Reverser Assembly

   A. To clean thrust reverser, proceed as follows:

WARNING

Cleaning solvents are both toxic and flammable. Avoid unnecessary or prolonged contact with the skin and keep away from all sources of ignition. Use only in well ventilated area. Failure to observe these essential precautions can result in personal injury and/or serious accident.

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(1) Cleaning materials used on thrust reverser assembly are as follows:

Solvent	Federal Specification P-D-680
Solvent	Federal Specification P-S-661
Methyl Ethyl Ketone	Federal Specification TT-M-261D
Ethyl Alcohol	Federal Specification O-E-00760C

(2) Clean thrust reverser assembly, as follows:

NOTE: Exercise care not to introduce cleaning agents into the pneumatic actuator package. The directional control valve exhaust port must be covered during cleaning. (See figure 78-30-3.)

(a) Clean thrust reverser assembly with P-D-680 solvent.

(b) Dry with filtered compress air at 20 psig maximum.

(c) remove cover from exhaust port.

(3) Cleaning metal parts after removal (except pneumatic actuator package, flex drive cables and the pinion gearboxes) as follows:

(a) Clean metal parts with P-S-661 solvent.

(b) Dry clean parts with lint-free cloth.

(4) Special cleaning (exterior surfaces of the pneumatic actuator flex drive cables and pinion gearbox) as follows:

(a) Clean parts with a lint-free cloth moistened with either P-S-661 solvent or TT-M-261D Methyl Ethyl Ketone.

(5) Cleaning electrical terminals, unpainted grounding surfaces and pins, as follows:

(a) Clean with coarse paper toweling and O-E-00760C Ethyl Alcohol.

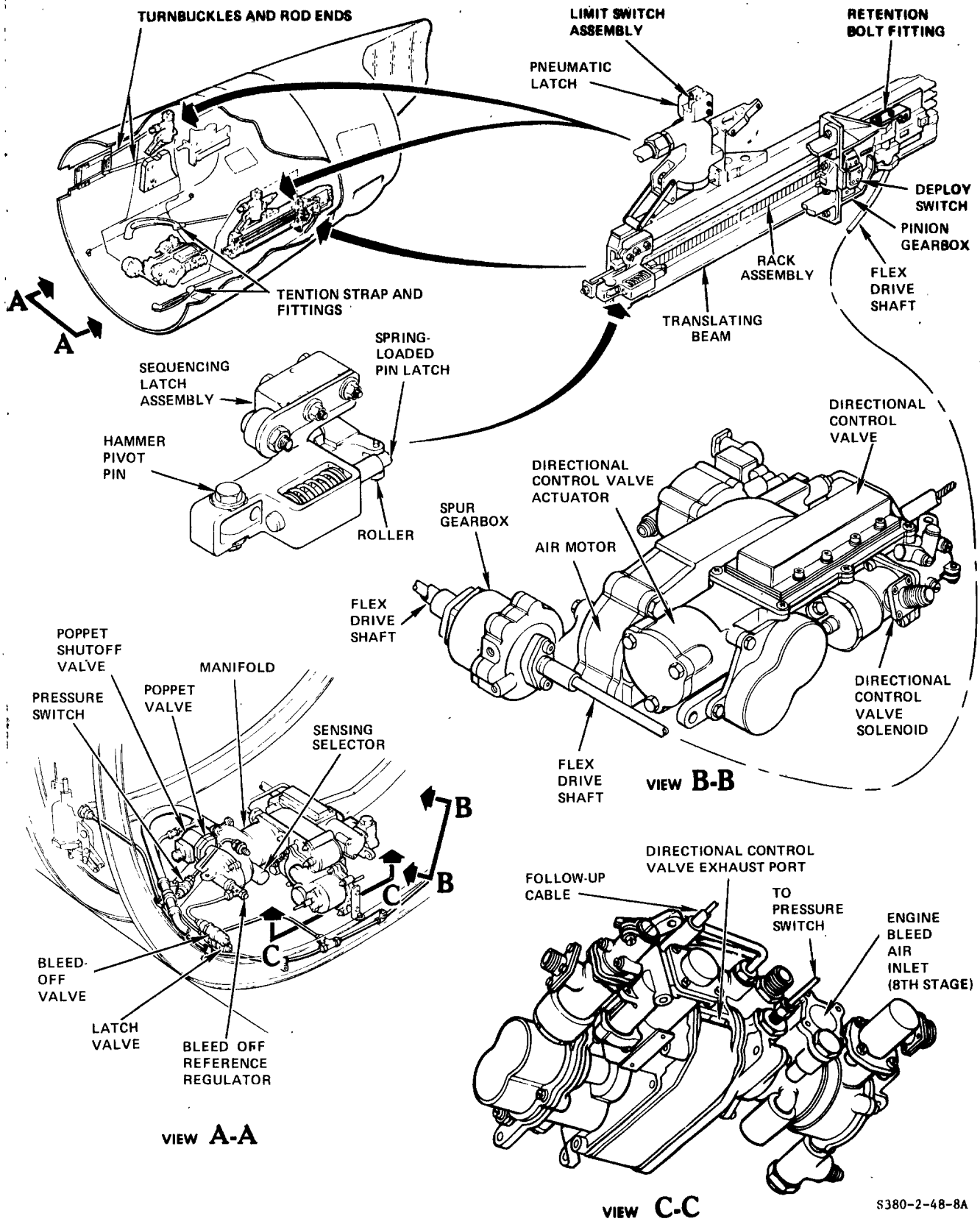
3. Cleaning Bleed-Off Valve

A. Cleaning bleed-off valve, proceed as follows:

NOTE: Bleed-off valves that have failed for any reason, may be restored to service by performing the following cleaning procedure.

(1) Remove nacelle aft fairing.

(2) Remove bleed-off valve from mounting with the valve connected to the electrical harness. Do not cut the electrical wiring connections.



Pneumatic Actuator Assembly and
Beam and Rack Major Components Location
Figure 78-30-3

MAINTENANCE PRACTICES (Continued)

(3) Hold the valve body with rubber cushioned pliers to avoid scoring the valve body during assembly.

(4) Disassemble the bleed-off valve to extent shown in Figure 78-30-04.

(5) Clean the valve body using clean cloth wetted with isopropyl alcohol. Allow the valve body to air dry before reassembly.

(6) Clean all parts separated from the bleed-off valve body with isopropyl alcohol and allow to air dry.

(7) Reassemble the bleed-off valve. Use cushioned pilers to avoid damaging valve body.

NOTE

If bleed-off valve cannot be repaired by cleaning, and has to be removed and returned to manufacturer for repair, the electrical leads are to be left as long as possible. This will ensure re-installation of valve.

(8) Reinstall bleed-off valve on fairing structure.

(9) Install nacelle aft fairing.

(10) Perform operation check of thrust reverser system.

4. Deleted. This information moved to inspection section of this chapter, p. 613.

5. Torque Translating Structure

Refer to chapter 5 for required frequency. Torque reverser translating structures as follows:

1. Remove nacelle aft fairing assembly.
2. Support nacelle aft fairing assembly in suitable stand.
3. Support structure as required, and torque nuts to 35-40 inch-pounds. (See Figure 78-30-5.)
4. Install nacelle aft fairing assembly.
5. Perform an operational checkout of thrust reverser system.

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MAINTENANCE PRACTICES (Continued)

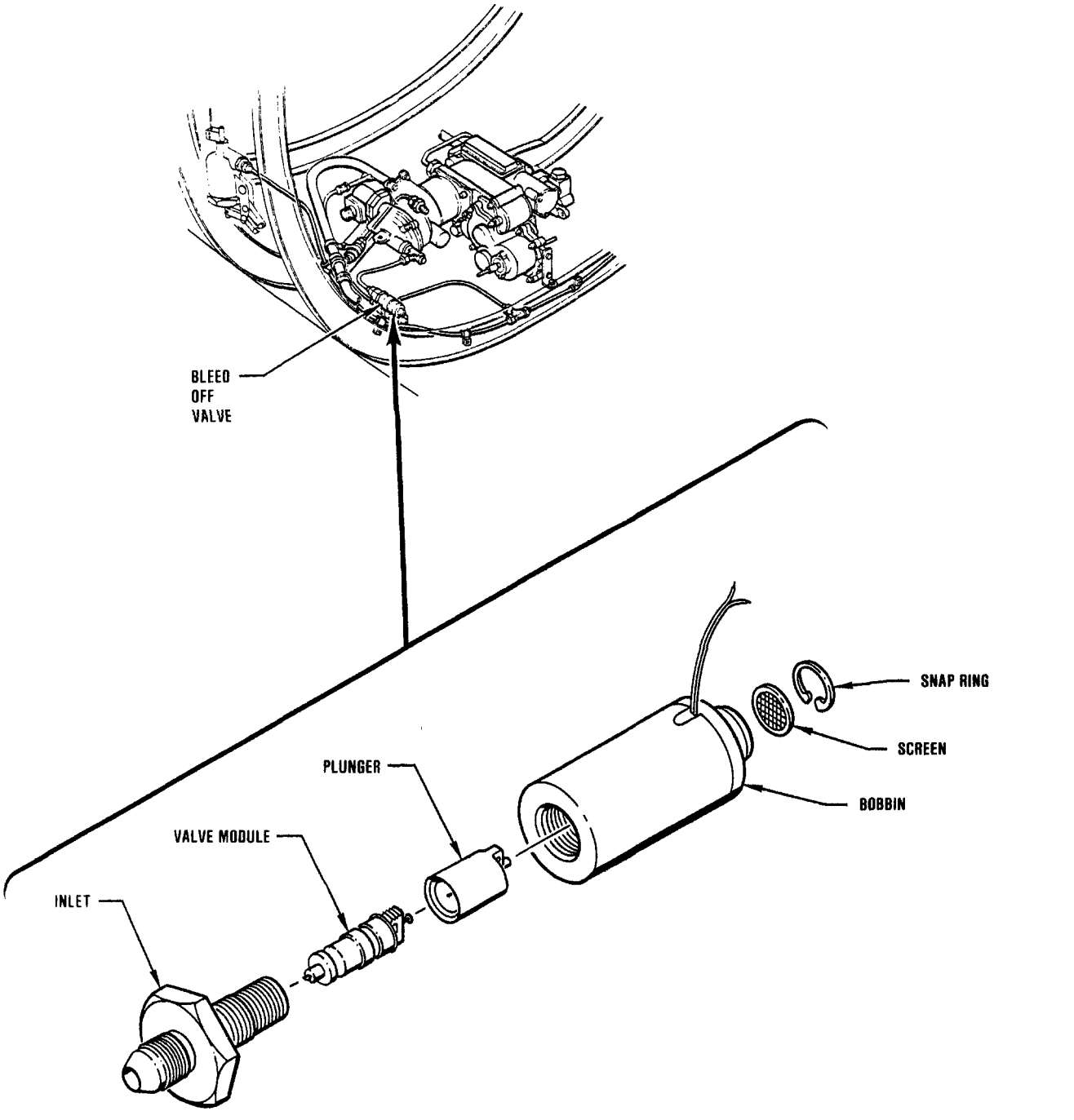

Figure 78-30-4 — Bleed-off Valve Disassembly for Cleaning

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MAINTENANCE PRACTICES (Continued)

5. Deleted

6. Removing Pneumatic Actuator Assembly and Spur Gearbox

   A. Removing pneumatic actuator assembly (P/N 126372-4-( ), AiResearch) and spur gearbox, proceed as follows:

      (1) Remove nacelle aft fairing.

      (2) Remove two flex drive shafts from pneumatic actuator spur gearbox. (See figure 78-30-3.)

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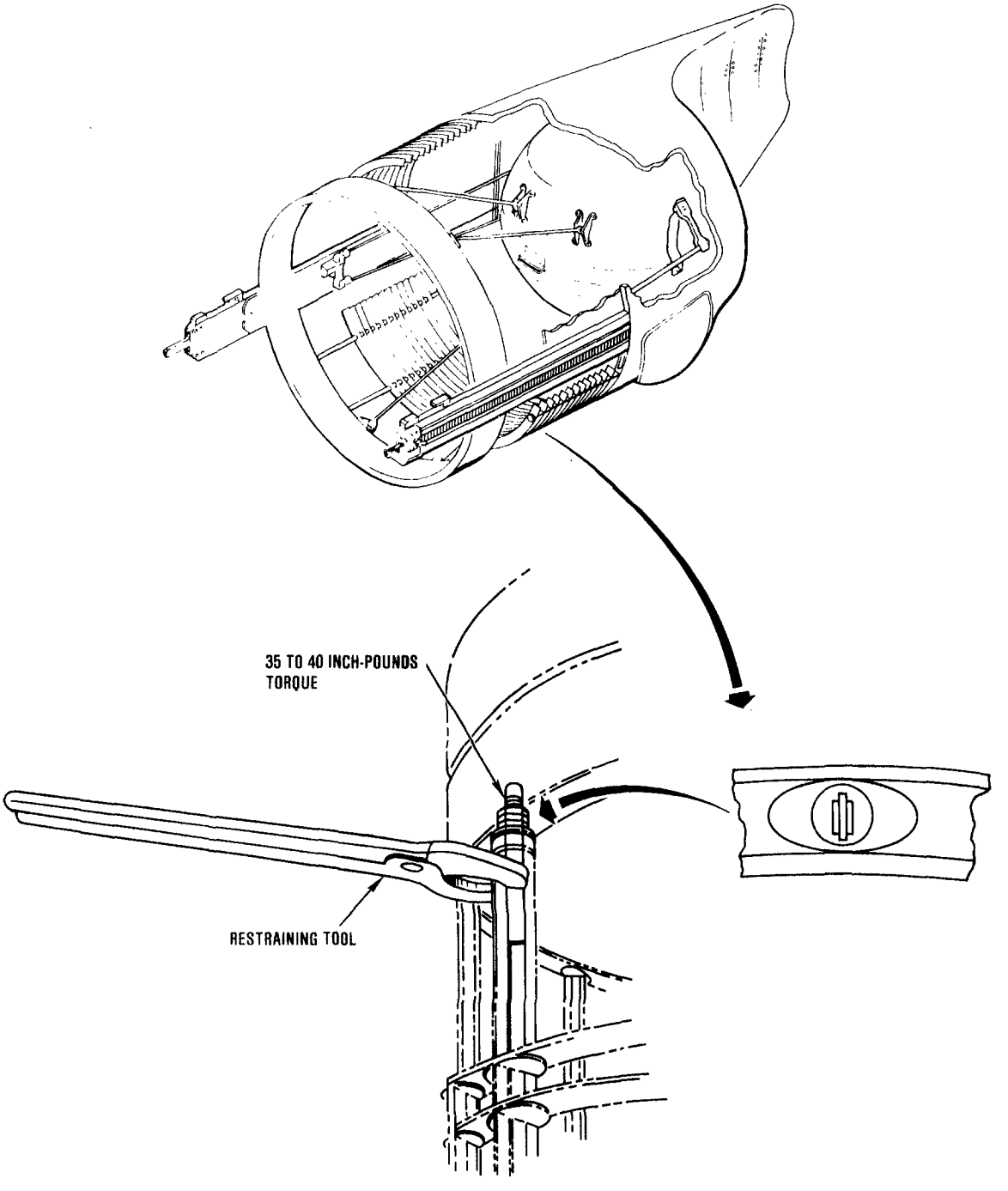
MAINTENANCE PRACTICES (Continued)



S380-2-41-10

Translating Structure Torque
Figure 78-30-5

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(3) Disconnect two electrical disconnect plugs from pneumatic actuator assembly.

(4) Remove clamp securing engine bleed air inlet line to nacelle aft fairing structure.

(5) Disconnect engine bleed air inlet line from pneumatic actuator assembly (four bolts). Install dust cover over open line.

(6) Disconnect small lines from bleed-off valve and pressure switch (see figure 78-30-3) from pneumatic actuator assembly. Install dust covers over open lines.

(7) Remove actuator follow-up cable.

(8) Remove four bolts securing the pneumatic actuator assembly and spur gearbox to the nacelle aft fairing structure. Remove assembly from nacelle aft fairing.

B. Installing pneumatic actuator assembly (P/N 126372-4-(), AiResearch) and spur gearbox, proceed as follows:

(1) Position pneumatic actuator assembly and spur gearbox in nacelle aft fairing and install four bolts, washers and nuts. (Pneumatic actuator assembly is mounted under aft mounting bracket on RH thrust reverser and above aft mounting bracket on LH side.)

(2) Install engine bleed air inlet line to actuator assembly with new gasket (P/N 232-20160-3, Aeronca, Inc.). Install four bolts and lockwire. Install clamp securing engine bleed air inlet line.

(3) Connect and tighten the small lines from the pressure switch and bleed-off valve.

(4) Connect two electrical connectors to actuator assembly and lockwire.

(5) Secure clamp securing engine bleed air inlet line.

(6) Rig thrust reverser system.

(7) Perform an operational check of thrust reverser system. (Refer to "Operation of Thrust Reverser-Engine not Operating.")

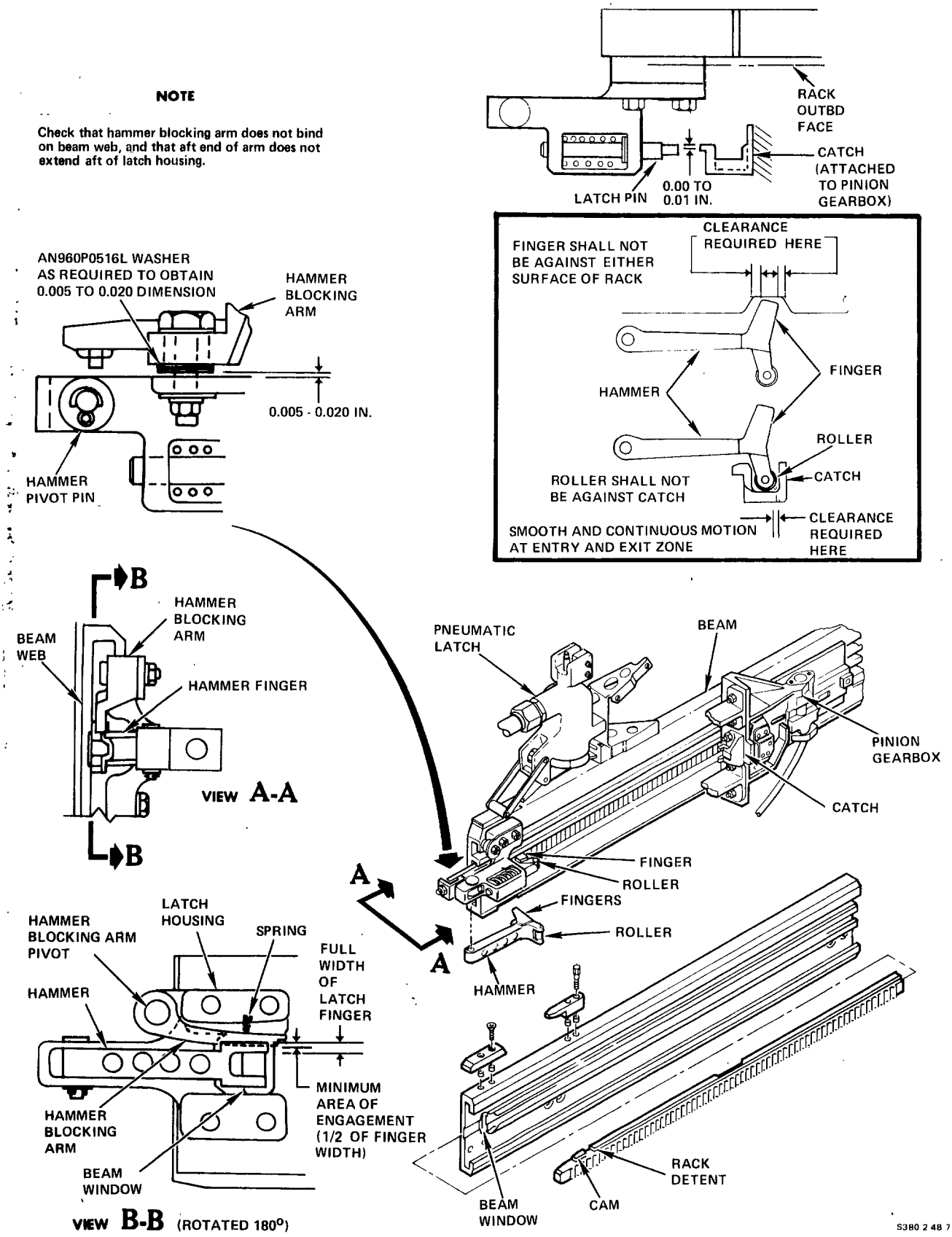
(8) Install nacelle aft fairing on engine.

(9) Perform an operational check of thrust reverser system. (Refer to "Operation Check of Thrust Reverser System-Engine Operating.")

7. Sequencing Latch Replacement

A. Removing sequencing latch assembly. (See figure 78-30-6.)

(1) Remove nacelle aft fairing.



Sequencing Latches Adjustments and Measurements
Figure 78-30-6

(2) Stow thrust reverser by controlled air or manually.

(3) Remove four nuts, washers and screws mounting latch housing to beam.

CAUTION: A spring between the latch housing and hammer blocking arm will propel out if not retained when removing the latch housing from beam. (See figure 78-30-6.)

(4) Hold blocking arm to retain spring and remove latch housing from beam.

B. Installing sequencing latch assembly. (See figure 78-30-6.)

NOTE: The following adjustments procedures must be performed if a new sequencing latch assembly is installed or a catch locking pin, hammer, or hammer blocking arm is replaced.

(1) Install sequencing latch (with latch pin) on beam.

NOTES: At this time it is not important as to the size of the latch pin.

Record all measurments as they are taken in the following steps.

(2) Using the largest diameter (5/16 in. portion) of the latch pin, measure from pin to rack teeth. Record measurement.

(3) Measure between catch (attached to pinion gearbox) to rack teeth. Record measurement.

(4) Subtract the measurements obtained in steps (2) and (3) and multiply this number by two.

(5) Using the answer obtained in step (4), subtract this from 0.312. This number is then used to select the correct pin. The pin size selected should be nearest equal to but not less than the answer obtained in this step.

(6) Remove sequencing latch and install correct size latch pin. Install sequencing latch on beam.

(7) Check that hammer blocking arm does not catch on beam lip and that 0.010 in. clearance exists between end of hammer and beam slot. (See figure 78-30-6.)

(8) Position translating structure as required so that latch pin touches the catch. Check that latch pin is flush (0.00 in.) with lip of catch or 0.01 in. farther from rack than lip of catch.

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(9) Check engagement of hammer blocking arm to latch finger. (See figure 78-30-6.)

(10) Translate the structure toward deploy. As the latch (hammer roller) approaches the point at which the latch engages the catch, visually check that the latch enters the catch with a smooth continuous motion and has a clearance. (See figure 78-30-6.) Also, check other sequencing latch roller clearance.

NOTE: If hammer roller has no clearance as shown in figure 78-30-6, add or subtract shim behind catch to obtain the required clearance. If shims are added or removed, perform steps (2) thru (6).

(11) Hand translate the rack towards the stow position. As the unit approaches the point at which the latch (hammer roller) disengages with a smooth continuous motion and that clearance exists between the rack detent and hammer fingers. (See figure 78-30-6.) Also, check other sequencing latch (fingers) clearances.

(12) Check latch for security and proper safeties. Rig thrust reverser.

(13) Check reverser operation.

(14) Install aft fairing on engine.

8. Replacement of Pinion Gearbox

A. Removing inbd pinion gearbox (see figure 78-30-7), proceed as follows:

(1) Remove translating structure.

(2) Remove reverser retention bolt fitting by removing two bolts and washers.

(3) Remove lock-nut and washer from pinion flex drive shaft in pinion gearbox.

(4) Remove two cap screws and washers securing the flex drive shaft end to the pinion gearbox and remove shaft from gearbox.

(5) Detach follow-up cable by removing nut on feed back slider bracket, and push-pull rod nut on gearbox. (See figures 78-30-16 and 78-30-18.)

(6) Remove four bolts and washers attaching the pinion gearbox to bracket frame (Tee-brackets) and one screw on aft end of gearbox to bracket and remove gearbox from nacelle.

NOTE: When removing pinion gearbox from nacelle, record shims between gearbox mounting flange and nacelle frame and record number of shim washers on aft mounting screw.

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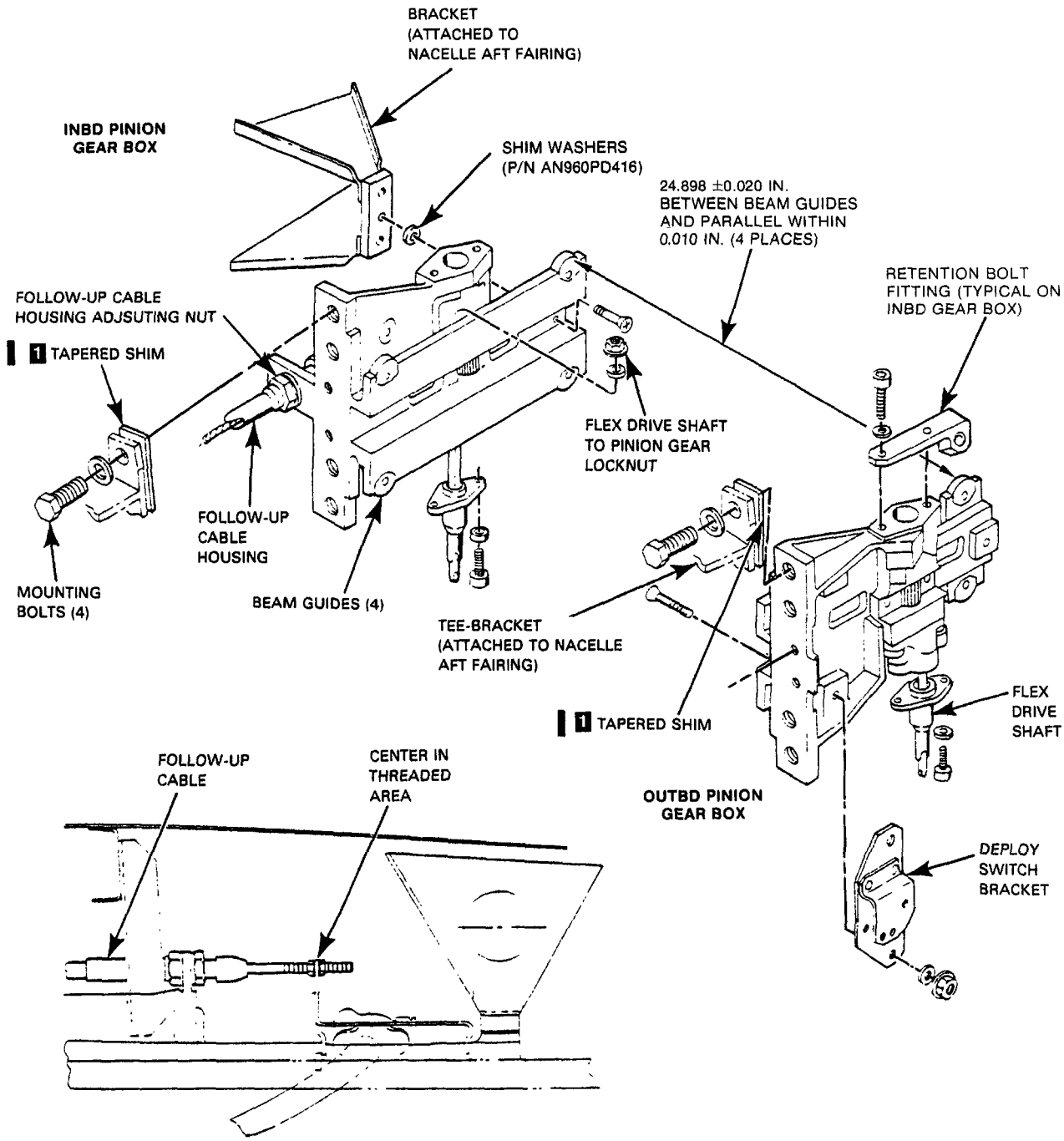

NOTE
CHEM FILM TREAT
SHIMS (ALODINE 1200 OR
EQUIVALENT) AFTER FABRICATION

SHIM MATERIAL AND SIZE (FABRICATE TAPER AS REQUIRED

1 ALUMINUM ALLOY 2024 T3 OR T4
0.010 X 0.060 X 1.0 X 6.0 IN.

ALUMINUM ALLOY (5052, 1100 OR 1145, TEMPER H19)
ALL LAMINATION STOCK (0.002 OR 0.003 IN. LAMINATION)
0.100 X 2.1 X 1.5 IN.

Pinion Gearbox Removal and Installation
Figure 78-30-7

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B. Installing inboard pinion gearbox housing (see figure 78-30-7), proceed as follows:

(1) Place pinion gearbox and shim or shims on nacelle aft fairing tee-brackets (shims recorded at removal) and install four bolts NAS 144-13 (NAS 144-14 and -15 optional) and washers (MS 20002C4).

(2) Attach gearbox aft mounting to nacelle bracket with screw and shims (shims recorded in removal).

(3) Measure the distance (24.898±.020 inches) between the inside surfaces of both gearboxes beam guides (four places).

(4) Check the parallel measurement of the four guides of the inboard gearbox to the four guides of the outboard gearbox. The beam guides must be parallel within .010 inch and still be within the 24.898±.020 inches tolerance.

(a) If a parallel adjustment has to be made, remove or add shim washers (P/N NA 960PD416) to aft mounting screw. For further adjustment, a tapered shim can be used between pinion gearbox forward mounting and tee-brackets. Shim outline must be within .03 inch of the outline of the gearbox mounting surface.

NOTE: Tapered shims must have a Chem Film treatment on surface before installation. Apply Alodine 1200 or equivalent.

(5) Rig thrust reversers.

(6) Install translating structure.

(7) Clean and lubricate rack and beam assemblies.

C. Removing outboard pinion gearbox (see figure 78-30-7), proceed as follows:

(1) Remove translating structure.

(2) Remove reverser retention bolt fitting by removing two bolts and washers. (See figure 78-30-7.)

(3) Remove lock-nut (P/N NAS1291C4M, 5/16 inch socket) and washer (P/N AN960-416) from pinion flex drive shaft in pinion gearbox.

(4) Remove two cap screws and washers securing the flex drive shaft end to the pinion gearbox, and remove flex shaft from gearbox.

(5) Detach thrust reverser deploy switch bracket from gearbox. (Refer to Removing Thrust Reverser Deploy Switch.)

(6) Remove four bolts and washers from nacelle frame (tee-brackets). Remove pinion gearbox from nacelle.

NOTE: When removing pinion gearbox from nacelle, record shims between gearbox flange and nacelle frame.

D. Installing outbd pinion gearbox housing (see figure 78-30-7), proceed as follows:

(1) Place pinion gearbox housing and shim or shims on nacelle aft fairing tee-brackets (shims recorded at removal) and install four bolts and washers.

(2) Measure the distance (24.898±0.020 in.) between the inside surfaces of the inside surfaces of both gearboxes beam guides (four places).

(3) Check the parallel measurement of the four beam guides of the inbd gearbox to the four beam guides of the outbd gearbox. The beam guides must be parallel within 0.010 in. and still be within the 24.898 in. tolerance.

(a) If a parallel adjustment has to be made, remove or add shim washers to aft mounting screw on inbd gearbox, or tapered shims can be used behind gearbox mounting flange.

NOTE: Tapered shims must have a Chem Film treatment on surface before installation. Apply Alodine 1200 or equivalent.

(4) Rig thrust reversers.

(5) Install thrust reverser deploy switch bracket with two screws, washers, and locknuts. (Refer to Installing Thrust Reverser Deploy Switch.)

(6) Clean and lubricate rack and beam assemblies.

9. Functional Check of Thrust Reverser Sequencing Latches

A. To perform a function check of the thrust reverser sequencing latches (see figure 78-30-6), proceed as follows:

(1) Remove nacelle aft fairing.

(2) With the thrust reverser in the stowed position, check that the spring-loaded latch pins operate freely and return to the fully extended position after being manually retracted against spring return pressure and released. Accomplish this check at varying degrees of compression from near full extention to full compression.

(3) Check that the spring-loaded latch pins can be rotated 360 degrees in housing when fully retracted without binding.

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(4) Translate the thrust reverser toward the deployed position. Check that both spring-loaded latch pins contact the forward faces of the catches at the same time and that the rollers on the latches move in and out of the catches freely.

(5) Check that at full deployment, the manner latch rollers are centered in; but do not touch the aft inside face of the catch, and that the hammer blocking arms cover a minimum of one-half of the face of the inboard hammer latch fingers (see figure 78-30-6).

(6) Translate the thrust reverser toward the stowed position and check for the following:

(a) The hammer blocking arm is smoothly rotated clear of the hammer latch finger by the ramp on the horizontal surface of the rack.

(b) The hammer latch fingers contacts the ramp on the vertical surface of the rack, aft of the ramp leading edge, and the hammer latch is forced slightly outboard.

(c) The hammer latch roller leaves the catch, passes through the rack detents and enters the beam with a smooth continuous motion.

(7) Check the hammer latch fingers for evidence of wear in the area where they contact the hammer blocking arm outboard surface at the point of contact with the hammer latch fingers. There should be no significant indications of wear or "rounding off."

(8) Perform an operational check of the thrust reverser system. Refer to "Operation of Thrust Reverser - Engines Not Operating."

(9) Reinstall nacelle aft fairing on engine.

10. Adjusting Thrust Reverser Control Lever Stop

A. To adjust the thrust reverser control lever stop, refer to 76-00-0.

11. Adjusting Thrust Reverser 80 Percent RPM Limit Switch

A. To adjust the thrust reverser 80 percent rpm limit switch (power safety switch), refer to 76-00-0.

12. Removing Thrust Reverser Deploy Switch

A. To remove the thrust reverser deploy switch (see figure 78-30-17), proceed as follows:

NOTE: Do not disturb the pinion gearbox mounting when performing the following procedures.

(1) Fully deploy thrust reverser. Refer to "Operation of Thrust Reverser - Engines Not Operating."

(2) Remove access door on outboard side of nacelle aft fairing, adjacent to outboard pinion gearbox.

(3) Remove two screws securing the switch bracket to the outboard pinion gearbox.

NOTE: On some reverser installations, the two deploy switch mounting bracket screws have a 3/32 inch Allen recess in the end of screw (P/N HL21-8-4), this aids in holding the screw while removing nuts.

(4) Move switch bracket outboard and aft to remove the complete assembly from the outboard pinion gearbox.

(5) Remove nut, washers and screws securing switch and roller actuating arm in switch bracket.

(6) Remove switch and roller actuating arm from switch bracket.

(7) If switch requires replacement, refer to "Replacing Thrust Reverser Deploy Switch."

13. Installing Thrust Reverser Deploy Switch

A. To install the thrust reverser deploy switch in the nacelle aft fairing (see figure 78-30-17), proceed as follows:

(1) Install switch and roller actuating arm in switch bracket and secure in place with screws, washers and nuts.

NOTE: Ensure that thrust reverser is fully deployed.

(2) Through access opening in outboard side of nacelle aft fairing, install switch bracket assembly to outboard pinion gearbox with two screws.

(3) Position thrust reverser switch to OFF and apply external electrical power or position electrical master and battery switches at ON.

(4) Check that DEPLOY light is illuminated. If DEPLOY light is not illuminated, carefully bend switch roller actuator arm as required to illuminate the DEPLOY light.

(5) Stow thrust reverser. Check that DEPLOY light goes out.

(6) Check complete installation for security and clearances.

(7) Perform an operational check of thrust reverser system. Refer to "Operation of Thrust Reverser - Engine Not Operation."

(8) Install access door on nacelle aft fairing.

14. Replacing Thrust Reverser Deploy Switch

A. To replace the thrust reverser deploy switch (see figure 78-30-17), proceed as follows:

(1) Remove the deploy switch from the nacelle aft fairing. Refer to Removing Thrust Reverser Deploy Switch.

(2) Remove potting from switch electrical connections.

(3) Remove switch from electrical wiring harness.

(4) Connect replacement switch to electrical wiring harness.

(5) Pot electrical switch connections.

NOTE: Potting compound to be used are as follows:

- RTV 8262 - Color red with primer SS-4004
  Catalyst 9858
  Manufacturer: General Electric Company
  Waterford, New York 12188

- Alternate Compound:
  RTV 833
  Cure Agent CD - Color red with primer SWS407
  Manufacturer: Stauffer-Wacker
  Silicon Corp.
  Adrian, Michigan 49221

- Pot life for both compounds is 4 hours.
  Cure time at room temperature is 72 hours; however, the switch may be installed, and aircraft operated 8 hours after applying compound.

(6) Install deploy switch in nacelle aft fairings. Refer to Installing Thrust Reverser Deploy Switch.

15. Removing Pneumatic Latch(es)

A. To remove the pneumatic latch(es), proceed as follows:

(1) Remove nacelle aft fairing from engine and position in a suitable stand.

(2) Remove the pneumatic line to latch. Cap and plug open fittings.

NOTE: Prior to removing the pneumatic latch(es), count and retain the washers installed between the latch and its mounting bracket. This will aid in rigging during the re-installation procedures.

(3) Remove the two screws securing the electrical switches to the latch. Remove switches from latch.

(4) Remove the three screws that secure the latch to supporting bracket. Retain spacer washers in with each mounting screws (see figure 78-30-13).

(5) Remove latch from nacelle aft fairing.

16. Replacing Pneumatic Latch(es)

A. To install the pneumatic latch, proceed as follows:

(1) Remove union from removed latch and discard "O" ring packing.

(2) Install new "O" ring packing (MS28778-4) on union and install and tighten in replacement latch.

(3) Install the three screws through latch attaching bracket; then install the spacer washers (AN960PD10L) that were removed during the removal procedures.

(4) Position the replacement latch in position and tighten the three mounting screws.

(5) Rig latch (refer to rigging procedures).

(6) Connect pneumatic line to latch.

(7) If pneumatic latch electrical switches are to be replaced, refer to "Pneumatic Latch Electrical Switch Replacement."

(8) Ensure that the switch(es) actuator roller arm (figure 78-30-13) is bent to the proper degree (10° to 15°), then install the latch electrical switch assembly and tighten mounting screws.

(9) Check that the latch plunger operates freely. The plunger is spring-loaded to the latched position.

(10) Depress the latch plunger from the switch end of latch, and check that switch deactuates. Switch deactuation is verified by an audible indication of two distinct clicks.

(11) While manually depressing the latch plunger from switch end, allowing latch plunger to return to its normal latched position, check that both switches actuate (two distinct clicks).

(12) Manually apply pressure to the latch plunger in the forward and aft direction with the plunger in the extended position, then the retracted position. Check that the switches actuate or deactuate with vertical latch plunger movement only.

(13) Partially deploy thrust reverser. Remove air source from reverser.

(14) Manually depress latch plunger and release. Check that the latch roller arms are retained in the unlatched position and that the switches remain in a deployed condition.

(15) Check complete installation for safeties, clearances and security.

(16) Perform an operational check of the thrust reverser system. (Refer to Operation of Thrust Reverser-Engines Not Operating.)

(17) Install nacelle aft fairing on engine.

17. Replacement of Pneumatic Latch Switches

A. To remove pneumatic latch switches, proceed as follows:

(1) Remove nacelle aft fairing from engine. Stow thrust reverser.

(2) Remove the two screws securing the switches to pneumatic latch and remove switches from latch.

(3) Carefully cut potting compound from switch terminals.

(4) Identify each wire to its respective switch terminal.

(5) Remove electrical wires from switches.

B. To install pneumatic latch switches, proceed as follows:

(1) Check replacement switch for proper angle (10° to 15°) of the switch actuator roller arm. (See figure 78-30-13.) Adjust (bend) roller arm as required.

(2) Connect electrical wires to switches.

(3) Pot electrical switch connections.

NOTES: Potting compound to be used are as follows:

- RTV 8262 - Color red with primer SS-4004
  Catalyst 9858
  Manufacturer: General Electric Company
  Waterford, New York 12188

- Alternate Compound:
  RTV 833
  Cure Agent CD - Color red with primer SWS407
  Manufacturer: Stauffer-Wacker
  Silicon Corp.
  Adrian, Michigan 49221

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- Pot life for both compounds is 4 hours. Cure time at room temperature is 72 hours; however, the switch may be installed, and the aircraft operated 8 hours after applying compound.

(4) After adequate potting cure time, place switch in pneumatic latch and install and tighten the mounting screws.

(5) Stow thrust reverser (if applicable).

(6) Check that the latch plunger operated freely. The plunger is spring loaded to the latched position. (See figure 78-30-13.)

(7) Depress the latch plunger from the switch end of latch, and check that switch deactuates. Switch deactuation is verified by an audible indication of two distinct clicks. Adjust tab if required.

(8) While manually depressing the latch plunger from switch end, allow latch plunger to return to its normal latched position. Check that both switches actuate (one or two distinct clicks).

(9) Manually apply pressure to the latch plunger in the fwd and aft direction with the plunger in the extended position, then the retracted position. Check that the switches actuate or deactuate with vertical latch plunger movement only. If switches actuate or deactuate with fwd and aft latch plunger movement, repeat steps (7) and (8).

(10) Partially deploy thrust reversers. Remove air source from reverser.

(11) Manually depress latch plunger and release. Check that the latch roller arms are retained in the unlatched position and that the switches remain in a deployed condition. If either switch switches to a latched condition, repeat steps (7) and (8).

(12) Check complete installation for safeties, clearances, and security.

(13) Perform an operational check of the thrust reverser system. (Refer to Operation of Thrust Reverser - Engines Not Operating.)

(14) Install nacelle aft fairing on engine.

18. Replacement of Flexible Drive Shaft, Pneumatic Actuator to Spur Gearbox

A. To replace the flexible drive shaft between air motor and spur gearbox, proceed as follows:

(1) Remove nacelle aft fairing.

(2) Remove mounting bolt and washer from spur gearbox to nacelle frame clip.

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(3) Remove the four bolts attaching spur gearbox to actuator. Separate gearbox from actuator and remove flex drive shaft.

(4) Align flex drive shafts (refer to rigging procedures) and install shafts.

(5) Install bolt, washer attaching spur gearbox to nacelle frame clip.

(6) Perform "Operational Check of Thrust Reverser System - Engine Not Operating."

(7) Install nacelle aft fairing.

19. Tailpipe Seal Repairs

A. Minor tailpipe seal repairs.
Minor repairs are considered small longitudinal cuts or tears and wear areas through the outer surface of tailpipe seal.

(1) Repair the damaged areas by covering the RTV 140. (General Electric Company, Waterford, New York 12188).

B. Major tailpipe seal repairs.
Major repairs is when a segment of seal has to be replaced or a complete seal replacement.

(1) Replacing a section of tailpipe seal. (Aeronca, Inc. P/N 232-20239-1).

(a) Remove rivets from seal retainers adjacent to damaged area and remove segment.

NOTE: Carefully remove rivets to avoid enlarging the 0.128 to 0.1285 - inch diameter holes.

(b) Cut new seal to right length (use damaged seal as pattern).

(c) Place new seal segment under seal retainers and install rivets (MS20613-4C3).

NOTE: Where the retainer seals overlap, use MS20613-4C4 rivets.

(d) Apply RTV 140 (General Electric Company, Waterford, New York 12188) over both ends of seal segment.

(2) Replacement of entire seal (Aeronca, Inc. P/N 232-200239-1).

(a) Mark seal retainers before removing to make sure retainers are installed in the same position.

(b) Remove all rivets (approximately 74 rivets) from both seal retainers and remove seal.

NOTE: Carefully remove rivets to avoid enlarging the 0.128 to 0.1285 inch diameter holes.

(c) Install seal and seal retainers around tailpipe with Cleco fasteners or equivalent fastener (check alignment marks) and install rivets (MS20613-4C3).

NOTE: Where the retainer seals everlap, use MS20613-4C4 rivets.

(d) Apply RTV 140 (General Electric Company, Waterford, New York 12188), where seal ends meet.

20. Installing Thrust Reverser Retention Bolts

A. In the event of a thrust reverser failure and continued operation of the airplane is necessary, the thrust reversers may be locked in the stowed position using the thrust reverser retention bolts carried as loose equipment aboard the airplane. This will allow the airplane to be operated (without thrust reversers) until the thrust reverser problem can be corrected.

(1) To install thrust reverser retention bolts in thrust reversers that have failed in the stowed position, proceed as follows:

(a) Obtain thrust reverser retention bolts; two long bolts and two short bolts, from loose equipment pouch in airplane.

(b) Make certain that thrust reverser is in the stowed position.

(c) Install the two short retention bolts through the $\frac{1}{2}$ inch-diameter holes in the access panel located on the nacelle aft fairing outboard centerline just forward of the translating fairing split line. (See figure 78-30-9.) The retention bolts must be engaged a minimum of 18 full turns and extend through the retention fitting into the retention slot machined into the translating beam. Torque retention bolts to 15-20 inch-pounds.

(d) Install the two long thrust reverser retention bolts through the $\frac{1}{2}$ inch-diameter holes on the inboard side of the nacelle aft fairing. The $\frac{1}{2}$ inch-diameter holes are located approximately three inches above the inboard centerline and three inches forward of the translating fairing split line. The retention bolts must be engaged a minimum of 18 full turns through the retention fitting into the retention slot machined into the translating beam. Torque retention bolts to 15-20 inch-pounds.

(2) If the thrust reverser has failed in the deployed poistion, it will be necessary to hand translate the thrust reverser to the stowed position before the retention bolts can be installed. To hand translate the thrust reverser to the stowed position, proceed as follows:

(a) Remove three access doors on the lower side of the nacelle aft fairing; one on each side of the pneumatic actuator and one forward of the pneumatic actuator.

(b) Through the access doors on each side of the pneumatic drive spur gearbox, remove the two internal wrenching screws securing the flex drive shaft to each side of the pneumatic drive spur gearbox.

(c) Remove flex drive shaft ends from pneumatic drive gear spur box and position shaft ends to be accessible through access doors.

(d) Using two 3/8 inch drive rachets, 10 to 15 inch extensions and 5/16 inch sockets, place sockets on ends of flex drive shafts (see figure 78-30-16). Simultaneously turn both flex drive shafts in the same direction to drive the thrust reverser to the stowed position.

(e) With the thrust reverser held in the stowed position, install thrust reverser retention bolts as indicated in step (1) of this procedure.

(f) Remove sockets from flex drive shafts. Stow flex drive shafts and install access doors.

21. Blocker Doors Lever Arms Penetrant Inspection - P/N 232-20258-505, -506, -507, -508, -509 and -510

NOTE: This penetrant inspection is not required, if blocker door lever arms have part numbers 232-20258-511 and -512.

A. Removing and installing blocker doors lever arms for a penetrant inspection, is as follows:

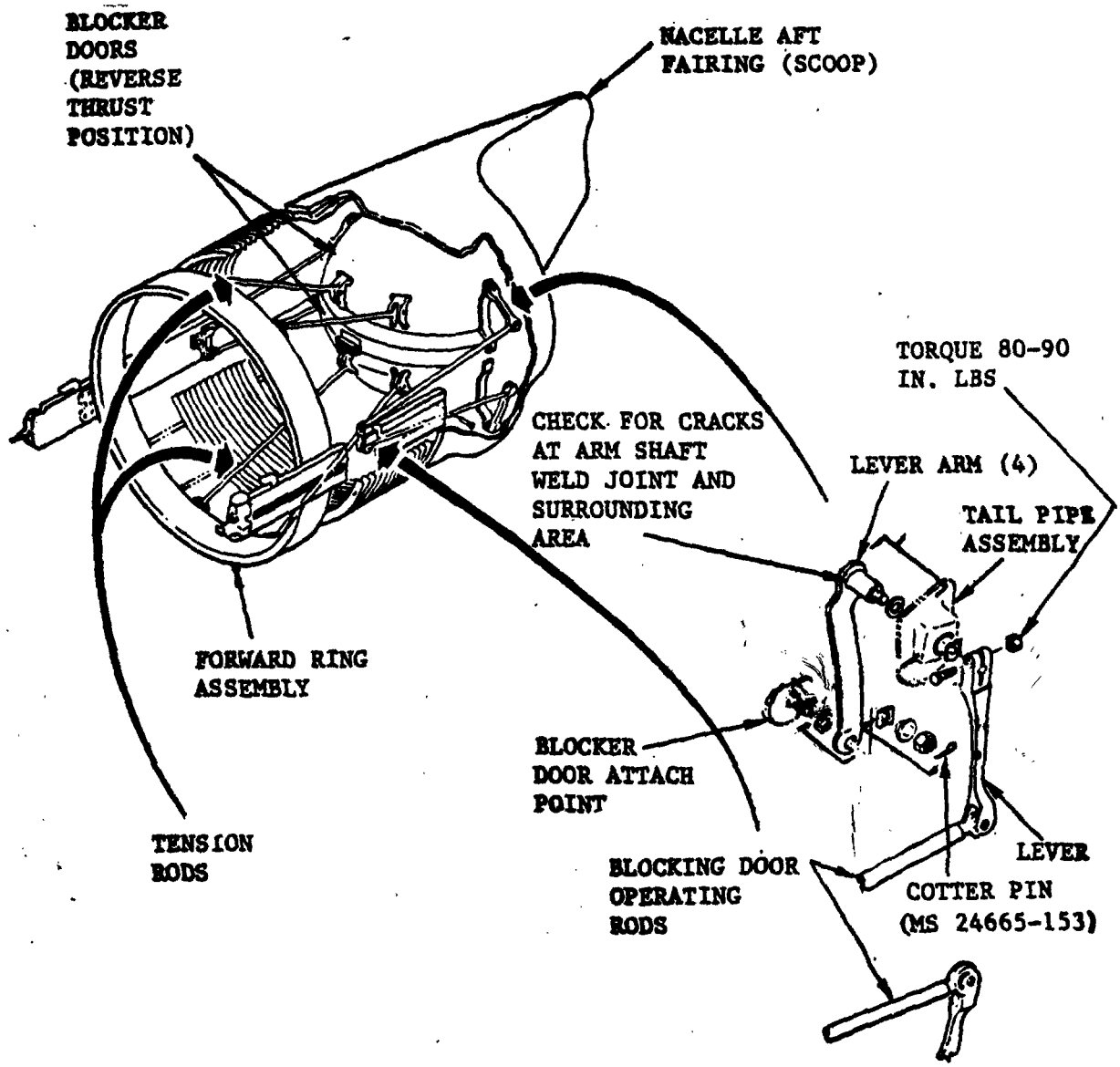
NOTE: Refer to figure 78-30-7A for reference to the following steps:

(1) Remove nacelle aft fairing (scoop) and deploy thrust reversers.

(2) Remove nut, screw, and spring washer attaching lever to lever arm (four places) and remove levers from lever arm shafts.

(3) Detach the lower blocker door by springing door and lever arm to remove arm shaft from tailpipe assembly and washer (two places).

(4) Rotate the blocker doors by hand into a forward thrust position (doors parallel).


BLOCKER DOORS (REVERSE THRUST POSITION)
NACELLE AFT FAIRING (SCOOP)
TORQUE 80-90 IN. LBS
CHECK FOR CRACKS AT ARM SHAFT WELD JOINT AND SURROUNDING AREA
LEVER ARM (4)
TAIL PIPE ASSEMBLY
FORWARD RING ASSEMBLY
BLOCKER DOOR ATTACH POINT
TENSION RODS
BLOCKING DOOR OPERATING RODS
LEVER
COTTER PIN (MS 24665-153)

NOTE: The shown blocker door lever arm and lever assemblies are typical in four places.

Lever Arms Penetrant Inspection
Figure 78-30-7A

(5) Detach upper blocker door by springing door and lever arm to remove arm shaft from tailpipe assembly and washer (two places).

(6) From the exhaust nozzle end of the reverser, remove cotter pin, nut, bushing, and spring washer from the blocker door attach point (four places).

(7) Remove four lever arms and spacers, and tag each arm as removed, for later reinstallation in their original position.

(8) Thoroughly clean and air dry the four lever arms, using Methyl Ethyl Keytone (Federal Specification TT-M-2610) or equivalent solvent.

(9) Inspect each lever arm using Zyglo or dye penetrant methods.

NOTE: Check for cracks and give particular attention to the shaft to arm weld and to the arm body in areas near the arm shaft juncture.

(10) Replace all lever arms showing evidence of cracks.

(11) Install spacer on each lever arm shaft and position lever arms in their original position on the tail pipe assembly.

(12) Install washer on each blocker door attach point (four places).

(13) Attach upper blocker door attach points into lever arms, by springing doors. Install spring washer, bushing, nut and new cotter pin on end of shaft (two places).

(14) Attach lower blocker door attach points into lever arms, by springing doors. Install spring washer, bushing, nut and new cotter pin on end of shaft (two places).

(15) Rotate the blocker doors to a reverse thrust position by hand.

(16) Attach lever arm to lever in their original position with spring washer, screw and nut (torque 80-90 inch pounds, four places).

(17) Check complete assembly for proper safeties, clearances and security of all attaching hardware.

(18) Operate thrust reverser with engines not operating. Refer to "Operation of Thrust Reverser - Engines Not Operating" in this chapter.

NOTE: If blocker doors are out of adjustment, refer to the applicable rigging procedures in this section

(19) Install nacelle aft fairing (scoop).

22. Replacement of Directional Control Valve Solenoid

A. Removing directional control valve solenoid assembly (P/N 320295-2 or -3, AiResearch) with nacelle aft fairing installed. (See figure 78-30-8.) Proceed as follows:

(1) Remove nacelle aft fairing lower access door.

(2) Disconnect electrical connector from valve solenoid.

(3) Remove Allen screw from mounting clip at forward end of valve solenoid.

NOTE: All Allen head screws removed, require a 9/64 inch Allen wrench.

(4) Remove three Allen head screws attaching solenoid valve assembly to pneumatic actuator assembly.

(5) Move valve solenoid assembly outboard (from engine centerline) to disengage pneumatic line from elbow. Remove unit from nacelle.

(6) Remove mounting clip from forward end of valve solenoid.

B. Installing directional control valve solenoid assembly (P/N 320295-2 or -3, AiResearch) with nacelle aft fairing installed, proceed as follows: (See figure 78-30-8.)

NOTE: The directional control valve solenoid that was removed has to be replaced with the same Part No. dash number (-2 or -3).

(1) Replace two "O" rings (P/N S8990-008, AiResearch) on mating surface of actuator and one "O" ring (P/N S8990-010, AiResearch) on pneumatic line end fitting. (P/N 3230171-1, AiResearch).

(2) Install mounting clip on forward end of valve solenoid.

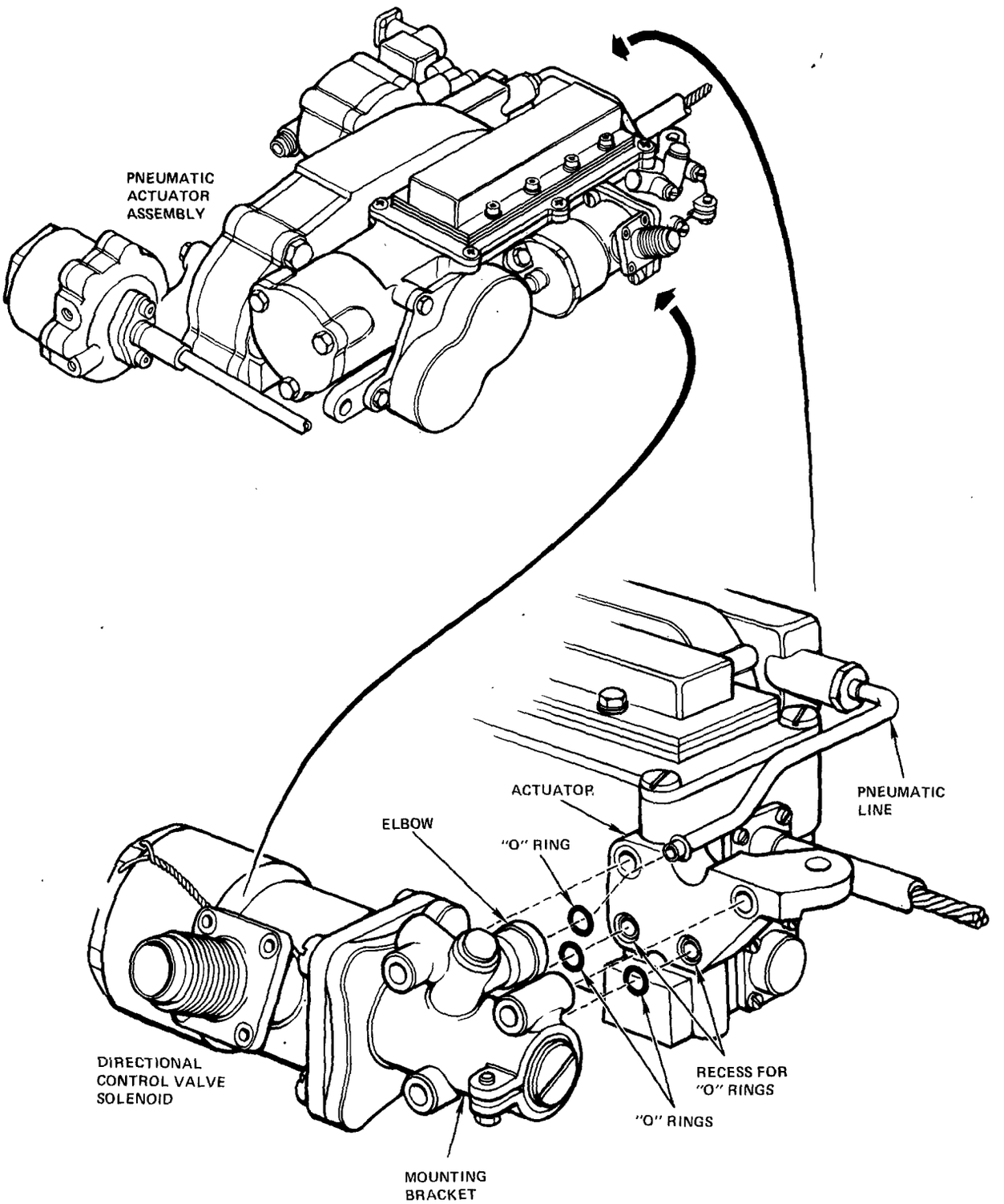
(3) Engage the valve solenoid (elbow) to the pneumatic line fitting and then align the valve solenoid assembly to the pneumatic actuator mating flange. Install three attachment Allen screws.

(4) Install attachment Allen screw in forward mounting clip.

(5) Connect electrical connector to solenoid valve and lockwire.

(6) Perform "Operational Check of Thrust Reverser System (Engine operating or not operating.")

(7) Install nacelle aft fairing access door.



S380 2 48 15

Replacing Directional Control Valve Solenoid
Figure 78-30-8

C. Removing directional control valve solenoid assembly (P/N 320295-2 or -3, AiResearch), with pneumatic actuator assembly not installed in nacelle, proceed as follows: (see figure 78-30-8.)

(1) Remove Allen head screw from mounting clip from forward end of valve solenoid.

NOTE: All Allen head screws removed, require a 9/64 inch Allen wrench.

(2) Disconnect pneumatic line (P/N 3230171-1, AiResearch) to the pneumatic actuator assembly at the actuator fitting and remove pneumatic line.

(3) Remove three Allen head screws attaching solenoid valve assembly to pneumatic actuator assembly.

(4) Remove valve solenoid assembly from pneumatic actuator assembly.

(5) Remove mounting clip from forward end of valve solenoid.

D. Installing directional control valve solenoid assembly (P/N 320295-2 or -3, AiResearch) with pneumatic actuator assembly not installed in nacelle proceed as follows: (See figure 78-30-8.)

NOTE: The directional control valve solenoid that was removed has to be replaced with the same Part No. dash number (-2 or -3).

(1) Replace two "O" rings (P/N S8990-008, AiResearch) on mating surface of actuator.

(2) Install the mounting clip on the forward end of valve solenoid.

(3) Align the valve solenoid assembly to pneumatic actuator mating flange. Install three attachment Allen screws.

(4) Install attachment screw in forward mounting clip.

(5) Install "O" ring (P/N S3990-010, AiResearch) on the pneumatic line end fitting (elbow end), and install line.

23. Replacement of Follow-up Cable

A. Removing follow-up cable (P/N 3230113-3, Airesearch), proceed as follows:

(1) Remove nacelle aft fairing.

(2) Remove five cover mounting screws.

NOTE: When removing screws from cover, note each screw location. Screws have different lengths.

CAUTION: DO NOT TIP THE COVER FORE AND AFT DURING REMOVAL, DAMAGE COULD RESULT.

(3) Carefully lift the cover straight up from the actuator assembly. As the cover is lifted, the guide (attached to cover) disengages from the follow-up cable and control valve rod ball end. (See figure 78-30-21.)

(4) Remove two screws attaching the follow-up cable housing to actuator assembly and remove follow-up cable from actuator assembly.

(5) Replace cover on actuator assembly to prevent contamination.

(6) Detach follow-up cable housing from pinion gear box bracket and follow-up cable from feedback slider. (See figure 78-30-17.)

(7) Remove clamp from follow-up cable. Note route of cable and then remove cable from nacelle.

(8) If follow-up cable is to be replaced with a new cable, also perform the following:

(a) Remove two self-locking nuts from pneumatic actuator drive assembly end of follow-up cable.

(b) Drill out two stakes locking flange to pneumatic actuator drive assembly end of follow-up cable housing. Unscrew flange from housing.

B. Installing follow-up cable (P/N 3230113-3, AiResearch), proceed as follows:

(1) If new follow-up cable is being installed, perform steps 1 (a) through 1 (c), otherwise proceed to step 2.
(a) Install flange on pneumatic actuator drive assembly end of new follow-up cable. Tighten until flange is firmly against end of cable housing. Stake housing to flange in some locations as old housing was staked to flange.
(b) Install two self-locking nuts on pneumatic actuator drive assembly end of follow-up cable. See figure 78-30-21 for position and clearances.

(c) Replace feedback slider bracket on inboard pinion gear housing with new slider bracket. (A malfunctioning follow-up cable will cause the slider bracket to be bent during thrust reverser operation, and a bent slider bracket will result in premature follow-up cable failure and erratic thrust reverser operation.)

(2) Carefully route follow-up cable in nacelle as noted in "Removing Follow-up Cable, step (7)."

(3) Rig thrust reverser system.

(4) Install clamp on follow-up cable.

(5) Check complete installation for security and safeties.

NOTE: The following rigging procedures are presented in a manner as to allow the person performing the task to remove the aft fairing from the engine, completely rig the reverser and re-install the aft fairing on the engine. It is recommended that if a component is replaced, the complete reverser should be rigged in sequence as per the following procedures.

- All illustrations for the rigging procedures are provided at the end of the procedures. This allows the mechanic to remove the applicable illustration and lay it adjacent to the procedure that is being performed. Two locally manufactured rigging tools may also be fabricated (see figures 78-30-24 and 78-30-25) to aid in the rigging procedures.

24. Removal of Nacelle Aft Fairing (Scoop)

A. Prior to removing the nacelle aft fairing (scoop), perform the following steps:

(1) Remove nacelle aft fairing access panels and doors (see figure 78-30-9.)

(2) Check for alignment of nacelle aft fairing (scoop). A 1/16 to 1/8-inch gap should exist between the scoop and the engine aft nacelle. If the scoop is in solid contact with the nacelle, re-rigging will be required during installation. Note the interference or contact points so that shims may be added during installation.

(3) Mark position of nacelle aft fairing (scoop) for alignment with nacelle aft fairing (see figure 78-30-11).

(4) Connect thrust reverser test set and deploy the reverser. Shut-off air and electrical power. See figure 78-30-37 and 78-30-38 for locally manufactured test sets.

(5) Remove the nacelle aft fairing (scoop) as follows:

NOTE: During removal of scoop attaching bolts and screws, care should be taken not to drop the shims.

(a) Remove angle from tailpipe assembly by removing two bolts (AN3C-3A). See figure 78-30-11.

(b) Remove angle from nacelle aft fairing (scoop) frame by removing two cap screws (NAS1351C38) and two washers (AN960C10L).

(c) Remove longeron from nacelle aft fairing (scoop) frame by removing one nut (MS21043-3) and one washer (AN960C10L).

(d) Remove four cap screws (NAS1351-C3-12) securing nacelle aft fairing (scoop) to tailpipe and pull nacelle aft fairing (scoop) aft to clear thrust reverser nozzle (tailpipe).

25. Thrust Reverser Alignment Check

A. To perform alignment check, proceed as follows:

(1) Deploy reverser then reduce air pressure and move reverser towards stow until the translating structure is 12 to 14 inches from the stow position.

(2) Disconnect the four tension strap turnbuckles (see figure 78-30-12).

(3) Check alignment as follows:

(a) Measure the horizontal and vertical distance between the engine tailpipe and the thrust reverser. The pipe should be centered. The vertical distance between the engine tailpipe and the top or bottom of the thrust reverser should be equal. The horizontal distance between the engine tailpipe and either side of the thrust reverser should be equal.

(4) If alignment is not satisfactory, the following items must be performed to determine the amount of shims to be added under the jack pads or that the engine requires alignment. Proceed as follows:

NOTE: Centering jacks have a limited amount of travel. They should not be turned counterclockwise over a maximum of 3 turns.

(a) Turn all four centering adjustment jacks clockwise until they stop (see figure 78-30-10).

(b) To check vertical alignment, turn the upper jack counterclockwise until the tailpipe centers or a maximum of three turns has been made. If the tailpipe fails to center, measure the difference between the upper and lower tailpipe to reverser cascade assembly. This dimension, plus 0.062 inch, will equal the shim thickness that should be added to the upper jack pad on the tailpipe to obtain the proper centering. This shim will be added in a subsequent step.

(c) To check horizontal alignment, the outboard centering jack or the two cantered inboard centering jacks may be turned. If the tailpipe fails to center, measure the difference between the

inboard and outboard tailpipe to reverser cascade assembly. This dimension, plus 0.125 inch, will equal the shim thickness that should be added to the two cantered inboard jacks pads. To shim the outboard jack pad, 0.062 inch must be added to the shim thickness.

NOTE: The two cantered inboard jacks must be turned the same number of turns each.

- If the reverser cannot be aligned properly within the aft nacelle fairing, it is possible that the engine will require alignment. Refer to Chapter 71 for the required engine alignment procedures.

26. Removal of Engine Aft Nacelle Fairing (Thrust Reverser)

A. To remove engine aft nacelle fairing proceed as follows:

(1) Open engine access doors (see figure 78-30-9).

(2) Disconnect electrical and pneumatic connections to the thrust reverser. Cap electrical plugs and all open pneumatic lines.

(3) Remove tension on the four turnbuckles securing the aft fairing to the engine (see figure 78-30-12).

(4) Disconnect the forward bolts and bushings securing the turnbuckles to the tension rods. Retain bolts for installation.

NOTE: Do not leave bolts in engine tension rods.

(5) Back-out (turn clockwise) the four center adjustment jacks (see figure 78-30-10).

(6) Remove two screws securing the forward inboard corners of the aft fairing to the pylon structure.

(7) Remove the nine screws securing the forward aft fairing ring to the aft fairing support ring.

NOTE: Support aft fairing before removing last screw (top center screw).

(8) Move aft nacelle fairing straight aft until clear of the tailpipe, then place in a suitable support fixture or dolly.

27. Removal of Translating Assembly From Aft Nacelle

A. To remove translating assembly from aft nacelle, proceed as follows:

(1) Connect thrust reverser test set.

(2) Stow thrust reverser. Note the alignment of the latch rollers to the beam. If rollers are not aligned they will require shimming prior to installation (see figure 78-30-13).

(3) Shut off air and test set electrical power. Remove test set and disconnect air pressure line and cap.

(4) Mark the relation of sequencing latches to beam, "inboard" and "outboard"; then remove the nuts, washers, and screws securing latches to beam (see figure 78-30-15).

(5) Remove the deploy switch actuating pin from the outboard rack (see figure 78-30-15).

(6) Remove the follow-up cable actuator from the inboard rack. Reinstall nut and lock washer on rack to retain shims.

(7) Remove cap screws securing the flex drive shaft ends to the spur gearbox (gearbox attached to actuator), and remove the flex drive shafts from the spur gearbox.

CAUTION: TO PREVENT DAMAGE TO BEAMS, THE TRANSLATING STRUCTURE MUST BE SUPPORTED AS THE RACKS SEPARATE FROM THE PINION GEARS.

(8) Using ratched and 5/16-inch sockets (figure 78-30-16) on the flex drive shaft ends, manually translate the structure aft and off the pinion gearbox gears.

(9) Move the translating structure aft to clear the beams from the pinion gearbox. Remove translating structure from nacelle aft fairing.

(10) Perform a preliminary inspection of the aft nacelle and list any obvious defects (excessive wear, bent components, twisted arms and etc.).

28. Pneumatic Components Check-Out Procedures

A. To check the pneumatic components of the thrust reverser assembly, proceed as follows:

(1) Connect test sets to thrust reverser.

(2) Manually position the follow-up cable feedback slider bracket to the full forward position (see figure 78-30-21).

(3) Select test set switch to DEPLOY. Slowly increase air pressure until the pneumatic actuator drive motor begins to operate.

(4) Move the feedback slider bracket aft or forward until the directional control valve on the pneumatic actuator assembly is in the neutral position (valve exhaust port closed). Secure or hold the feedback slider bracket to prevent the direction control valve from moving. (See figure 78-30-16.)

(5) Set Deploy switch on test set to "OFF." (Leave "PWR" and "ARM" switches "ON.")

(6) Increase or decrease air pressure as required to accomplish the following checks (see figure 78-30-18 for component identification).

NOTE: Components may be checked by applying the required air pressure. As each valve port "opens" at its respective setting, it may be covered with a shop towel or equivalent material to reduce the noise. If this is not accomplished, it is difficult to determine which valve is exhausting. By placing a finger adjacent to each valve will also aid in determining the opening point of the valve.

(7) To check pneumatic components, proceed as follows:

(a) Arm light pressure switch: (Monitor "ARM" light on thrust reverser test set.)

Airplanes 380-1 through -57
Close 2.5 psi (light on)
Open "0" psi (light out)

Airplanes 380-57 and Subs
Close 4.75 psi (light on)
Open 2 psi (light out)

(b) Checking pneumatic latches.

NOTE: During these tests, the forward latch roller must be in the stowed (manually pushed forward and held) position so that the latch plunger can travel full stroke.

(1) Turn air pressure off. Depress and release inboard and outboard latch plungers. Check for freedom of movement (no sticking or binding allowed).

(2) Manually apply pressure to the latch plunger in the forward and aft direction with the plunger in the extended position, then the retracted position. Check that the switches actuate or deactuate with fore and aft pressure on plunger during movement. Switch operation may be verified by an audible indication of two (or one click if both switches operate together) distinct clicks.

(3) Slowly apply air pressure. Latch plunger(s) should retract at 5±1 psi.

(c) Directional control bleed valve relieves 12±5 psi (see figure 78-30-18).

(d) Sensing selector valve relieves 17-19 psi - closes 12 psi (see figure 78-30-18).

(e) Bleed-off reference regulator valve relieves 19 psi (see figure 78-30-18).

(f) Bleed-Off Valve (figure 78-30-18).

Apply air pressure. Depress "BLEED-OFF VALVE" switch on thrust reverser test set. Check that valve opens (relieves air pressure) when switch is depressed and closes when switch is released.

(g) Pneumatic Lines.

Apply 20 pounds air pressure - Check for leaks. Repair as required.

(h) Poppet Shut-Off Valve (internal valve).

Decrease air pressure to "0" psi. Release feedback slider bracket and move as required to open directional control valve on pneumatic actuator unit. Increase air pressure to 5 psi. Air should exhaust from directional control valve exhaust port.

(8) Remove air pressure. Position "PWR" and "ARM" switches on thrust reverser test set to "OFF."

29. Aft Nacelle Mechanical Component Check-Out

A. Check flex drive shafts as follows:

(1) Remove the flex drive shafts (drive cables) from the pinion gear housings as follows (see figure 78-30-17):

(a) Remove retention bolt fittings from pinion gearbox.

(b) Remove lock nut (P/N NAS 1291C4M) and washer (P/N AN 960-416) from pinion flex drive shaft in pinion gear box.

(c) Remove two cap screws and washers securing the flex drive shaft end to the pinion gear box and remove shaft from gearbox.

(2) Manually rotate cables. Should it be necessary to utilize a wrench to rotate cables, remove cable from the housing and remove grease; then inspect cable for kinks, broken or distorted wire strands in the outer layer. Replace flex shaft if any one of these items are present. Lubricate with MIL-G-23827 or MIL-G-21164 grease. Re-install cable in cable housing. Both cables should require approximately the same torque to rotate. This assures symmetrical drive operation.

(3) Check pinion gear for general condition and freedom of rotation.

(4) Check alignment of pinion gear housing as follows:

MAINTENANCE PRACTICES (Continued)

A. Check flex drive shafts as follows (Cont'd)

(a) Remove aft nacelle from maintenance stand and place vertically on forward mount ring on level floor.

(b) Measure the distance (24.898 ±.020 inches) between the inside surfaces of both pinion gear box beam guides (four places).

(c) Check the parallel measurement of the four beam guides of the outboard gear box. The beam guides must be parallel within .010 inch and still be within the 24.898 ±.020 inches tolerance.

(d) If a parallel adjustment is required, remove or add shim washers (P/N NA 960PD416) to aft mounting screw on inboard gearbox. For further adjustment, a tapered shim can be used between pinion gearbox forward mounting and "T" brackets. Shim outline must be within 0.030 inch of the outline of the gearbox mounting surface.

NOTE

Tapered shims must have a chem film treatment on surface before installation. Apply Alodine 1200 or equivalent.

(e) Reinstall aft nacelle in maintenance stand.

(5) Check the Feedback Follow-Up Cable System for freedom of operation as follows:

(a) Utilizing a Chatillon (P/N DPPH-150), push-pull scale (or equivalent), manually push the feedback slider bracket and follow-up cable full forward (1 inch minimum); this force must not exceed 3.5 lbs.

(b) Measure the force required to pull the feedback slider bracket aft. This force must not exceed 3.5 lbs. If forces exceed 3.5 lbs., disconnect the follow-up cable from the motor gearbox assembly and recheck cable force to determine whether excessive friction is being caused by the cable or the directional control valve. Cable friction should not exceed 2-1/2 lbs. Irrespective of whether the cable or the directional valve is at fault, if force exceeds 3.5 lbs., the cable must be removed and the slider mechanism inspected for wear and lack of lubrication.

(6) Inspect the feedback slider mechanism as follows (see Figure 78-30-16):

(a) Disconnect follow-up cable from feedback slider bracket.

CAUTION

Do not damage cable shaft when removing self-locking nut.

(b) Remove the two set screws and slider from housing.

(c) Insect the feedback slider bracket for deformation. All flanges should be at a right angle to the slotted shoe.

NOTE

If cable end is bent, the slider will be bent.

(d) Install new slider if necessary.

(e) If excessive friction is determined to be caused by the follow-up cable, replace the cable. If excessive friction is determined to be caused by the motor drive unit, replace the motor drive unit.

30. Cascade Assembly — Inspection and Rigging

A. To inspect and rig cascade assembly, proceed as follows:

(1) Torque translating structure nuts (Figure 78-30-5) to 35-40 inch-pounds.

(2) Visually inspect the beams for general condition. The beam channels should be free of excessive galling and scoring, and the plating should be intact. Check the rack teeth and stops for general condition. Pay particular attention to the threaded stop pin. Check the pin for bending and thread condition. If the stop pin is bent, replace the rack.

(3) Remove cotter keys from clevis pins that connect blocker door operating rods to yoke base assemblies. Loosen rod end jam nuts.

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MAINTENANCE PRACTICES (Continued)

A. To inspect and rig cascade assembly, proceed as follows: (Cont'd)

(4) Remove the four (4) phillips head screws from the yokes and remove yokes from the yoke base.

NOTE

Do not loose shims (one per side).

(5) Remove the four (4) clevis pins from the door actuation rods. Disconnect the four (4) actuating rods at the outside door arms. Verify actuating rods are not bent.

(6) Manually operate the racks within the beams and check for freedom of movement.

(7) Manually push the racks until contact is made against the ring and check that the yoke base contacts the ring squarely in the horizontal plane. A small amount of misalignment may be removed by loosening the two (2) yoke base mount screws and repositioning the yoke bases. If the yoke base does not contact the ring squarely, the yoke base shall be replaced.

(8) Rig blocker door tension rods ("T" bars) (Figure 78-30-22) as follows:

NOTE

One set of "T" bars may be adjusted, then the cascade assembly may be rotated 180° to facilitate rigging the opposite set.

(a) Position the cascade assembly horizontally on a work bench and loosen "T" bar rod and end jam nuts.

(b) Disconnect "T" bars.

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(c) Position the lower door in the cascade assembly in the nested position. The door must not rock and it should nest as flat as possible in the cascade assembly. Check that outer door arms do not contact the cascade assembly. It may be necessary to grind the inside door actuator link lever assemblies as per Service Letter No. 37. It also may be necessary to remove a small amount of material from the sheet metal that is adjacent to the inner door actuator arms. Minimum rivet edge distance must be maintained.

(d) With door correctly nested, the "T" bar rod ends should be adjusted to obtain a slip fit of the "T" bar attach bolts.

(e) Rotate cascade assembly 180° and repeat steps (a) through (d) on opposite door.

(f) With the "T" bars adjusted correctly, the vertical distance between the doors should be a minimum of 22-1/2 inches (loose) measured at the leading edge of the doors and 21-1/2 inches (loose) measured at the trailing edge of the doors.

(g) Place blocker doors in the fully deployed position and check for maximum of 0.150 inch door mismatch. Repeat steps (a) through (b) if mismatch is out of tolerance.

(h) Tighten (20-25 inch pounds) "T" bar bolts and jam nuts.

(9) Check blocker door lever arms (figure 28-30-7A) as follows:

(a) Remove nut, screw, and spring washer attaching lever to lever arm (four places) and remove levers from lever arm shafts.

(b) Remove the lower blocker door by springing door and lever arm to remove arm shaft from tailpipe assembly and washer (two places).

(c) Manually position the blocker doors in the forward thrust position (doors nested).

(d) Remove upper blocker door by springing door and lever arm to remove arm shaft from tailpipe assembly and washer (two places).

(e) Remove cotter pin, nut, bushing, and spring washer from the blocker door attach point (four places).

(f) Remove four lever arms and spacers, and tag each arm as removed, for later reinstallation in their original position.

(g) Inspect all blocker door actuating arms per dimensions (figure 78-30-23). Replace any arms that do not meet the depicted specifications.

(h) Install spacer on each lever arm shaft and position lever arms in their original position on the tailpipe assembly.

(i) Install washer on each blocker door attach point (four places).

(j) Attach upper blocker door attach points into lever arms, by springing doors. Install spring washer, bushing, nut and new cotter pin on end of shaft (two places).

(k) Attach lower blocker door attach points into lever arms, by springing doors. Install spring washer, bushing, nut and new cotter pin on end of shaft (two places).

(l) Manually rotate the blocker doors to a reverse thrust position.

(m) Attach lever arm to lever in their original position with spring washer, screw and nut. Torque nut to 80-90 inch pounds, four places.

(10) Adjust blocker door actuating rods (figure 78-30-22) as follows:

NOTE: The travel of both yoke assemblies from stow to deploy is 5.590±.000 - .060 inches. The travel distance of the left and right yokes must be the same.

(a) Install the 2 yokes to the yoke bases (include shims). Install actuating rods in cascade assembly and attach actuating rods to outer doors arms.

(b) With clamp installed holding blocker doors together, utilizing the thrust reverser rigging tool, or suitable measuring device, manually position the yoke 5.590 inches from the ring as measured at the pressure pad and hold securely in position. Left and right yokes may be adjusted separately.

(c) Adjust actuator arm rod ends sufficiently to allow a slip fit of the clevis pins into the yoke assembly.

NOTE: Clevis pins are to be installed with the head toward the center of the cascade assembly at this time. Clevis pins will be reinstalled with the heads outboard during final reassembly.

(d) Remove thrust reverser rigging tool, or suitable measuring device, and door clamping device. Manually move both yoke assemblies to the stow position. Both yoke assemblies must contact the ring evenly.

(e) If the yokes do not contact the ring evenly, reduce to 5.560 inches (1/2 turn of rod end equals 0.030 inches). If yokes still do not contact ring evenly reduce to 5.530 inches. Both yoke assemblies must be adjusted to the same dimension.

MAINTENANCE PRACTICES (Continued)

(10) Adjust blocker door actuating rods (Cont'd)

(f) If with the thrust reverser rigged to 5.530 in. dimension, the yoke still will not contact the ring, repeat cascade rigging. Pay particular attention to interference between inner arm and adjacent structure.

(g) With a suitable clamp, secure the yoke assemblies to the ring. Be certain that the "T" bar rods, door actuating rods, and blocker doors, are not preloaded. Remove clamp.

(h) Remove the four yoke assembly screws and install the clevis pins (heads outbd) and safety. Reinstall the four yoke assembly screws. Check witness holes on actuating rods. Tighten rod jam nuts.

(11) Check sequence latch hammer blocker arm to beam clearance (Figure 78-30-14) as follows:

(a) Install sequence latches on the beams with mounting screws and tighten to allow enough movement to check clearances.

(b) Check clearances between hammer blocking arm and beam, this dimension must be at least 0.010 (+0.005/-0.000) in. (See Figure 78-30-14.)

(c) Reposition latch on its mounting bolts to ensure that clearance is maintained.

(d) File hammer blocker arm to obtain specified clearance if necessary. Maintain proper radius if filed.

(e) Remove sequence latches. Retain for reinstallation.

(12) Inspect cascade assembly for proper assembly, safeties, foreign material, proper rod extension in witness holes, and correct installation of bolts and pins.

(13) Lubricate beam and rack as required.

31. Initial Follow-Up Cable Rigging

A. To perform initial follow-up cable rigging, proceed as follows:

(1) Place the blocker doors in the stowed position. Clamp the yoke to the ring with suitable clamps.

(2) Measure the distance from the side of the rack stop to the side of the beam stop using rigging tool or suitable measuring device. (See Figure 78-30-20.)

(3) Check the threaded portion of the follow-up cable and confirm that the slider is so positioned that an equal number of threads exist on both sides of the slider ear. Adjust to center if unequal. (See Figure 78-30-16.) Maintain clearance of 0.001-0.002 in. between nut and slider bracket.

(4) Deleted.

(5) Apply 5 ±1 psi to the system and select DEPLOY to position the directional control valve. A small gap should now be apparent at the bottom of the slot in the directional control valve body. (See Figure 78-30-16.)

(6) Select STOW and determine that a small gap is now apparent at the top of the slot in the directional control valve body.

(7) Manually position the follow-up cable slider to the same dimension measured in step (2). (Measure from the aft side of the slider finger to the aft side of the pinion gear housing.) (See Figure 78-30-21.)

(8) While holding the slider to the dimension above and utilizing a feeler gauge, measure the gap at the top of the slot in the directional control valve. This dimension should be 0.020 to 0.050 (optimum 0.032) in. Adjust the gap by repositioning the follow-up cable outer housing if these dimensions do not exist. This is accomplished by turning the jam nuts on the housing at the inbd pinion gearbox safety jam nuts.

NOTE

Jam nuts shall always be tight when checking clearance.

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MAINTENANCE PRACTICES (Continued)

31. Initial Follow-Up Cable Rigging (Cont'd)

(9) Select DEPLOY on the reverser test set. Position slider until a 0.020 to 0.050 in. gap is obtained at the bottom of the slot. Check that a min of 0.030 in. over-travel exists between the fwd side of slider finger and fwd side of pinion gear housing.

(10) Turn off air pressure and electrical power to test set.

32. Translating Structure Installation, Alignment and Adjustment

A. Prior to installing translating structure in aft nacelle, proceed as follows:

1. Deploy the blocker doors and clamp into position. (This will retract the racks into the beams and prevent damage to the deploy stop pin stinger.)

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(2) Secure (safety wire or string) both pneumatic latches in the unlocked position.

(3) Install the outbd cable in pinion gear housing with cap screws. Torque cap screws to 40-50 in.-lbs.

(4) Leave the inbd cable disconnected from pinion gear housing and both cables disconnected at gearbox.

(5) Install the translating structure into the aft nacelle.

(6) To check alignment of translating structure to aft nacelle, proceed as follows:

(a) Place the translating structure in the aft nacelle. As the structure is moved fwd, observe the beam guide pins to ensure that they are engaging the beam guides with no interference or binding. This engagement should not displace the beam. Slight contact is acceptable. Remove or install shims as necessary if interference exists between the beam guide and pin.

(b) Move translating structure fwd and aft, check for binding, drag, or other interference. The structure should move freely with no asymmetrical drag.

NOTE: If beam stops do not make contact simultaneously, check pinion gear housing alignment per paragraph 29, step (4).

(c) As the translating structure is being moved fwd and aft in the nacelle, observe the inbd and outbd beam stops. Both sets of stops should make contact simultaneously, Max differential between stops should not exceed 0.015 in.

33. Pneumatic Latch Rigging Check and Adjustment

A. To check and/or adjust the pneumatic latches, proceed as follows:

(1) Install translating structure in nacelle (if removed).

(2) Remove blocker door clamp (if installed).

(3) Position reverser to the full stowed position.

(4) Release pneumatic latches. (See figure 78-30-13.)

(5) Check that beam stops are in contact with pinion gearboxes. If adjustment is necessary shim washers may be added or removed from under the pneumatic latch mount. Shims must always be added under the top mount and under either bottom mount to reposition to the roller.

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NOTES: • The following procedure is for the LH Thrust Reverser. The RH Thrust Reverser pneumatic latches are turned over 180° and shimming is reversed.

• The 37°(±0°30') angle is critical. This adjustment is required to provide the proper over-center condition of the latch rollers.

(6) Utilizing a 37° wedge (locally manufactured), check that angle between the beam and pneumatic latch roller arm is 37°(±0°30').

(7) Add or remove shims from the top mount or from both bottom mounts to obtain the proper angle.

(8) After the 37°(±0°30') angle has been established, verify that a 0.020 in. to 0.050 in. gap exists between the latch roller and the beam stop. (See figure 78-30-13.) To obtain this clearance, add or remove shims from the top mount and from both bottom mounts.

34. Flex Drive Shaft Alignment

A. To align the flex drive shafts, proceed as follows:

(1) Install clamping device to secure blocker doors in stowed position. (See figure 78-30-25.)

(2) Manually hold translating structure unit firmly against the stow beam stops.

(3) Manually position the drive motor so that a slip fit of the outbd flex drive shaft exists.

(4) Engage the inbd flex drive shaft into the pneumatic actuator drive unit. Now attempt to engage the flex drive shaft into the pinion gear housing. If unable to engage flex drive shaft, reposition the flex drive cable one flat at a time at the drive unit until a slip fit can be obtained at the spline of the pinion gear housing.

(5) Remove both flex drive shafts from the pneumatic actuator drive unit and re-install to verify they remain a slip fit.

(6) Install cap screws securing the flex drive shaft end(s) to the spur gearbox(es). (See figure 78-30-16.)

(7) Install cap screws (torque to 40-50 in.-lbs) and washers securing the flex drive shaft end(s) to the pinion gearbox(es). (See figure 78-30-17.)

(8) Install lock nut and washer on pinion gearbox.

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(9) Install retention bolt fitting on gear housing.

(10) Remove locking devices from the beams and blocker doors.

(11) Check complete installation for security, proper safeties and binding components.

35. Sequence Latch, Deploy Switch Actuating Pin, and Deploy Stop Installation

A. To install the sequence latch, deploy switch actuating pin and deploy stop, proceed as follows:

(1) Install the outbd deploy switch actuating pin. (See figure 78-30-15.)

CAUTION: To prevent damage to pin, it must be installed from outbd side toward inbd side on rack.

NOTE: The actuator may be ground to prevent contact with rack teeth if a washer is removed from the follow-up cable actuator during adjustment. The actuator must not be allowed to rotate 180° if washers are added. Install new actuator if required.

(2) Install the inbd follow-up cable actuator on the stinger with sufficient washers beneath the stop. To prevent stop from contacting rack teeth. Sufficient washers should then be installed between the nut and the stop to allow approximately 1 to 1-1/2 threads to protrude through the nut.

(3) Reinstall the sequence latches in proper positions. Ensure that blocking arm spring is installed. (See figure 78-30-14.)

36. Operational Check of Translating Structure Subsequent to Adjustment

A. To perform operational check of translating structure subsequent to adjustment, proceed as follows:

(1) Apply 5 (±1) psi air pressure and partially deploy the reverser; then stow thrust reverser. Check pneumatic latches for smooth operation.

(2) Verify that the gap in the directional control valve window has not changed. Readjust follow-up cable outer housing to obtain 0.020 to 0.050 in. (optimum .032 in.) clearance as required (check with air pressure ON) if a change has occurred.

(3) Deploy the thrust reverser and observe the blocker door action as they reach the full deploy position. Door to door contact should be made evenly with no bounce or relaxing noted. Add two washers under the deploy stop and repeat the deploy test if bounce or relaxing of the doors is noted. Continue to add washers until doors contact smoothly.

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(4) Subsequent to smooth blocker door operation with 5 (±1) psi air applied, repeat the deploy test utilizing increasing air pressure until 20 psi has been reached.

(5) Subsequent to smooth blocker door operation at 20 psi and with the reverser in the DEPLOY position, check for 0.010 to 0.050 in. at directional control valve window.

(6) Disconnect air supply and test set.

(7) Inspect complete assembly for proper lockwire, cotter pins, correct installation of bolts, pins, and foreign objects. Lubricate racks and beams as required.

37. Installation of Engine Aft Nacelle Fairing

A. To install aft nacelle fairing on engine, proceed as follows:

(1) Ensure that the four aft nacelle fairing positioning jacks are fully retracted. (See figure 78-30-10.) Pin should be flush with stop body (turn cw to retract).

(2) Secure the two inbd and two outbd tension straps over their respective teflon rub pads on aft engine mount ring.

(3) Check that thrust reverser is partially deployed (12 to 14 in.)

(4) Ensure that the latch rollers are retained in the unlatched position. (See figure 78-30-13.)

(5) Place thrust reverser electrical lead out of adjacent access opening.

(6) Check that turnbuckle bushings in outbd tension straps may be installed without using force.

(7) Install shims as required from measurements obtained in step 25.

CAUTION: Last 6 in. of travel before aft fairing reaches fwd frame is critical for clearance of pneumatic lines.

(8) Support the aft nacelle fairing and move fairing fwd over engine tail pipe until fwd frame of aft fairing contacts aft fairing attachment on engine with two or more persons or using a suitable dolly.

(9) Rotate aft nacelle fairing as necessary to align attachment holes in aft nacelle fairing fwd frame with attachment holes in attachment ring on engine.

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(10) Install screws through ring on engine into nut plates on aft side fwd frame of aft fairing.

(11) Install screws, washers, and nuts attaching corners of aft fairing to pylon.

(12) To align aft fairing to engine, proceed as follows:

NOTE: Disregard steps (a) and (b) if the alignment was checked and shim thickness was determined in a previous step. Remove aft fairing and install shims as determined by the two following steps if alignment was not checked in a previous step.

(a) Turn the upper jack ccw until the tailpipe centers or a max of three turns has been made to check vertical alignment. Measure the difference between the upper and lower tailpipe to reverser cascade assembly if the tailpipe fails to center. This dimension, plus 0.062 in., will equal the shim thickness that will be added to the upper jack pad on the tailpipe to obtain centering.

NOTE: The two cantered inbd jacks must be turned the same number of turns.

(b) The outbd centering jack of the two cantered inbd centering jacks may be turned to check horizontal alignment. Measure the difference between the inbd and outbd tailpipe to reverser cascade assembly if the tailpipe fails to center. This dimension, plus 0.125 in., will equal the shim thickness that will be added to the two cantered inbd jacks pads. Add 0.062 in. to the shim thickness to shim the outbd jack pad.

(c) Adjust the aft nacelle fairing positioning jacks to obtain a dimension of 0.150 in. (min) between the leading edge of the cascade support channels and the tailpipe.

(d) Check that the adjustable jacks are at least finger tight and then tighten 1/4 turn.

(e) Connect test set and apply 5 psi air pressure. Slowly translate the thrust reverser toward the stow position. Check for proper clearance between fwd end of translating beams and tailpipe. Check engine alignment if clearance is not adequate. (See chapter 71.)

(f) Reconnect the four turnbuckles to the tension straps using existing hardware with the thrust reverser in the stowed position. Tighten the turnbuckles until all slack is removed from the tension rods and there is 1/4 in. (1/8 in. either side of the centerline) total lateral movement with thumb

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pressure applied to the side of the turnbuckle barrel at the rod to turnbuckle connections. Install the turnbuckle locks.

CAUTION: Be certain when inbd and outbd straps are installed, they are in tension and not in compression due to buckling of straps at attach points.

(g) Cycle thrust reverser unit 5 complete cycles using 5 psi and visually noting (while unit is translating) that initial adjustment was not affected by connecting tension rods.

(13) Do not disconnect test sets at this time. Install aft nacelle fairing (scoop).

38. Aft Nacelle Fairing (Scoop) Installation

A. Proper installation of aft nacelle fairing (scoop) requires that in the fully stowed position, a gap of 1/16 to 1/8 in. is between the thrust reverser nacelle and scoop. Reinstall the scoop with shims at the same location when the scoop was removed, if scoop installation was checked prior to removal of the thrust reverser and found to have proper clearance. (See figure 78-30-11.) To install aft nacelle fairing scoop, proceed as follows:

(1) Deploy thrust reversers.

(2) Position nacelle aft fairing over tailpipe assembly and guide nacelle aft fairing fwd [guide longeron into nacelle aft fairing (scoop) frame hole] until fairing is aligned with four tailpipe assembly cap screw holes. Install four cap screws.

(3) Install washer and nut on longeron.

(4) Install angle with two cap screws and washers to nacelle aft fairing (scoop).

(5) Attach angle to tailpipe assembly by two bolts.

(6) Stow reverser and check alignment mark (marked before removal and check clearance between nacelle aft fairing (scoop) and nacelle aft fairing). Record clearance. The alignment and the clearance should be the same as before removal.

(7) If the clearances were found inadequate during the inspection, proceed as follows:

(a) Place a straight edge along the inbd edge of the scoop. Measure and record the distance between the inbd edge and the four attach bolt brackets located inside the sugar scoop.

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(b) Measure and record the distances between the thrust reverser nacelle aft most rib to cascade assembly ring aft surface at the four scoop attach bolt location with the thrust reverser fully stowed.

(c) Shim the six attach bolt locations until adequate clearance is provided. Shim thickness can be calculated by taking the difference between the scoop measurement and adding 0.120 in.; i.e., DSS - DCR = +0.120 in. (DSS = distance sugar scoop, DCR = distance cascade ring, Ths = thickness shims).

(d) The scoop can be installed on the thrust reverser assembly per steps (2) thru (6) when the proper shims are installed.

(8) Inspect complete assembly for security and proper safeties. Install all access panels.

(9) Remove test sets and connect air line and electrical plug. Safetywire air line fitting and electrical plug.

(10) Perform Operational Check-out of Thrust Reverser System - Engine(s) Operating.

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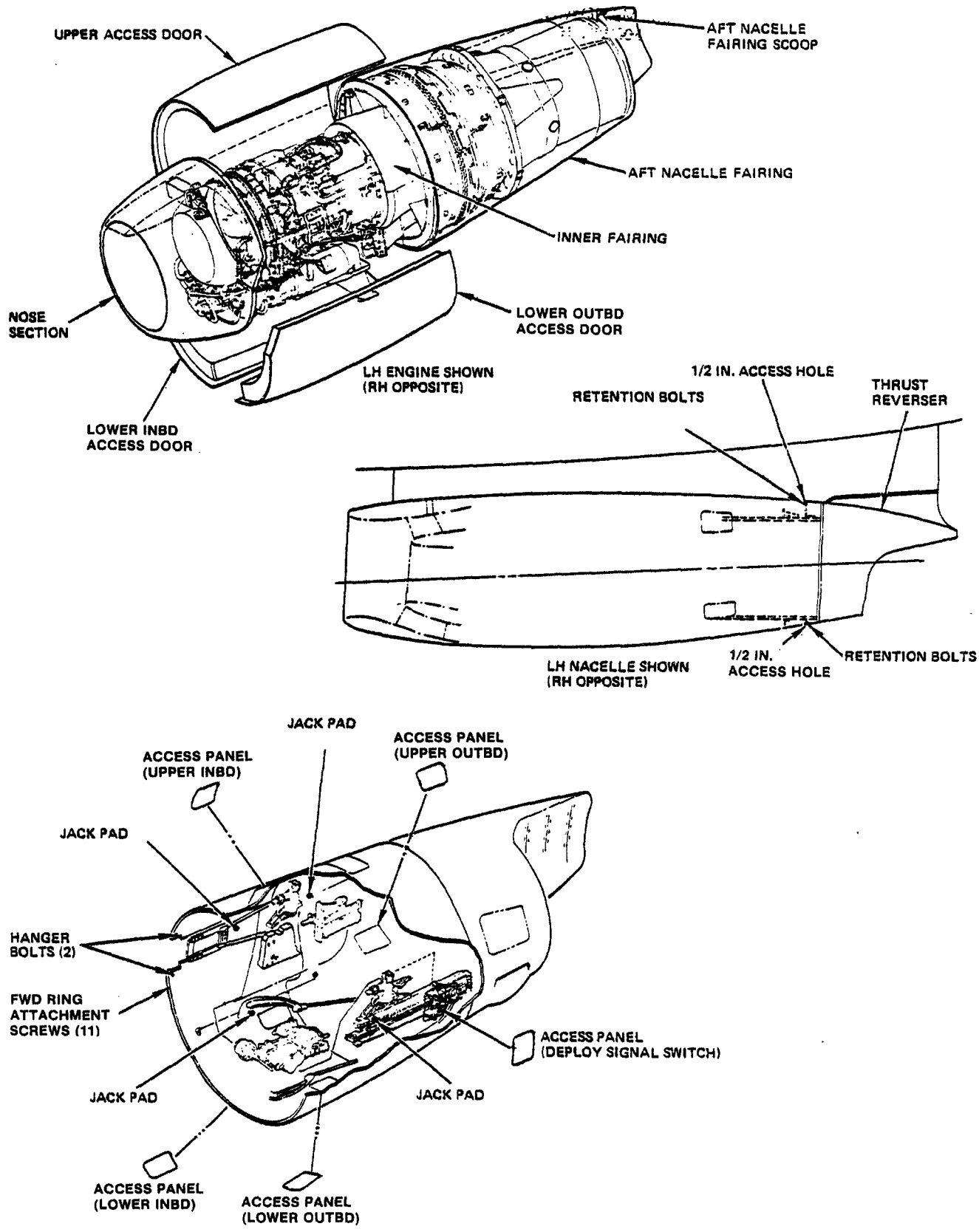

Engine Nacelle Access Doors and Fairings
Figure 78-30-9

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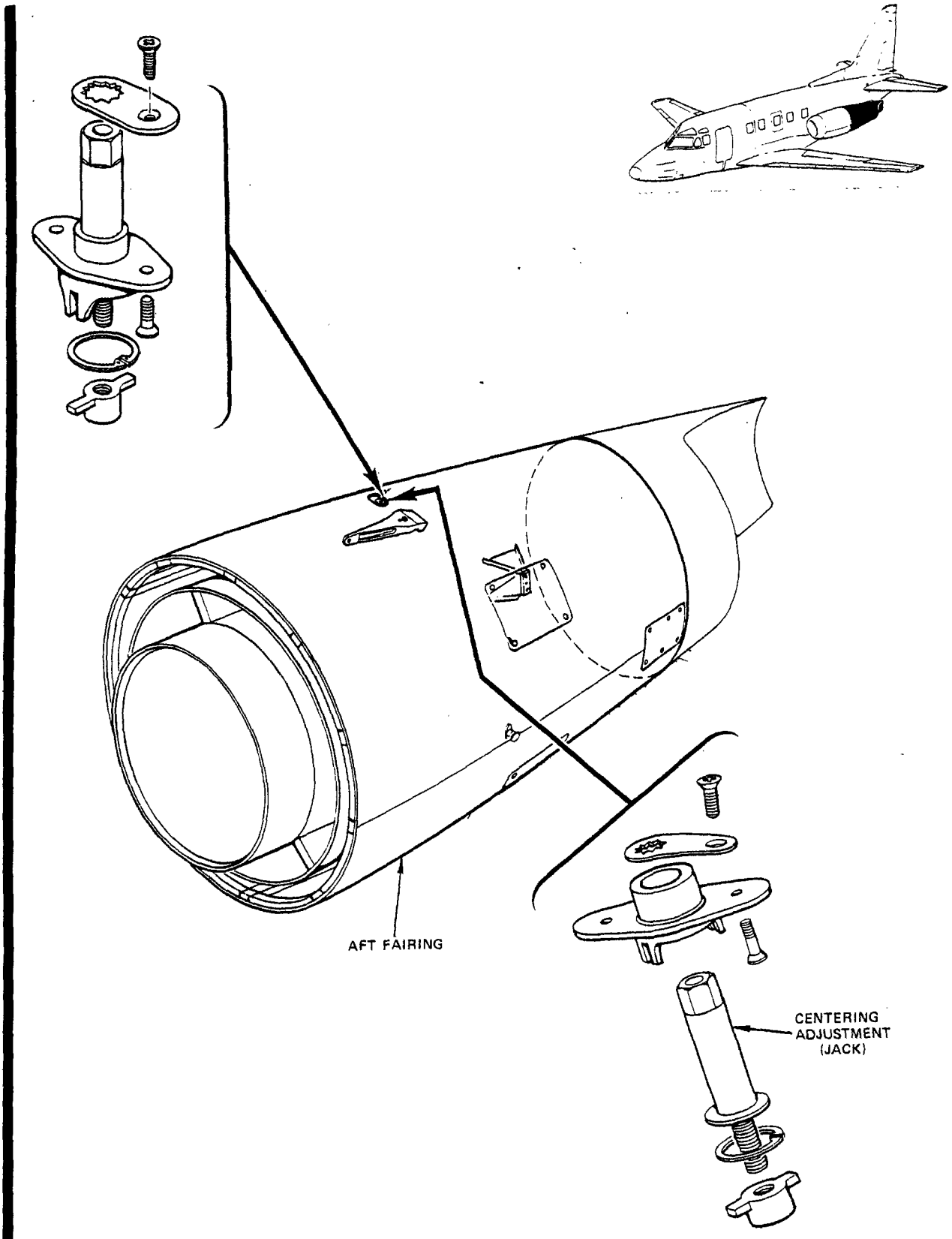
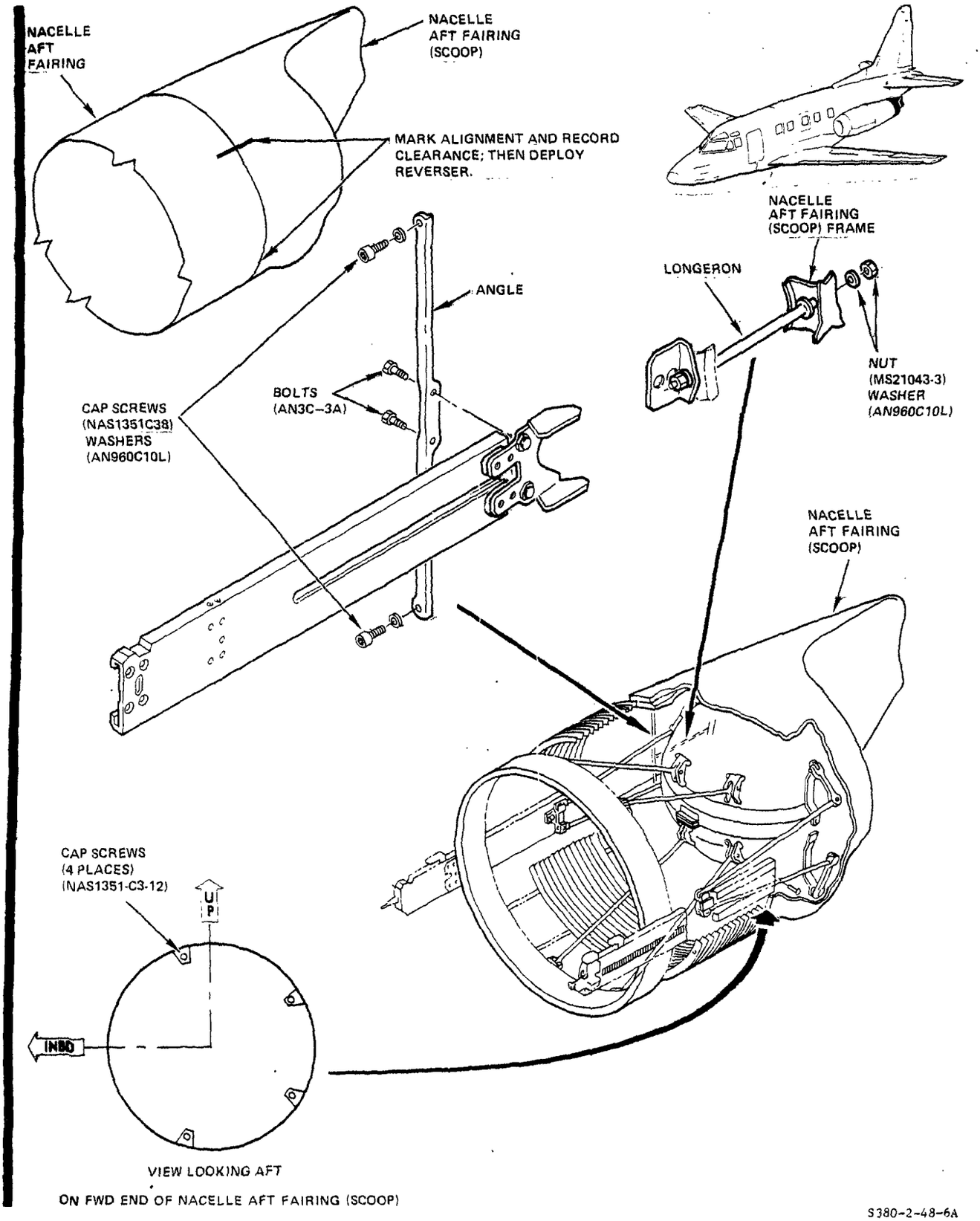
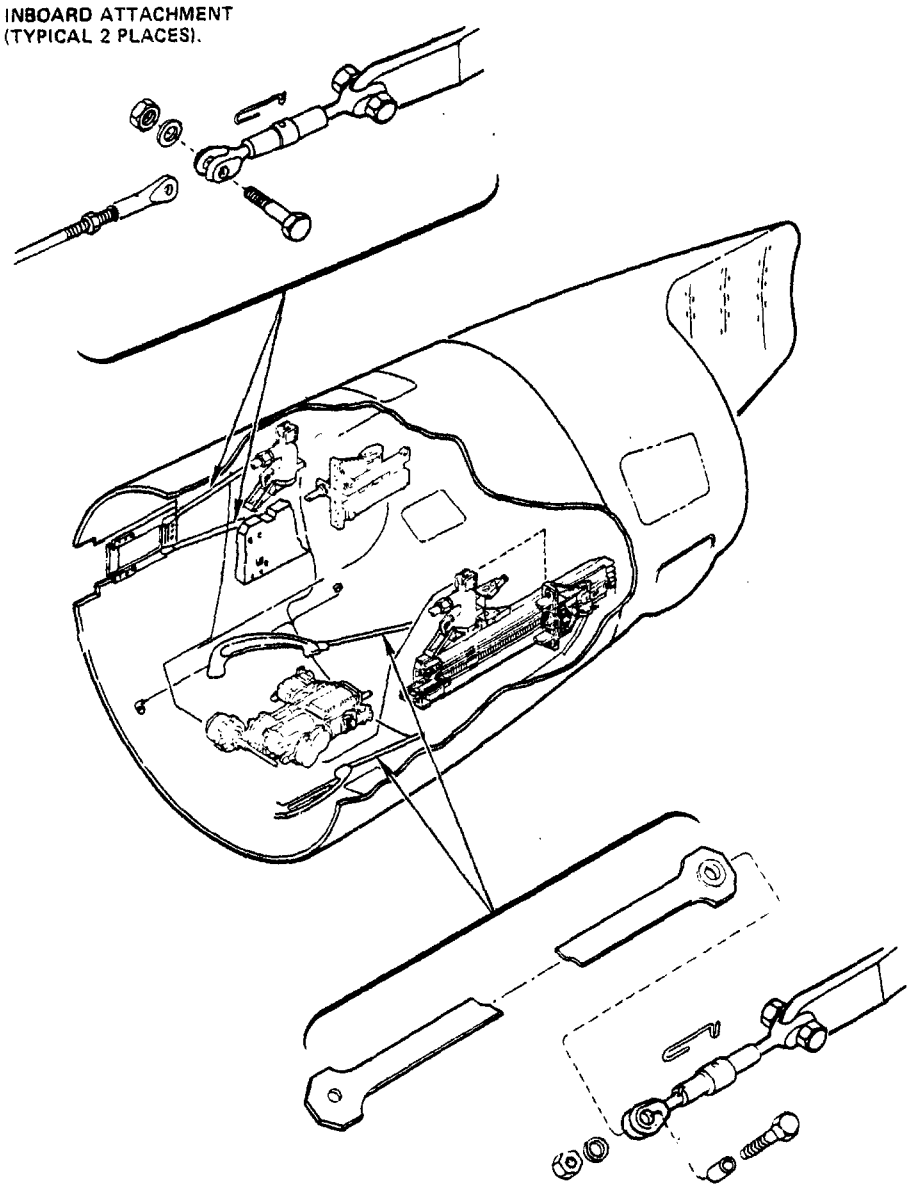


Aft Fairing
Figure 78-30-10

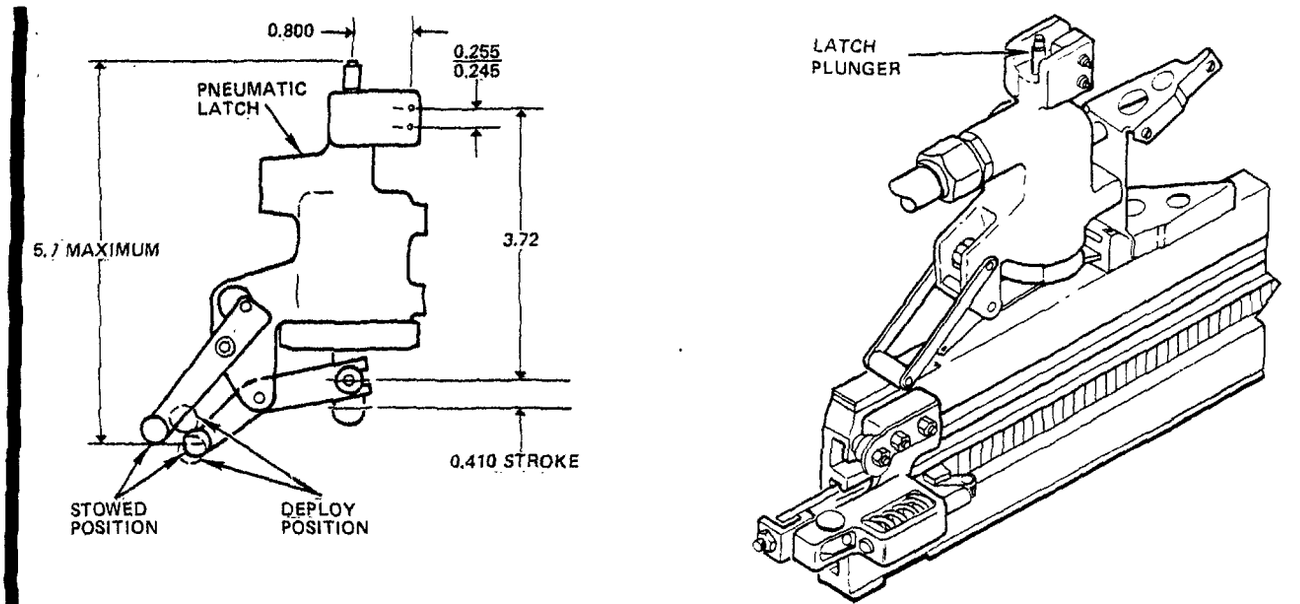

NACELLE AFT FAIRING
NACELLE AFT FAIRING (SCOOP)
MARK ALIGNMENT AND RECORD CLEARANCE; THEN DEPLOY REVERSER.
NACELLE AFT FAIRING (SCOOP) FRAME
LONGERON
ANGLE
NUT (MS21043-3) WASHER (AN960C10L)
BOLTS (AN3C-3A)
CAP SCREWS (NAS1351C38) WASHERS (AN960C10L)
NACELLE AFT FAIRING (SCOOP)
CAP SCREWS (4 PLACES) (NAS1351-C3-12)
UP
INBD
VIEW LOOKING AFT
ON FWD END OF NACELLE AFT FAIRING (SCOOP)

S380-2-48-6A

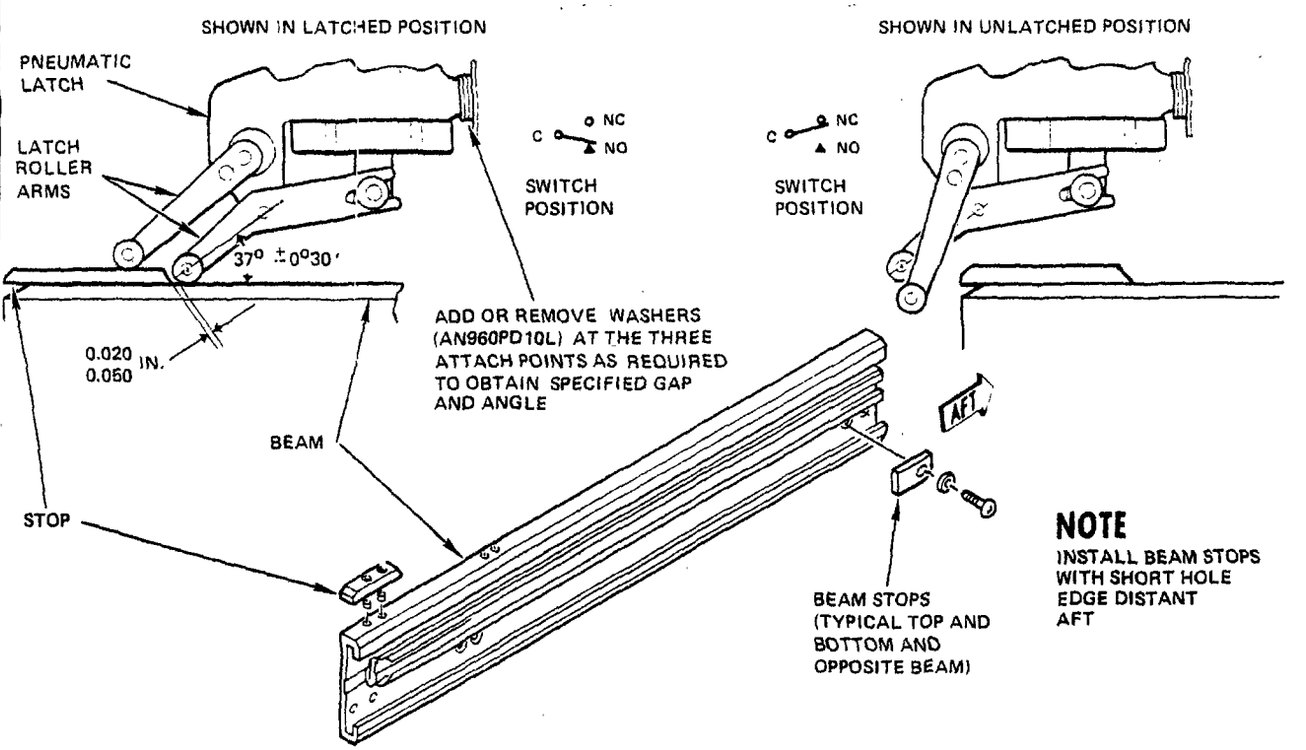


380-2-42-15

Aft Nacelle Tension Rod Assemblies
Figure 78-30-12

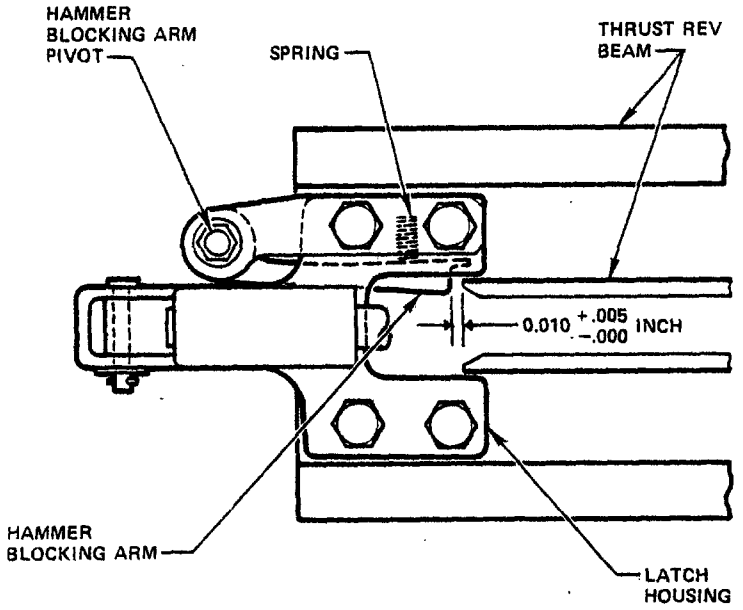


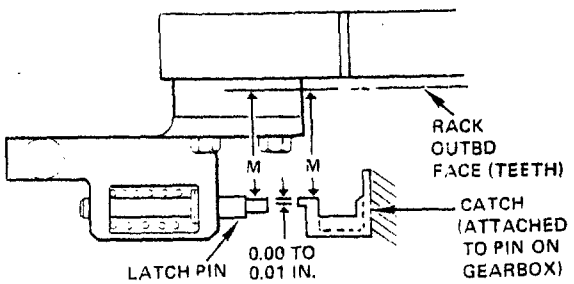
NOTE: ALL DIMENSIONS IN INCHES.

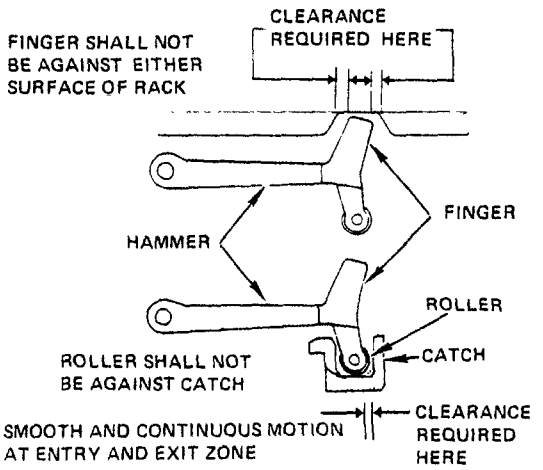

SHOWN IN LATCHED POSITION
SHOWN IN UNLATCHED POSITION
PNEUMATIC LATCH
LATCH ROLLER ARMS
37° ±0°30'
0.020 IN. 0.060
C NC NO
SWITCH POSITION
C NC NO
SWITCH POSITION
ADD OR REMOVE WASHERS (AN960PD10L) AT THE THREE ATTACH POINTS AS REQUIRED TO OBTAIN SPECIFIED GAP AND ANGLE
BEAM
STOP
AFT
BEAM STOPS (TYPICAL TOP AND BOTTOM AND OPPOSITE BEAM)
NOTE
INSTALL BEAM STOPS WITH SHORT HOLE EDGE DISTANT AFT

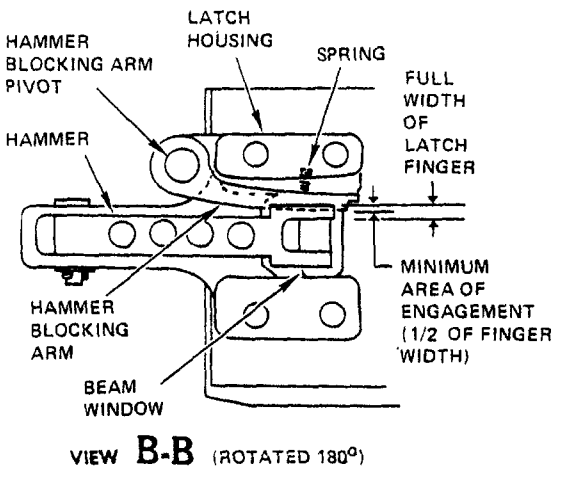
380-2-42-21

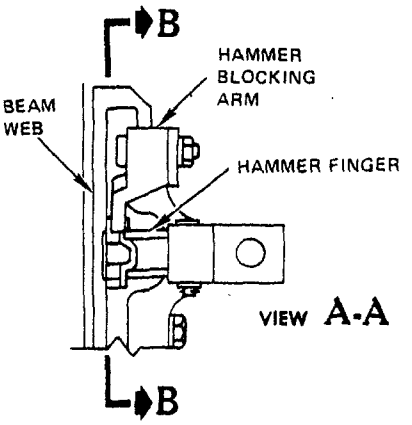
Pneumatic Latch Valve Adjustments
Figure 78-30-13





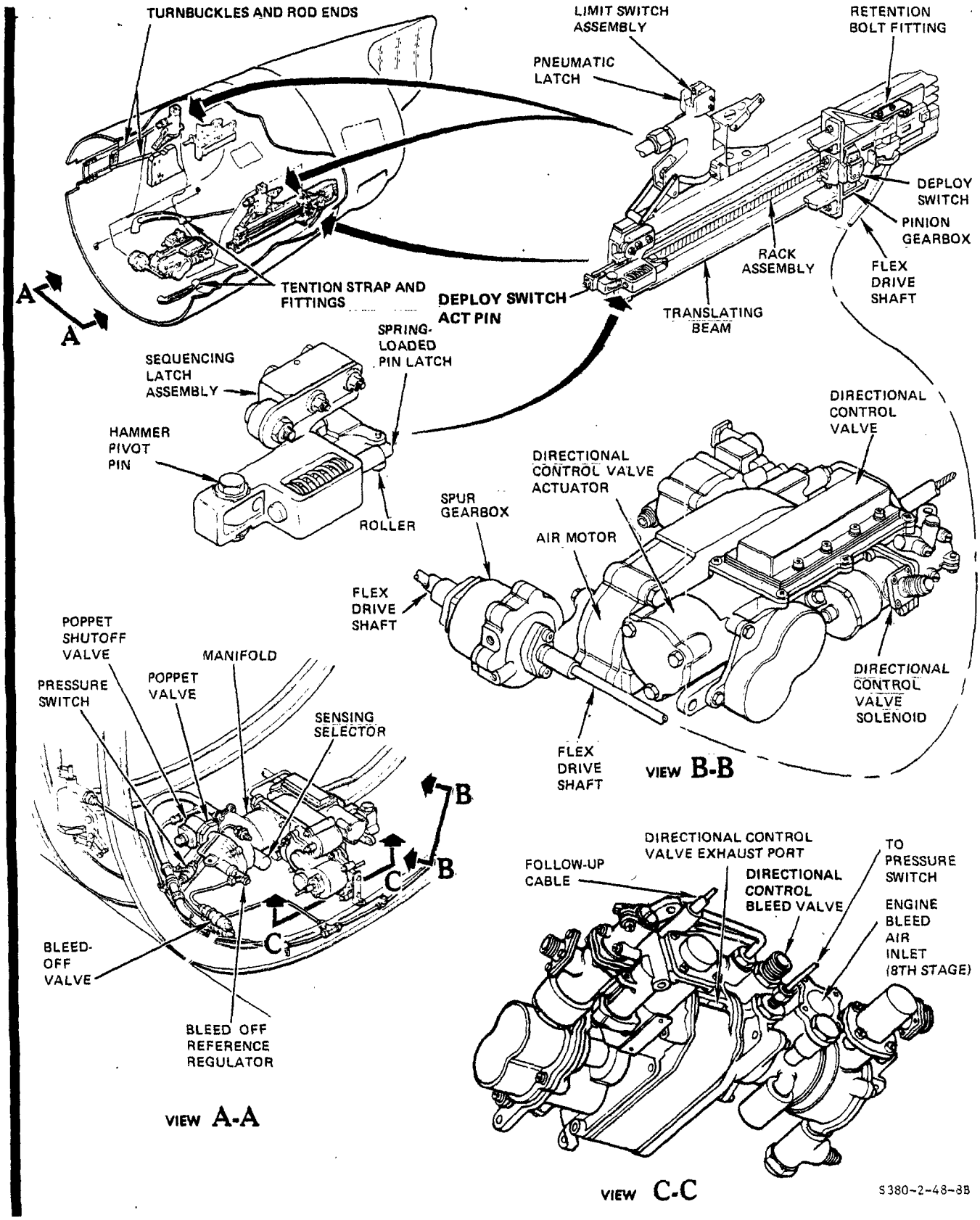




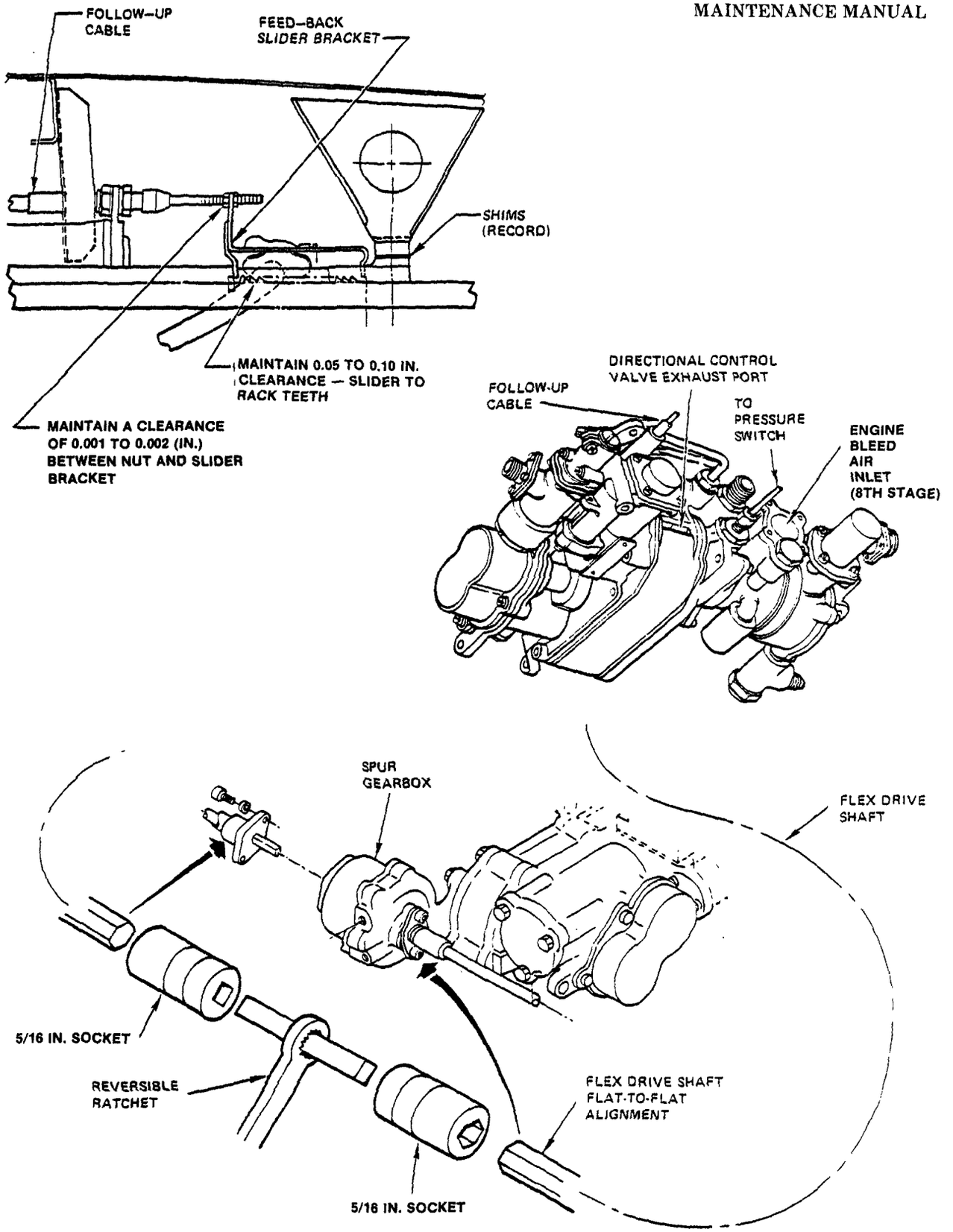


S380-2-42-17

Sequencing Latch Adjustments
Figure 78-30-14



Rack Components
Figure 78-30-15


FOLLOW-UP CABLE
FEED-BACK SLIDER BRACKET
SHIMS (RECORD)
MAINTAIN 0.05 TO 0.10 IN. CLEARANCE — SLIDER TO RACK TEETH
MAINTAIN A CLEARANCE OF 0.001 TO 0.002 (IN.) BETWEEN NUT AND SLIDER BRACKET
DIRECTIONAL CONTROL VALVE EXHAUST PORT
FOLLOW-UP CABLE
TO PRESSURE SWITCH
ENGINE BLEED AIR INLET (8TH STAGE)
SPUR GEARBOX
FLEX DRIVE SHAFT
5/16 IN. SOCKET
REVERSIBLE RATCHET
FLEX DRIVE SHAFT FLAT-TO-FLAT ALIGNMENT
5/16 IN. SOCKET

Feedback Cable and Actuator Motor Components
Figure 78-30-16

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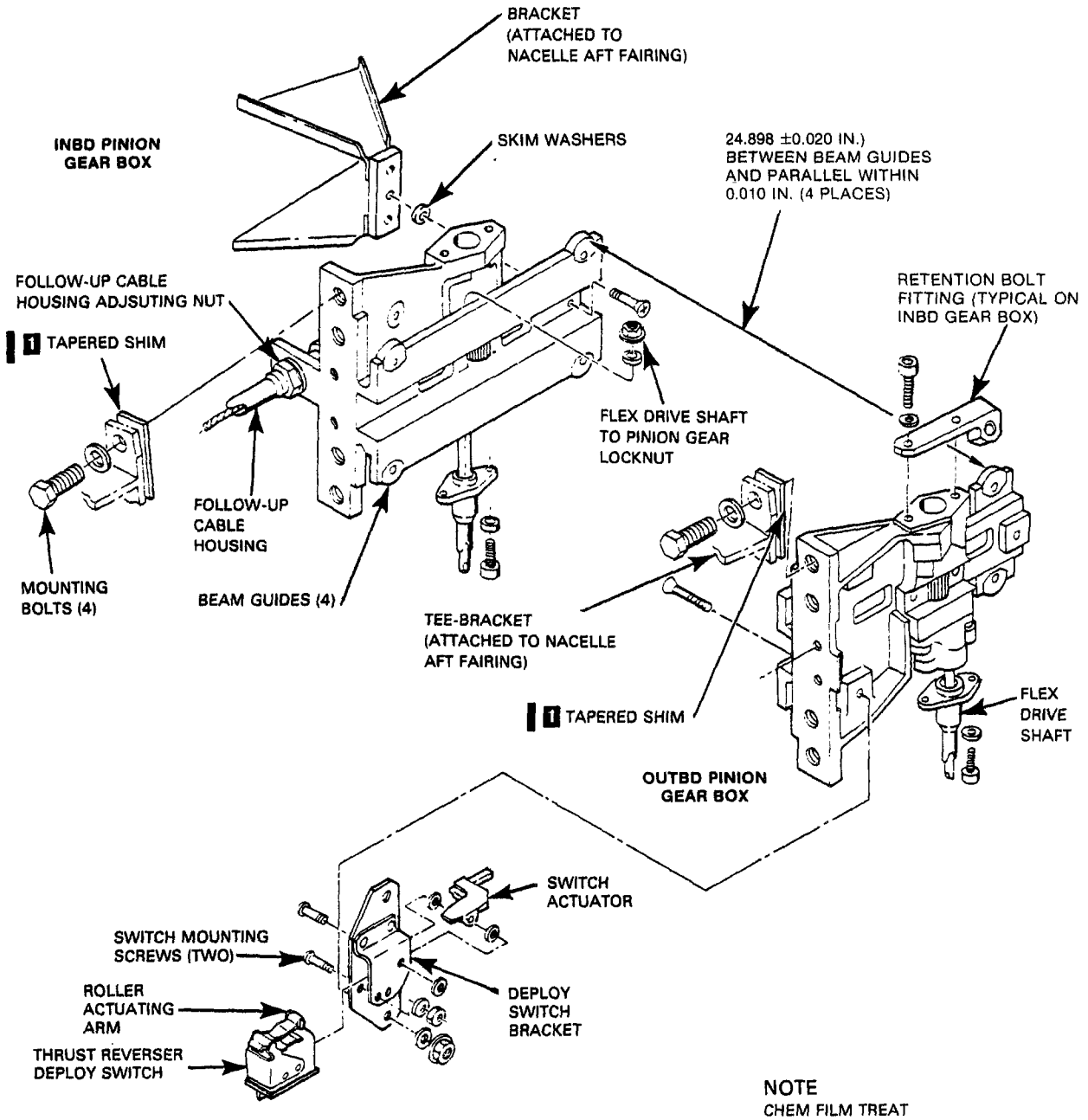

SHIM MATERIAL AND SIZE (FABRICATE TAPER AS REQUIRED

1 ALUMINUM ALLOY 2024 T3 OR T4
.010 X .060 X 1.0 X 6.0 IN.

Pinion Gearbox Assemblies
Figure 78-30-17

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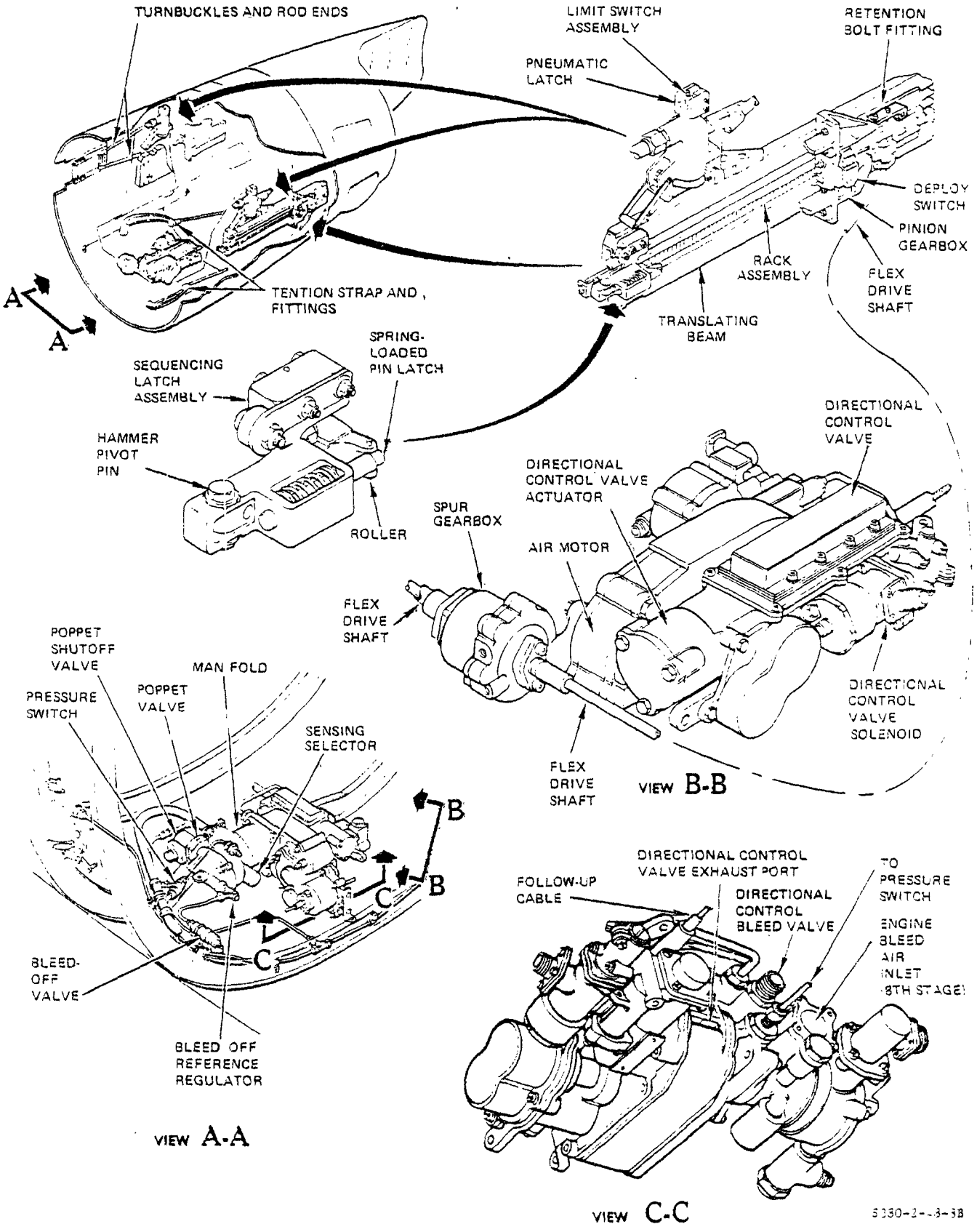

Figure 78-30-18 — Pneumatic Actuator Assembly and Beam and Rack Major Component Location

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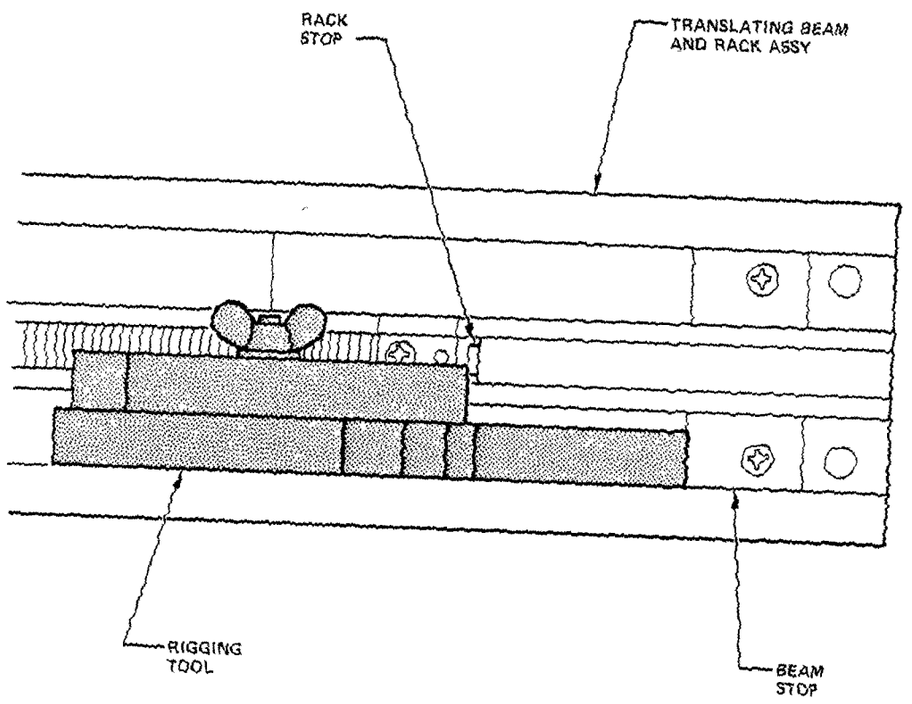
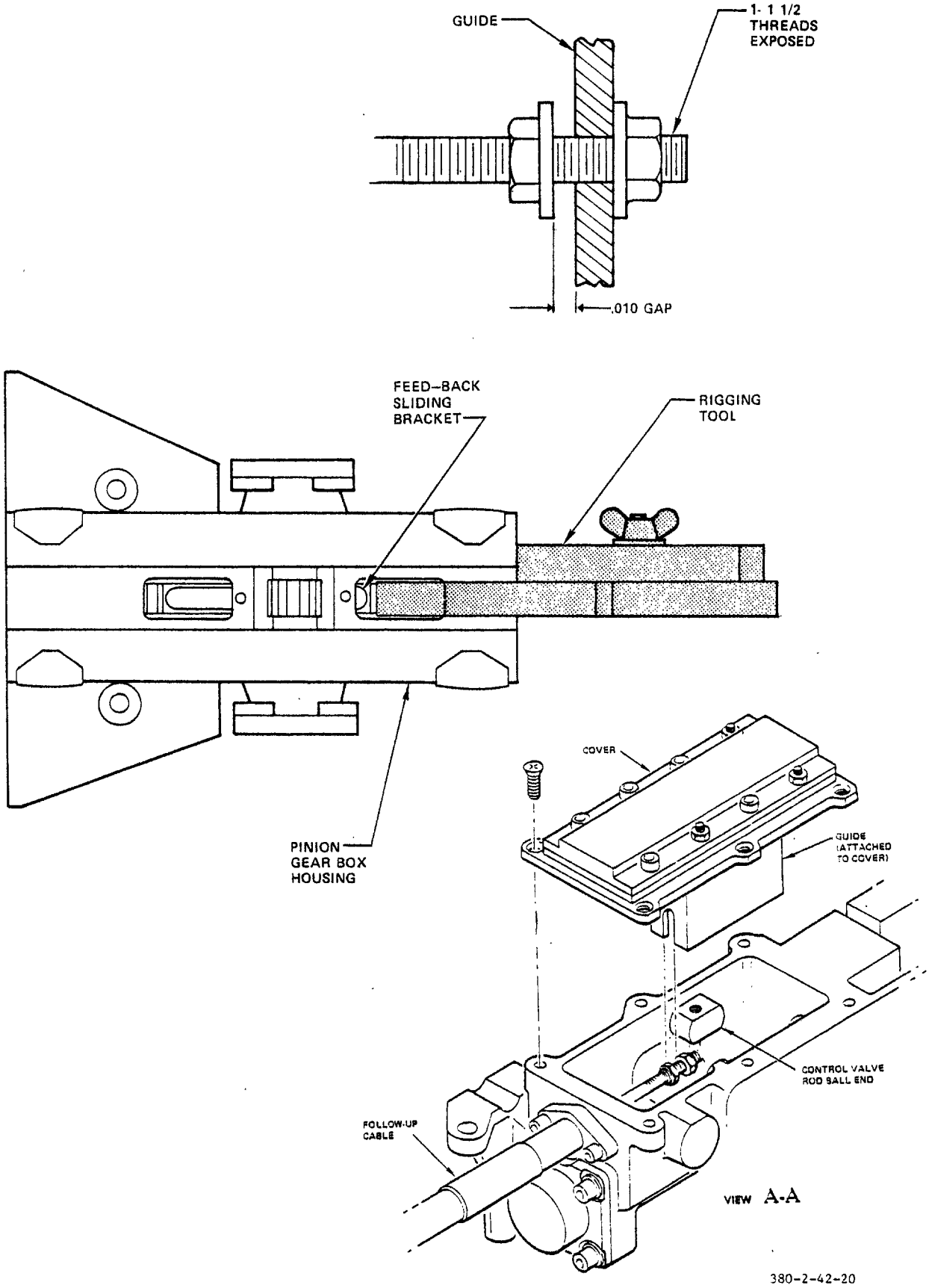
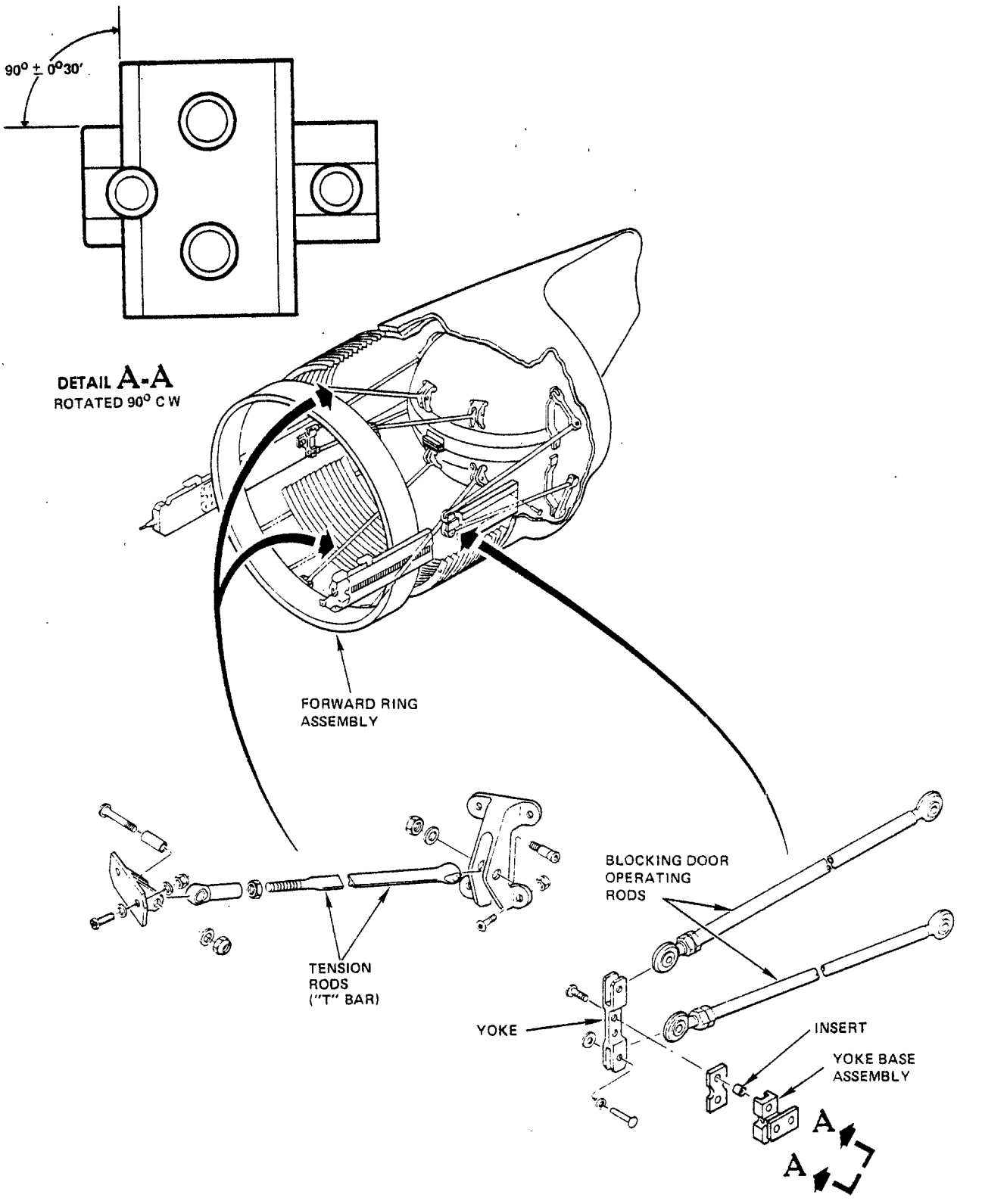

RACK STOP
TRANSLATING BEAM AND RACK ASSY
RIGGING TOOL
BEAM STOP

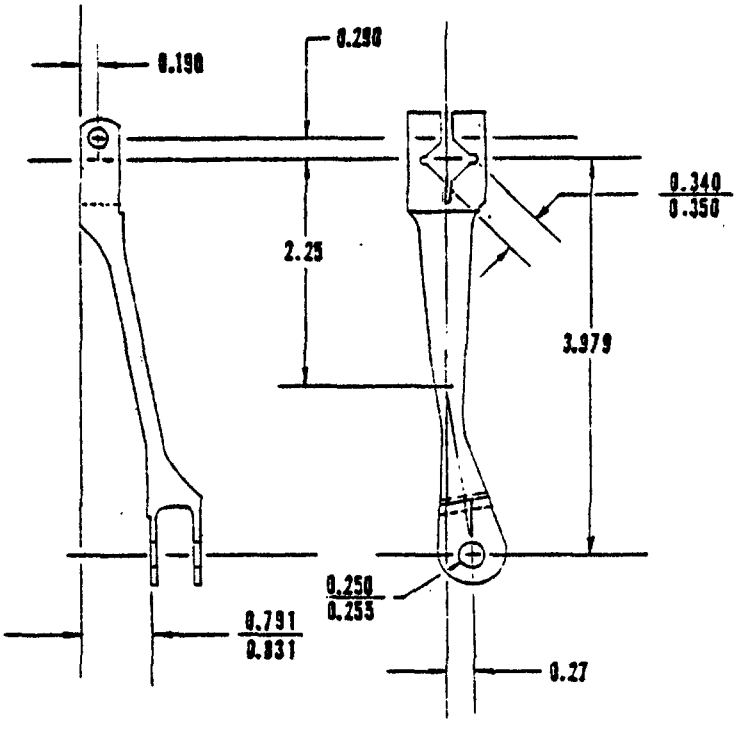


Initial Follow-Up Cable Adjustments
Figure 78-30-21

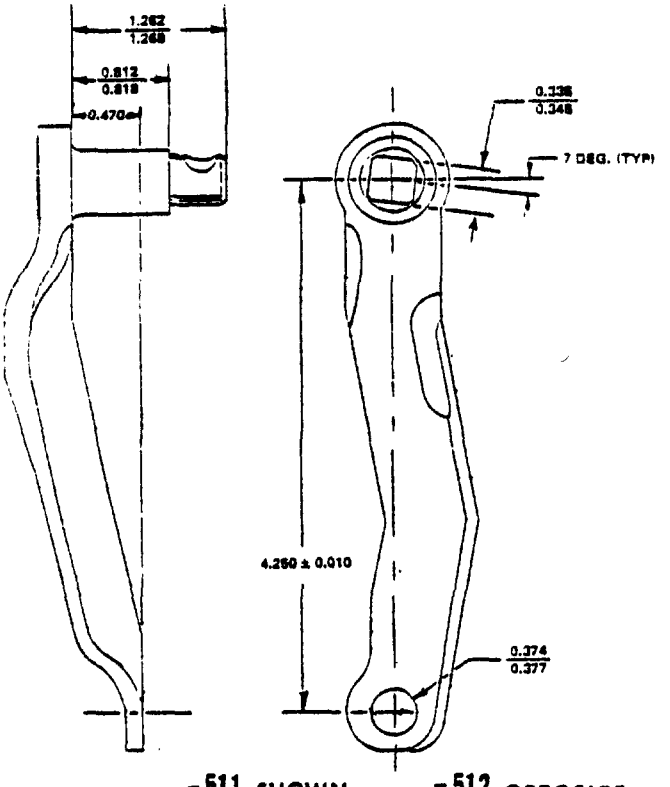


330-2-42-22

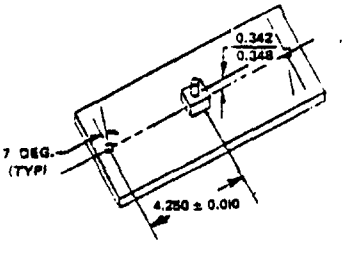
Actuating Rod Adjustment With Rigging Tool
Figure 78-30-22



-1 SHOWN -2 OPPOSITE

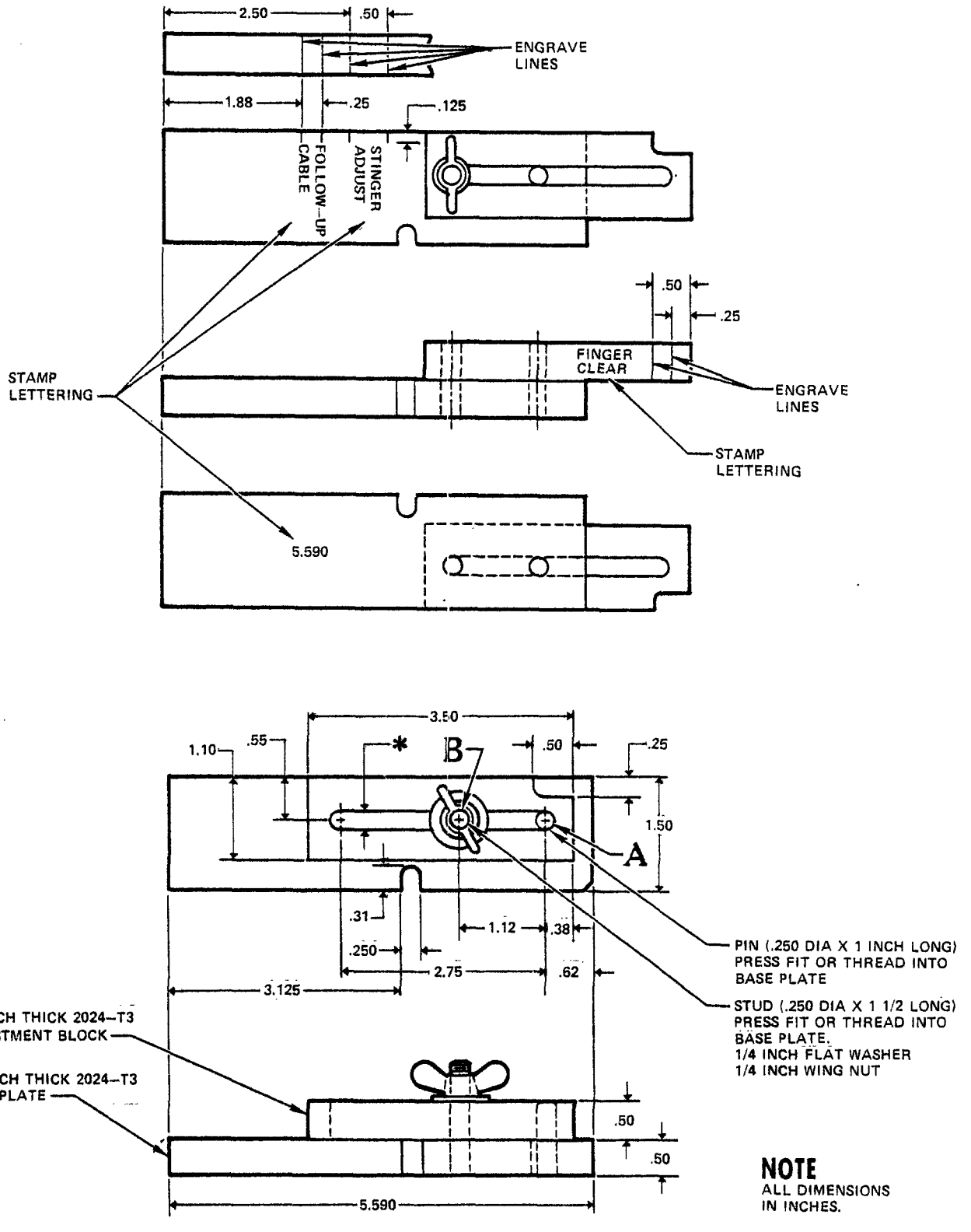


-511 SHOWN -512 OPPOSITE



CHECK FIXTURE

Thrust Reverser Blocker Door Arms
Figure 78-30-23

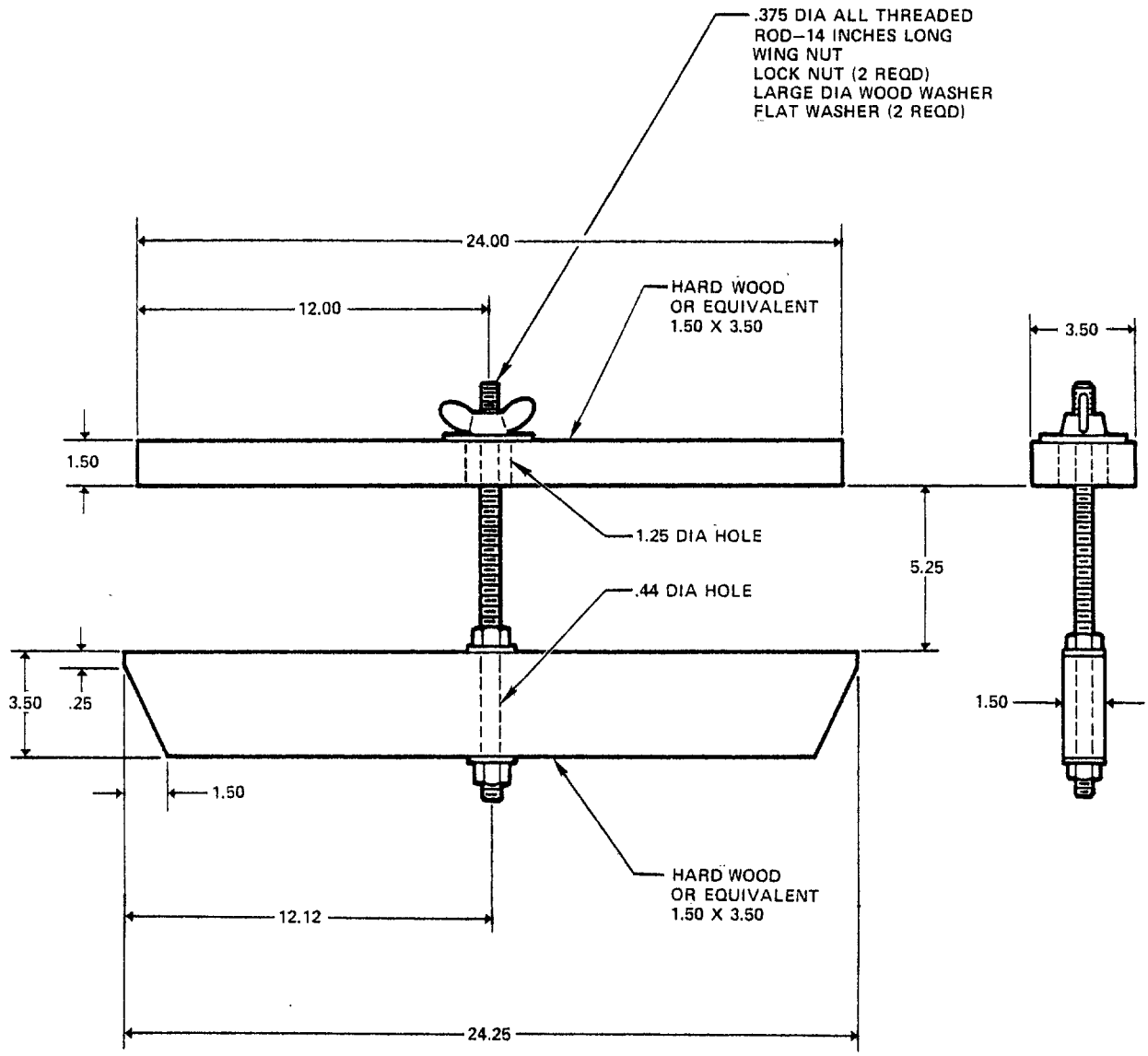


NOTE
ALL DIMENSIONS IN INCHES.

* SLOT TO SLIP FIT OVER A AND B

S380-2-42-23

Local Manufactured Rigging Tool
Figure 78-30-24



NOTE
ALL DIMENSIONS
ARE IN INCHES

S380-2-42-24

Local Manufactured Rigging Tool
Figure 78-30-25

Thrust Reverser Inspection Procedures to Increase Time Between Overhaul Hours.

A. The following procedures when complied with, will extend the time between overhaul (TBO) hours from 2000 hours to 4000 hours. It is mandatory that all issued Aeronca Service Bulletins be incorporated at the time of this inspection. It is also recommended that all Aeronca issued Service changes be incorporated at the time of this inspection. Refer to table 1 and 2 following these procedures for the service publication list. To accomplish these procedures, proceed as follows:

(1) Remove engine aft fairing from pylon. (Refer to "Removing Engine Aft Fairing," Chapter 71) and position in suitable stand.

(2) Remove nacelle (engine) aft fairing (scoop). Refer to "Removal and Installation of Nacelle Aft Fairing (Scoop)."

(3) Remove translating structure from nacelle (engine) aft fairing. Refer to "Replacement of Translating Structure."

(4) Prior to cleaning the aft nacelle structure, all electrical equipment, the actuator components, flexible drive shafts, pinion gear boxes, bleed valves, three way valves and pneumatic latches must either be protected from the cleaning materials, or removed.

WARNING

THE APPROVED CLEANING MATERIALS ARE BOTH TOXIC AND FLAMMABLE AND CLEANING MUST BE PERFORMED IN A WELL VENTILATED AREA. AVOID UNNECESSARY OR PROLONGED CONTACT OF THE CLEANING SOLUTIONS WITH THE SKIN AND KEEP CLEANING SOLUTIONS AWAY FROM ALL SOURCES OF IGNITION. FAILURE TO OBSERVE THESE PRECAUTIONS COULD RESULT IN PERSONAL INJURY OR SERIOUS ACCIDENT.

B. To clean interior of aft nacelle, proceed as follows:

(1) The interior of the aft nacelle may be cleaned using a pressure type spray and P-D-680 cleaning solvent or by wiping the surface with a clean cloth dampened with P-D-680 cleaning solvent. Before the solvent has evaporated, dry the surfaces with clean dry cloths.

NOTE

To avoid contamination of the cleaning solvent, always pour the solvent from the container onto the cloth. The cleaning cloth should never be dipped into the solvent.

(2) Loosen caked deposits of dirt, grease, etc., with a soft non-metallic brush saturated with cleaning solvent then wipe dry with a clean dry cloth.

(3) Clean the unpainted electrical grounding surfaces, connector pins, terminals and electrical contacts with ethyl alcohol O-E-00760C and coarse paper toweling.

(4) Clean the exterior surfaces of the components removed or protected. (Remove protective covering from components left installed in nacelle.) Wipe surfaces with a clean cloth moistened with either cleaning solvent P-D-680 or methyl ethyl ketone TT-M-261D, then wipe dry with clean dry cloths.

(5) Clean the translating structure and exterior surfaces of the aft nacelle by applying cleaning solvent (P-D-680) to surfaces using a non-metallic brush, sponge or clean cloth. Allow the solvent to remain on the area for a short period of time (10-15 minutes) then agitate the solvent using a brush, sponge or cloth. Wipe the surfaces thoroughly with clean dry cloths. All waste solvent, wiping rags, etc., should be collected and disposed of when cleaning operation has been completed. Inspect the cleaned areas to determine that there is no trapped solvent within the structure.

C. (1) To inspect the aft nacelle fairing components, proceed as follows:

(a) Titanium inner skin and all visible internal frames for warping, buckling, and distortion.

(b) Inspect complete aft nacelle fairing for nicks and dents and for loose or missing fasteners.

(c) Damaged or missing insulators.

(d) Cracks in #2 frame in area of 232-20265 longeron attachment (see figure 78-30-30).

(e) Damaged or worn 232-20269 blisters.

(2) Inspect the engine exhaust nozzle assembly as indicated in Chapter 78 of General Electric CF700 Turbofan Maintenance Manual (SEI-187).

(3) Inspect the thrust reverser exhaust nozzle, (attached to the translating structure) for the following defects:

(a) Warped, buckled or distorted skins.

(b) Cracks in skin.

(c) Loose or missing fasteners and/or separated spot welds.

(d) Security of attachment to translating structure.

(4) Inspect the cascade assembly (attached to translating structure) for the following defects:

(a) Inspect the cascade vanes and tension straps for cracks.

(b) Check tension straps for proper torque. Refer to "Torque - Translating Structure".

(c) Inspect for loose fasteners in blocker panels and that all attaching hardware is secure.

(5) Inspect the aft nacelle centering jacks for proper operation.

D. To repair the aft nacelle structure and fairing, proceed as follows:

NOTE

The following repairs are applicable to areas of the aft nacelle structure and the aft nacelle fairing structure. Repairs may also be accomplished to the cascade assembly, blocker panels, and to the thrust reverser exhaust nozzle if disassembly is not necessary. Repairs to the cascade assembly, exhaust nozzle, and blocker panels which require disassembly or separation from the structure are prohibited. If disassembly or separation is necessary, the cascade/exhaust nozzle/blocker panel assembly must be returned to the manufacturer for disassembly, repair and reassembly.

(1) Repair Materials (As Required)

The following is a list of the materials required for repair of the aft nacelle and thrust reverser components:

Material	Spec/Mfg.
Commercially Pure Titanium 0.020 Thick	MIL-T-9046 Type I, Composition B
2024-0 Aluminum Alloy 0.025 Thick	QQ-A-250/4 Heat Treat to T42
Asbestos Sheet 0.032 Thick	Rabestos Manhattan RM NOA56
Adhesive DC93-076	Dow-Corning Co.
Welding Wire W-17-4PH	AMS 5825
Welding Wire	AMS 5837A

(2) Nacelle Aft Fairing Repairs

(a) Warping, buckling and distortion to the titanium inner skin will

require that the skin be replaced with a new skin, locally fabricated from 0.020 titanium. If the frames securing the inner skin in place are damaged, new frames must be locally fabricated from the same type and gage of material as the damaged frame and then installed prior to the installation of the new inner skin. Refer to figure 78-30-30 for fastener information.

(b) Replace all loose or missing fasteners. Refer to figure 78-30-30 for the size and type of fastener required for replacement.

(c) Replace all damaged insulators using 0.032 thick asbestos sheet cut to size then cemented in place with DC-93-076 adhesive. (Refer to figure 78-30-30).

(d) If cracks are detected, repair cracks on the inboard side of the second frame of the aft nacelle fairing where the tubular intercostal connects to the frame, repair by stop drilling each crack and then installing a repair doubler fabricated of 0.025 2024-T42 aluminum on the aft side of the frame extending at least 3/4 inch beyond stop drilled holes. Secure repair doubler to frame using MS20470-AD4 rivets. (Refer to figure 78-30-30).

(e) Replace damaged or worn blisters with new undrilled blisters. Secure in place using size and type of fasteners as shown on figure 78-30-30.

(3) Thrust Reverser Exhaust Nozzle Repairs

(a) Warped, buckled or distorted skins will require replacement unless the damage can be cold worked back to the original contour without disassembly from the cascade/exhaust nozzle/blocker panel assembly

(b) Cracks in the exhaust nozzle of the thrust reverser may be repaired if the repair can be accomplished without disassembly from the cascade/exhaust nozzle/blocker panel assembly. Stop drill each end of the crack, then Heli-arc weld the crack using W-17-4-PH welding wire.

(c) Replace loose or missing fasteners in the exhaust nozzle assembly with the same size and type of fastener as originally installed.

(4) Cascade Assembly Repairs

(a) Broken or cracked cascade vanes may be repaired by Heli-arc welding using AMS5837A filler rod; providing access is available and disassembly is not required.

(b) Broken or cracked cascade tension straps require a complete teardown of the assembly to effect a repair. The complete assembly should be returned to the manufacturer for dissembly and repair.

(5) Aft Nacelle Centering Jacks

Replace all aft nacelle centering jacks that are bound up or are difficult to adjust with new centering jacks that operate properly.

(6) Replacing Torque Box Bearings

The torque box bearings are removed from the torque box fitting using a locally fabricated bearing puller and receiver combination. The torque box bearings are pressed into the bearing housing with a medium press fit. The bearing housing in the torque box fitting must be supported to prevent damage when pressing in a new bearing. The new bearing must be installed within ±0.010 in. of bearing housing outer surface.

(7) Inspect all pneumatic tubing for kinks, dents, corrosion, and security of attachment. Replace and/or repair as required.

(8) Inspect electrical harness for damage, security, evidence of overheating, cracked insulation, condition of switch potting compound, and proper wire routing. Replace and/or repair as required.

(9) Inspect pneumatic latches for security, cracked mounting brackets, and latch and rollers for freedom of rotation and/or wear. Manually actuate latch plunger and latch arms to the latched and unlatched position to ensure that operation is smooth without binding. Check latch dimensions. (See figure 78-30-31.) Replace latch if any dimensions exceeds depicted limits.

(10) Remove pinion gearboxes. (Refer to Replacement of Pinion Gearbox.) Inspect as follows:

NOTE: Remove only one gear box at a time; then perform the following inpsection to prevent intermixing of parts and shims. Reinstall gear box and remove opposite gear box and perform the same inspection. Tapered shims may be found on the mounting brackets during removal of gear boxes. Note the position of the shim taper; then tag or attach shim to its respective mounting position. This will aid during the gear box installation procedures and will prevent a major rigging error.

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(a) Inspect pinion gearbox bearings (4 each per gearbox) for looseness on airplanes before S/B 41. Remove and replace bearing attaching rivet if bearing is loose. (See figure 78-30-32.)

(b) Check the inbd faces of the four bearings are in a common plane within 0.003 in. This may be accomplished by placing the pinion gearbox on a surface plate (inbd bearing faces down) and using a thickness gage to determine the plane. Replace bearings if the stated tolerance is out of limits. Replace gearbox if still out of tolerance with new bearings.

(c) Check pinion gear teeth for excessive wear, chipped teeth, and general condition. Replace gearbox(es) if any one of these defects are found. Inspect deploy switch, switch brackets, and actuating arm (outbd gearbox) for cracks, security, and general condition.

(d) Check gearbox(es) for cracks and corrsion. The applicable item may be replaced if any cracks are found on a bolt-on item.

(e) Inspect catch on fwd end of gearbox for physical damage and cracks. (See figure 78-30-32.) Check latch pin engagement hole for elongation (due to misaligned pin). Replace catch(es) as required.

(f) Reinstall pinion gearbox(es). (Refer to Replacement of Pinion Gear Box.)

(11) Inspect feedback cable as follows:

NOTE: Label screws to ensure that they are installed in their identical positions.

(a) Remove the five screws securing cover to directional control valve. (See figure 78-30-36.)

(b) Remove directional control valve cover. Use care when removing cover to prevent damage to gasket and entry of foreign objects.

(c) Hold the feedback cable at one end while alternately pushing and pulling against the cable at the opposite end. Replace the feedback cable assembly if movement is detected.

(d) Install the directional control valve cover. Ensure that engagement of the follow-up cable and control valve rod ball ends exists during cover installation. Install the five screws in their original location.

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(12) Inspect flex drive shafts as follows:

(a) Inspect the outer housings and attach flanges for damage, worn spots, and proper routing. Replace the housing if more than three adjacent woven strands are broken or any attaching flange is cracked.

(b) Remove the inner flex shaft from outer housings. Do not clean or remove grease. Inspect for kinks, broken or distorted wire strands in the outer layer. Check swagged end for necking and distortion. Replace flex shaft if any one of these items are present. Lubricate with MIL-G-23827 or MIL-G-21164 if required.

(c) Reinstall inner flex shaft in outer housings

(13) Inspect gear racks as follows: See figure 78-30-33 for corresponding steps to inspection areas.

(a) Inspect leading edges of ramps on both gear racks for burrs and chipping of the spray coating. Replace if damaged.

(b) Inspect rack teeth for heavy wear patterns, chipping, and excessive wear. Replace rack if any one of these conditions exists.

(c) Inspect inserts for damage and security. Replace as required.

(d) Check pin for cracks, weld at base, and distortion or outer surface damage. If pin requires replacement, proceed as follows:

(1) Cut off old pin and drill out to 0.60-0.65 in. depth. Insert new pin to a depth of 0.50-0.55 in. (total pin length of 1.15-1.18 in.). Weld around base of pin. Clean weld to min radius. Material composition is 17-4 PH CRES per AMS5643.

(e) Check pins on end of racks for wear and looseness. Replace if required.

(14) Inspect translating beams as follows: See figure 78-30-34 for corresponding steps to inspection areas.

NOTE: Remove only one beam at a time; then perform the following inspection to prevent intermixing of parts and shims. Reinstall beam and remove opposite beam and perform the same inspection.

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(a) Remove the aft four screws and stops from beam.

(b) Disconect blocker door operating rods and yoke from yoke base.

(c) Move aft end of beam slightly outbd and remove rack and yoke base from beam; then remove the five front beam screws and nuts and remove beam from translating structure.

(d) Inspect beam surfaces for breaks in the hard-coat (surface) areas. (See figure 78-30-34.) Breaks in hard-coat areas will be visible by galling. Replace beam if galling is present.

(e) Check beam flatness by placing the inbd face on a surface table (or using a straightedge) and attempting to rock the beam (or straightedge). Beam flatness must be within 0.005 inch. Replace beam if limits are exceeded.

(f) Check beam area where the yoke base travels. A total width of this area must not exceed 0.375 (±0.010) in. Replace beam if limits are exceeded.

(g) Inspect dimensions of beams. (See figure 78-30-34, views A-A and C-C). Surfaces in the vertical plane and surfaces within the horizontal plane must be parallel to other bearing surfaces in the same plane within 0.001 in. Replace beams if dimensions are out of limits.

(h) Check beam ramps, guides, and stops for wear and security. Replace any loose or damaged inserts. Ramp hardware may be secured by application of nylock VC-3 or loctite to threads.

(i) Accomplish Aeronca S/B S75-78-30-011 on guide pin attaching hardware (if not previously complied with).

(j) Accomplish a penetrant inspection on yoke and yoke base. Replace if cracked. Check yoke base inserts for security, damage and alignment. (See figure 78-30-34.) Yoke base alignment deviations (see figure 78-30-34, view D-D) of 5° or less may be mechanically corrected. Replace if deviation is greater than five°.

(k) Remove blocker door operating rods from blocker door arms and perform a penetrant inspection of rods. Inspect rod end bearings for freedom of operation and wear. Check that rods are not bent (0.100 in. max.) over full length of rod(s). Replace rods as required. Temporarily install rods in their original location.

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NOTE: Do not remove rod end bearings from rods for the penetrant inspection and ensure that each rod is installed in its identical location.

(l) Remove one blocker door tension rod and perform penetrant inspection of rod end, rod, and attaching brackets. Temporarily replace tension rod. Repeat procedure on the remaining rods. Replace rods or attaching brackets as required if cracked. (See figure 78-30-34).

NOTE: Do not remove end fittings of tension rods or attaching brackets from blocker doors and fwd ring for the pentrant inspection.

(m) Install rack, yoke, and yoke base in beam; then install beam on translating structure. Temporarily install blocker door operating rods to yoke.

(n) Perform steps (a) through (j) and (m) on opposite beam assembly.

NOTE: The blocker door lever arm inspection procedure states that the -511 and -512 lever arms do not require this inspection, however, this procedure required all (inner and outer) lever arms to be inspected. A dimensional check of all lever arms must also be checked. (See figure 78-30-305.)

(15) Remove blocker door lever arms and perform the listed inspections and installation procedures. (Refer to Blocker Doors Lever Arms Penetrant Inspection - P/N 232-20258-505, -506, -507, -508, -509, and -510.)

(16) Inspect blocker doors for cracks, loose attaching fitting, hot spots, and chafing. Replace as required.

(17) Inspect tailpipe seal for cuts, tears, and general condition. (Refer to Tailpipe Seal Repairs.)

(18) Check that all operating rods and arms are temporarily installed, then rig translating structure. (Refer to Initial Translating Structure Rigging.)

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(19) Subsequent to installing the translating structure in the nacelle aft fairing, perform the following items as referenced in this chapter.

a. Adjusting Thrust Reverser Blocker Doors

b. Sequencing Latch Replacement

c. Functional Check of Thrust Reverser Sequencing Latches

d. Dimensional Requirements of Replacing Pneumatic Latch(es)

NOTE: The maintenance requirements as outlined in the Maintenance Manual contains procedures that perform a complete removal, installation, and operational check on one item; therefore, when performing the functional checks, the steps that are redundant or have been previously accomplished may be disregarded. The most important part of each check is that the dimensional limits and functional operation of each component are complied with. The order in which the items are performed are at the option of the individual person performing the tasks.

(20) Lubricate thrust reversers. [Refer to Servicing Thrust Reverser System (Lubrication and Cleaning)].

(21) Check complete thrust reverser assembly for component security, proper safetywire, and lock pin installation, proper electrical cable routing, pneumatic line security, chaffing between component parts, and the freedom of all foreign objects.

(22) Install nacelle (engine) aft fairing (scoop). [Refer to Removal and Installation of Nacelle Aft Fairing (Scoop).]

(23) Install engine aft fairing on pylon. (Refer to Installing Engine Aft Fairing in chapter 71.)

(24) Perform applicable operational test of thrust reverser.

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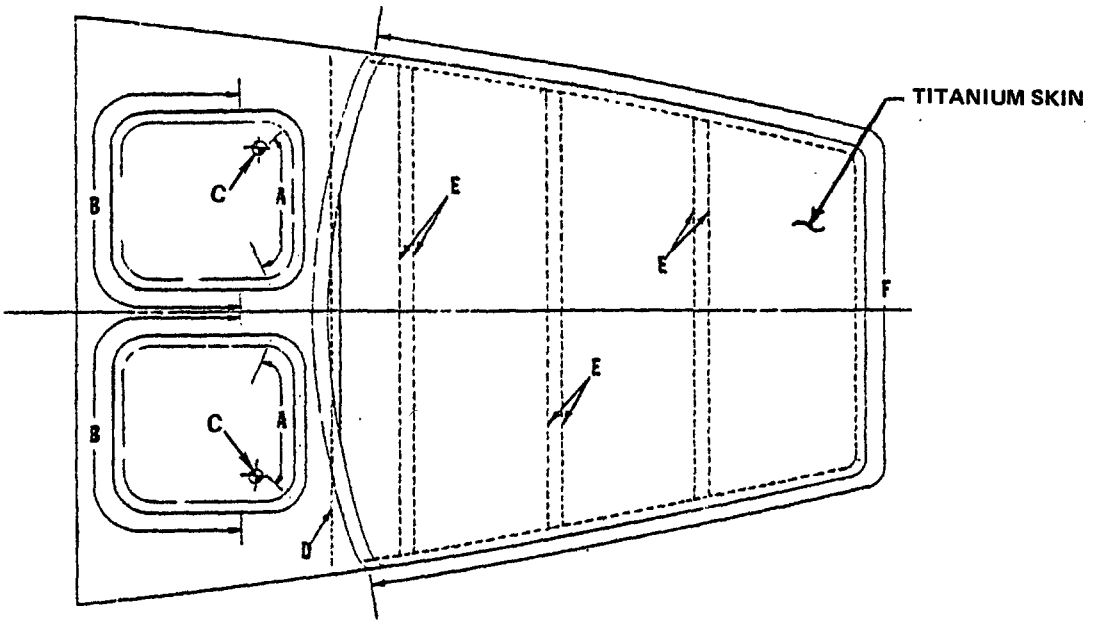
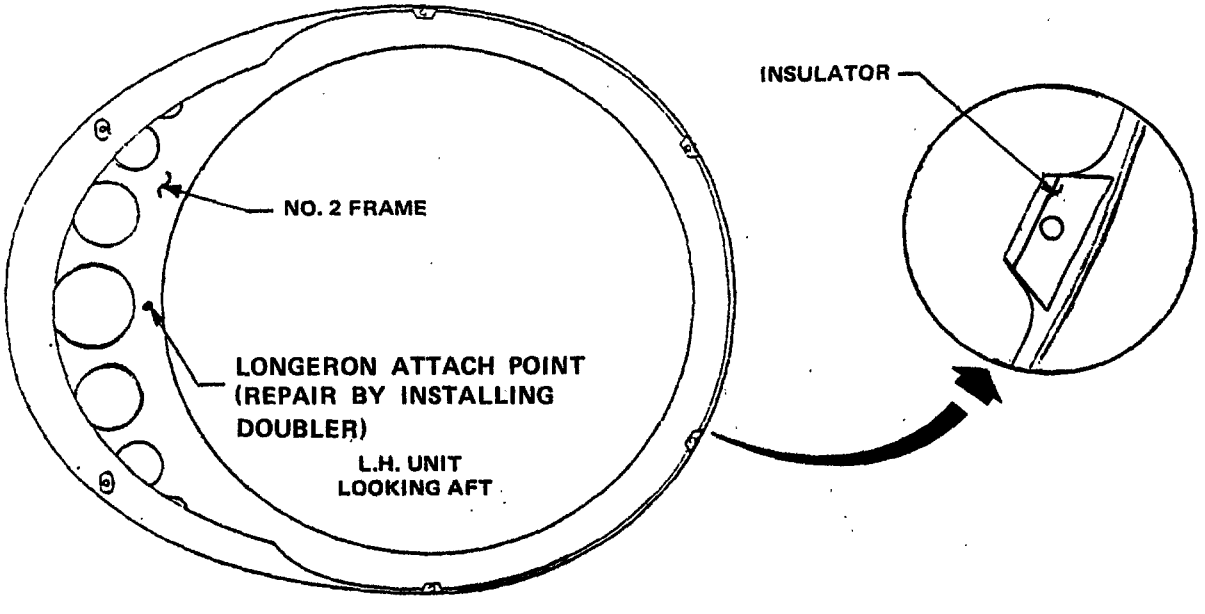
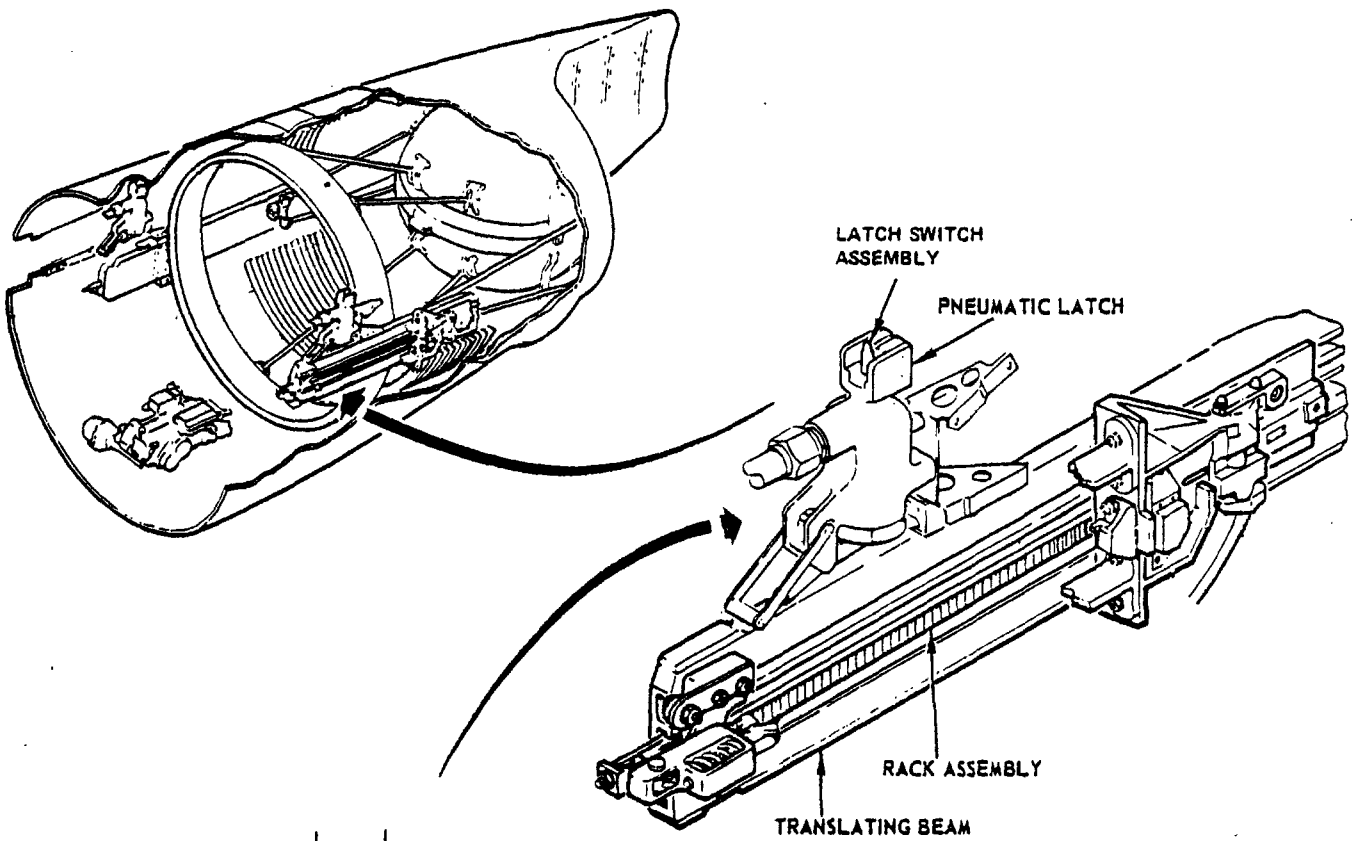
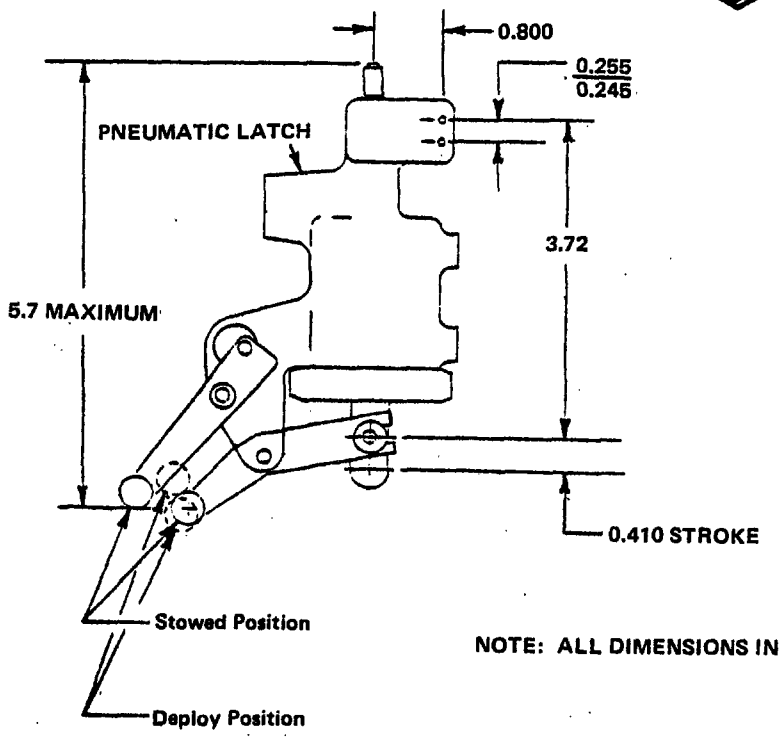


A	CR2235-4-1	D	NAS1738M4
B	MS20426AD4	E	NAS1739M4
C	LOW CORNER	F	NAS1739M4 OR MS20427M4 (EITHER OR - DO NOT MIX)



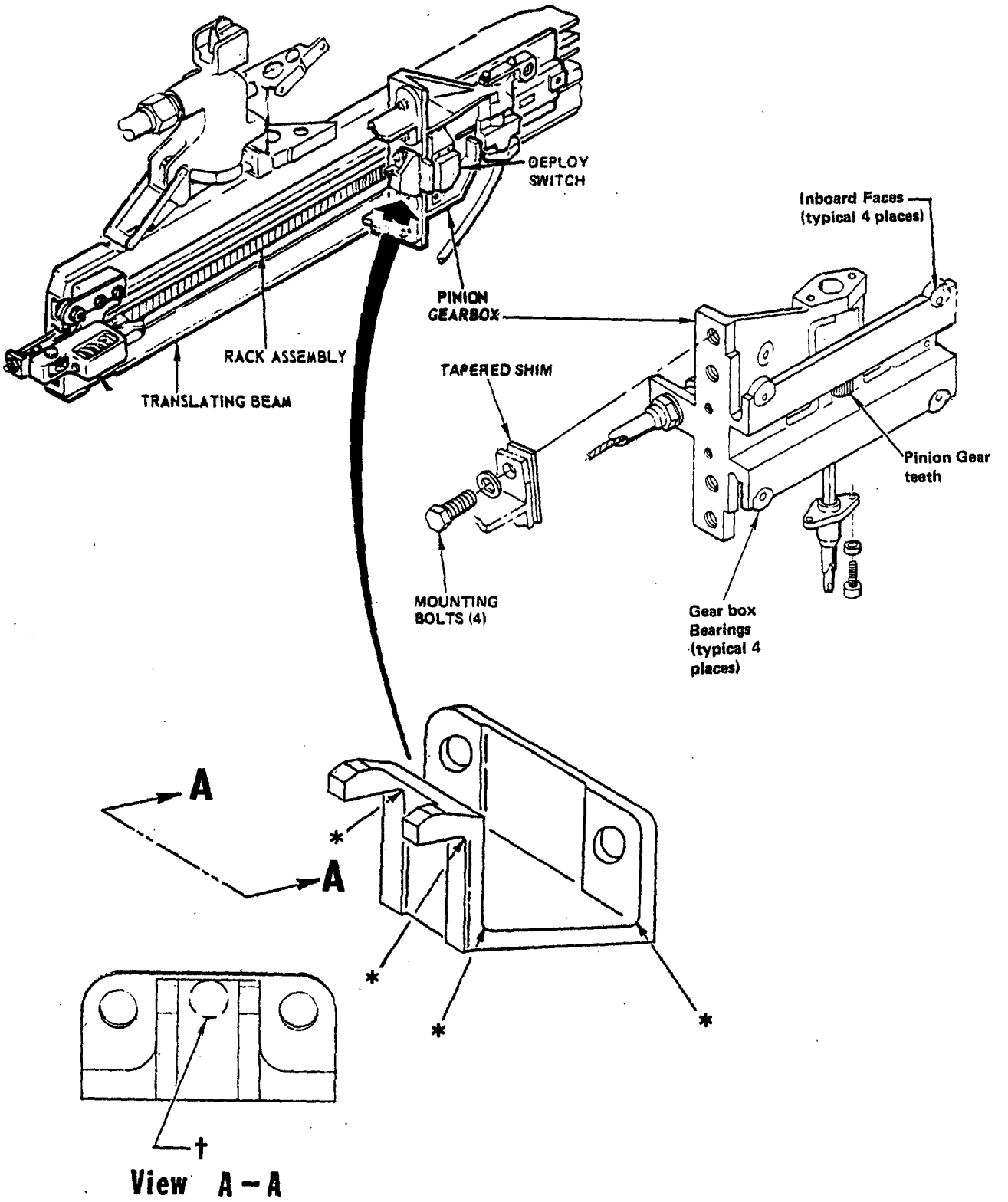
380-2-48-20A





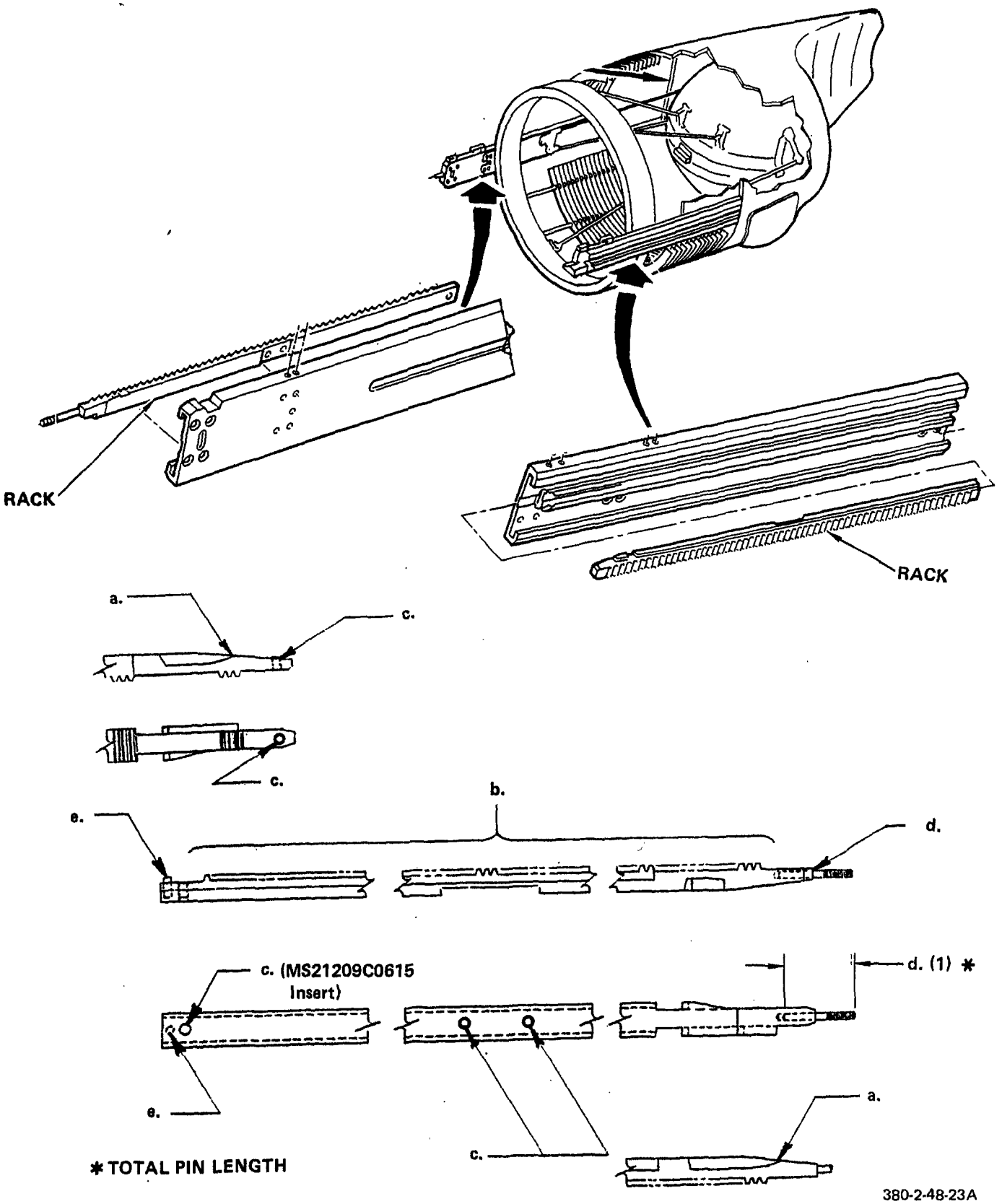
NOTE: ALL DIMENSIONS IN INCHES.

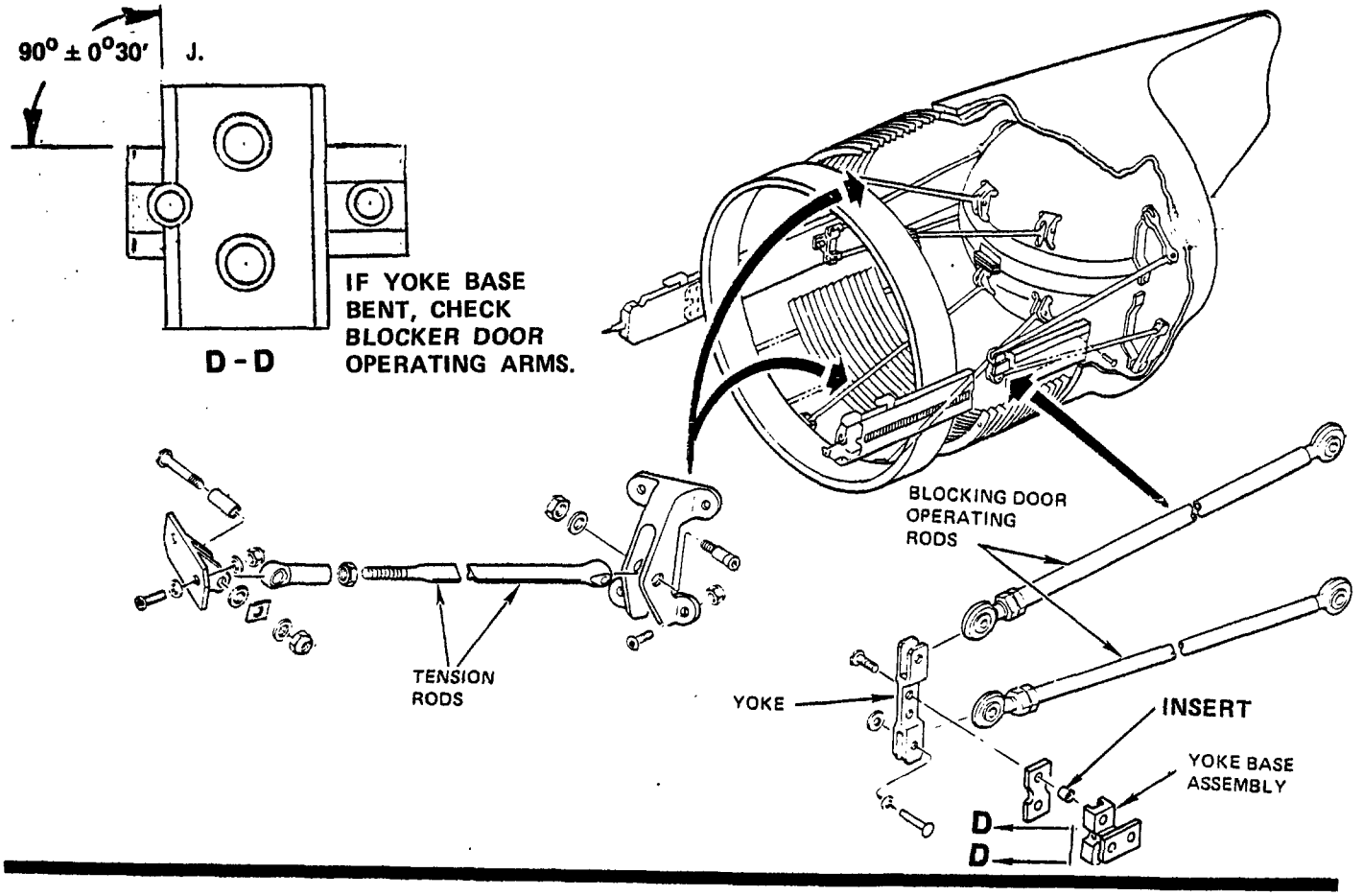
380-2-48-21



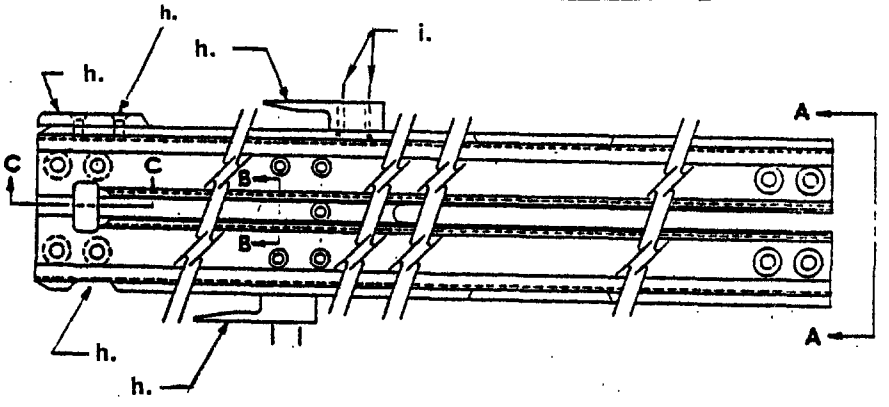
* Inspect for cracks
† Latch pin engagement hole.

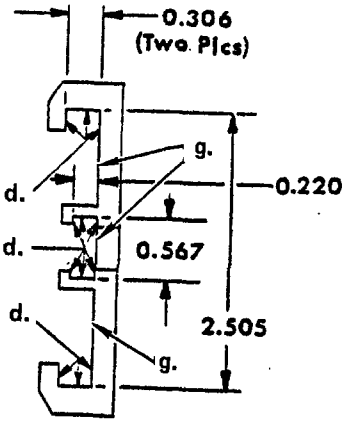
380-2-48-22

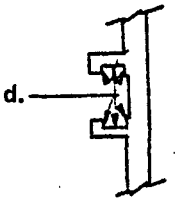

RACK
RACK
a.
c.
c.
b.
e.
d.
c. (MS21209C0615
Insert)
d. (1) *
e.
a.
c.
* TOTAL PIN LENGTH
380-2-48-23A



BEAM INSPECTION

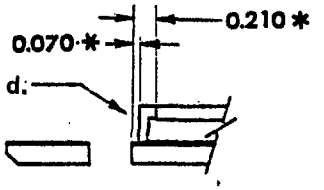






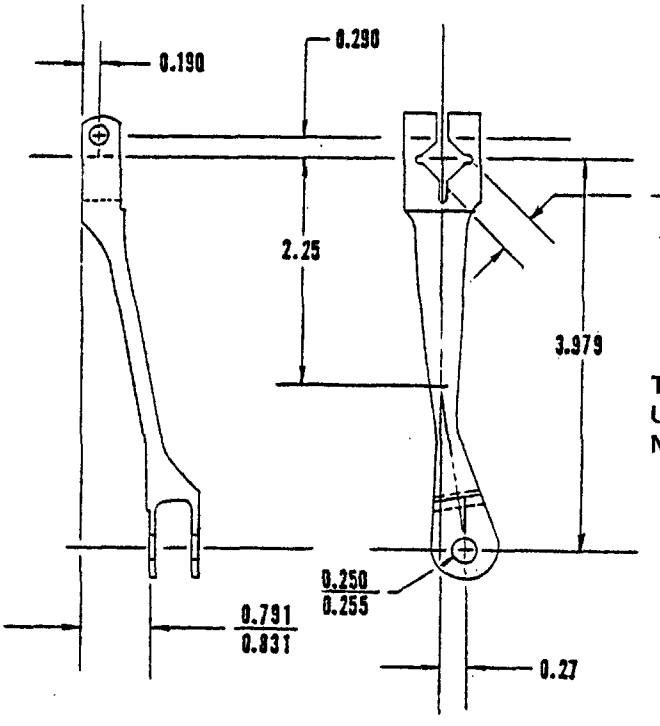
CHECK FOR BRIGHT ALUMINUM IN ALL AREAS MARKED D. IF BRIGHT ALUMINUM PRESENT, REPLACE BEAM.

NOTE: All dimensions in inches.



*±0.010

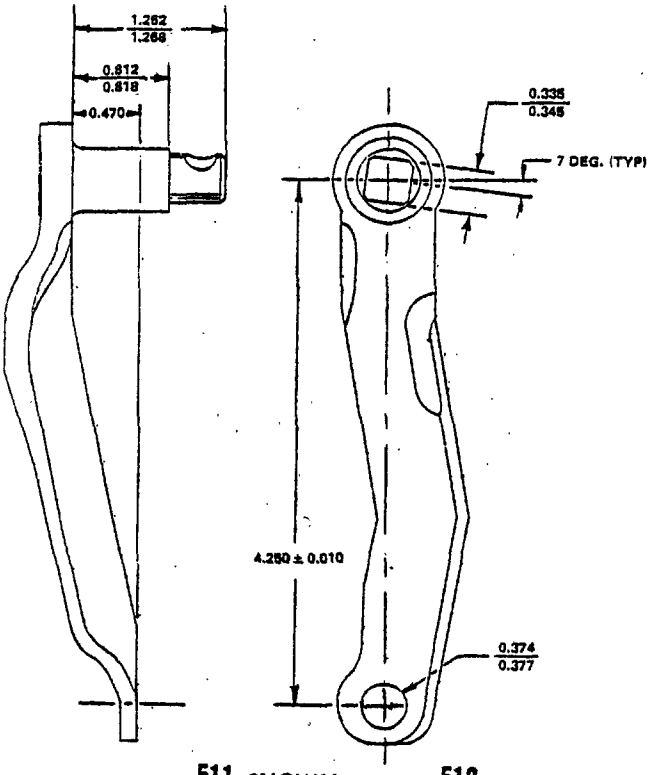
380-2-48-24A



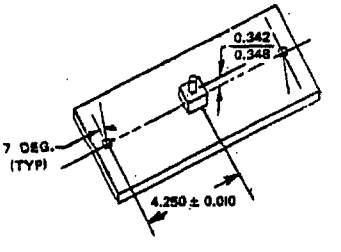
NOTE

TOLERANCE ± 0.010 UNLESS OTHERWISE NOTED

-1 SHOWN -2 OPPOSITE

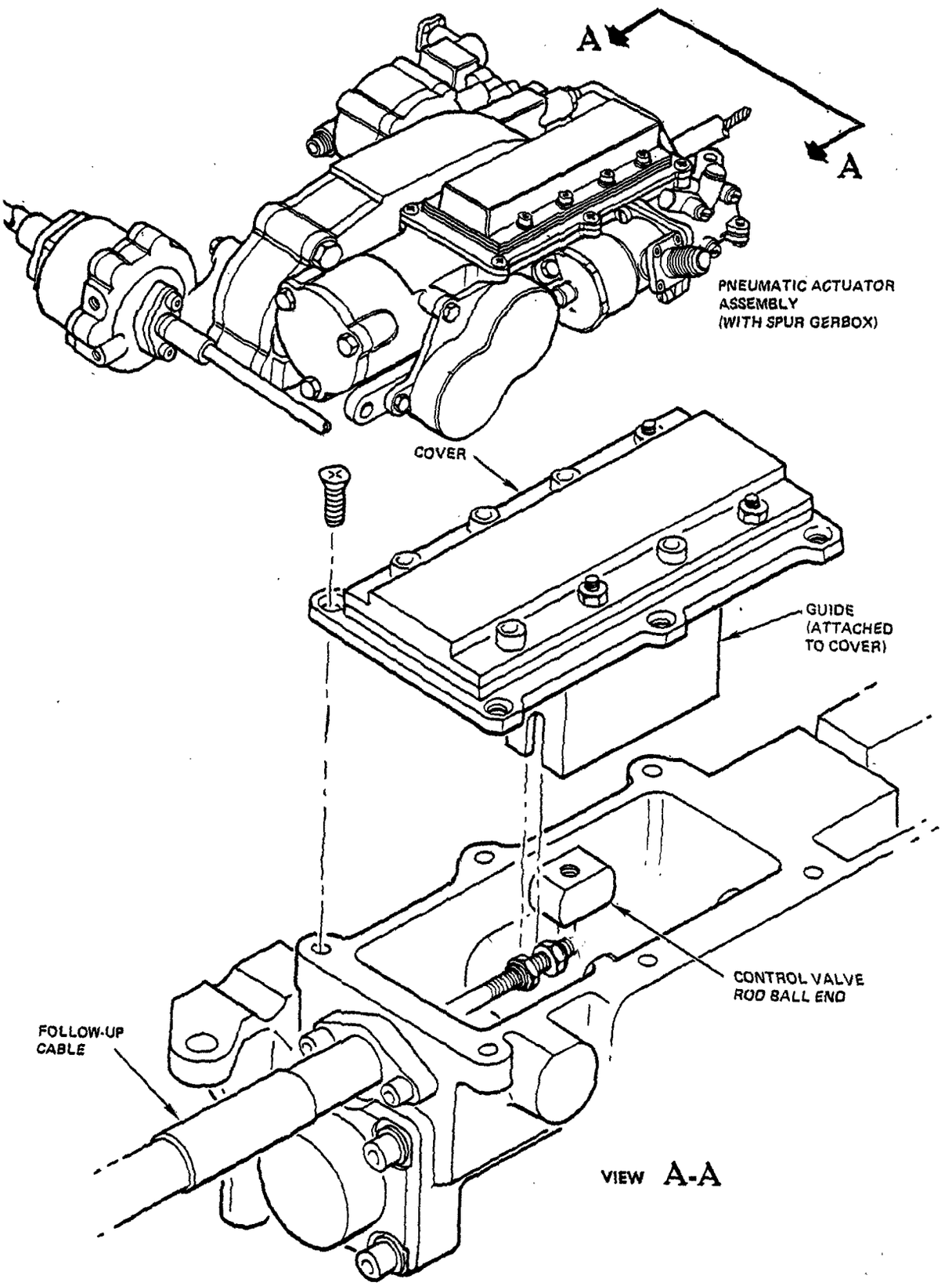


-511 SHOWN -512 OPPOSITE



CHECK FIXTURE

380-2-48-25A



Thrust Reverser Inspection/Repair
Figure 78-30-36

Table 1 - Aeronca and Sabreliner Service Bulletins

74-15 - Thrust Reverser (Aeronca) - Rework of Actuator Link Lever Arms, and Trim of Doubler in Exhaust Nozzle
74-19 - Thrust Reverser (Aeronca) - Inspection for Clearance of Blocker Doors
75-10 - Thrust Reverser (Aeronca) - Inspection of Engine Aft Fairing Installation
75-14 - Thrust Reverser (Aeronca) - Replacement of Existing P/N 232-20258-501, or -503 and -502 or -504 Lever Arms for Blocker Doors
SSB#14 - Thrust Reverser (Aeronca) - Inspection of 232-20258-505, -506, -507 and -508 Lever Arms
S75-78-30-010C - Blocker Door Actuator Link Lever Assemblies, Part Nos. 232-20258-503/-504, -505/-506, -507/-508, -509/-510 and -511/-512. Field Replacement of -507/-508 and -509/-510 Assemblies
S75-78-30-012 - Reinforcement of P/N 232-20101-59 Brackets

Table 2-Aeronca Service Changes

SC-S75-001 - Relocation of Bleed-Off Valves
SC-S75-002 - Pressure Switch Replacement

MAINTENANCE PRACTICES (Continued)

1. OPERATIONAL TEST OF THRUST REVERSER SYSTEM — ENGINE OPERATING

CAUTION

To completely check out the thrust reversers, the cockpit indications must be checked per step A and the actual thrust reverser operation must be checked per step B. Certain thrust reverser malfunctions cannot be detected from cockpit indications alone. Only close observation of thrust reverser operation can verify that the thrust reversers are deploying and stowing properly.

A. To check the thrust reverser system cockpit indications, proceed as follows:

NOTE

All engine nacelle access doors and panels must be installed and closed, and the thrust reverser in the stowed position. Control gust locks must be disengaged. This operational check is used to verify system integrity.

(1) Start both engines. (Refer to chapter 71).

(2) Place LH and RH power levers at IDLE (both engines operating at IDLE), and the thrust reverser control switch at ON. Check that both reverser ARM lights (amber) are on.

(3) Place LH reverser lever to deploy (to interlock stop). Check that the UNLOCK indicator light (red) comes on. Thrust reverser should deploy within two seconds. Check that DEPLOY indicator light (blue) is on and engine rpm is approximately 64 to 68%, but does not exceed 68% (If engine rpm is incorrect, check throttle rigging).

(4) When thrust reverser deployment has been verified, position the reverser lever to the stow position. Check to make certain that both the DEPLOY and UNLOCK indicator light goes out. Both ARM lights should be ON as long as the power levers are at IDLE and the thrust reverser control switch is at ON. Thrust reverser should stow within 3 seconds.

CAUTION

Do not operate engine at 72% rpm with reverser deployed, for more than 5 to 10 seconds.

(5) Deploy thrust reverser as indicated in step 3 of this procedure. When DEPLOY lights come on, increase the rpm of the engine to 72% rpm by moving the reverser lever aft past the interlock stop. When the 72% rpm engine speed is obtained, stow the reverser by positioning the thrust reverser control switch to EMERG STOW. Check and make certain that the DEPLOY and UNLOCK indicator lights are out, and ARM indicating light stays on.

(6) After thrust reverser is stowed, check that the ARM indicating light goes out at 79-82% rpm, when advancing the reverser lever to the maximum position (physicaly stop). After engine rpm has stabilized, check that rpm of the engine to see if it is within 95-97% rpm, but does not exceed 97.5% rpm.

NOTE

If engine rpm is not in range specified, reposition the stop by adjusting stop screws. (See Figure 76-00-1A).

(7) Slowly decrase engine speed to IDLE by positioning the reverser lever to STOW (forward and down), and check that ARM light comes on at 82-79% rpm. If ARM light does not come on within these limits, refer to "Adjusting Thrust Reverser 80 Percent RPM Limit Switch."

(8) Perform RH engine thrust reverser check, repeat steps (3) through (7).

(9) Deploy both thrust reversers simultaneously. Deploy lights should be within one second of each other. Engines should spool up together and be within 2% of the same rpm at idle (If engines do not spool up together or are not within 2%, check throttle rigging).

(10) On S/N 380-5, -10, -12 and subs, check operation of auto stow system if required. (Refer to "Operational Test of Thrust Reverser Auto Stow System").

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MAINTENANCE PRACTICES (Continued)

1. OPERATIONAL TEST OF THE THRUST REVERSER SYSTEM — ENGINE OPERATING (Cont'd)

(11) Position thrust reverser control switch at OFF and place switch guard down. Thrust reverser ARM indicator lights should go out.

B. To check thrust reverser deploy and stow operation, proceed as follows:

(1) While the thrust reversers are alternately deployed and stowed and the engines run at varying rpm's, observe the thrust reversers for the following indications:

On Deploy

(a) Translating structure operates smoothly without bouncing or chattering.

(b) Doors close smoothly and firmly with NO bounce.

(c) Doors do not flutter or vibrate with engine thrust against them.

(d) Doors do not move in any way after initial contact. They close and remain rock steady until stowed.

On Stow

(a) Doors clear tail pipe cleanly.

(b) Translating structure operates smoothly without bouncing or chattering.

(c) Scoop approaches nacelle and stops firmly. The scoop does not stop then relax away from the nacelle on either or both

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sides. (Check follow-up cable if both sides relax. (Check pinion gear housing, drive cables, and translating structure if only one side relaxes).

(2) Position thrust reverser control switch at OFF and place switch guard down.

(3) Shut down engines (Refer to Chapter 71).

2. Operational Check of Thrust Reverser - Engines Not Operating

A. The reverser may be operated while installed on the engine or with the nacelle aft fairing removed and secured in a supporting fixture by one of the following methods:

NOTE: Control gust locks must be disengaged.

(1) If the reverser is installed on the engine, connect a flexible hose (one inch diameter) from the thrust reverser pneumatic test panel, Aeronca P/N M304522 (figure 78-30-37) or a locally manufactured pressure regulator assembly to the forward end of the 234-22550-1 airline on the left engine or the 234-22550-3, air line on the right engine. Access to these lines is through the right hand bottom engine access door. Disconnect the line at it's forward end (first fitting out of engine), loosen the Wiggins coupling, slide fitting back and connect the flexible hose.

(2) If the reverser is not installed on the engine, connect a flexible hose (one inch diameter) from the thrust reverser test panel, P/N M304522 or the locally assembled regulator assembly to the pneumatic actuator air supply line P/N 232-20169-505, LH or 232-20169-503, RH.

NOTE: Ensure that the nacelle aft fairing is securely attached to a supporting fixture and that there is sufficient clearance for reverser operation.

B. Connect pneumatic source to thrust reversers as follows:

(1) Connect the pneumatic test panel or pressure regulator assembly to the air source (shop air or nitrogen bottle).

NOTE: If the Aeronca electrical test set (P/N 234-23210) or the locally manufactured test set is used, perform all procedures of step (2). If the airplane controls are used (using a electrical jumper cable between thrust reverser and the airplane electrical plug), perform all procedures of step (3).

(2) To stow or deploy the thrust reversers using test sets, proceed as follows:

(a) Connect the power supply lead of the Aeronca test set P/N 234-23210 or locally manufactured test set (figure 78-30-38) to a 115 vac power source. Connect the other test set lead to the nacelle aft fairing connector of the thrust reverser electrical harness.

(b) Push test set POWER switch. Test set 110 volt AC and 28 volt DC power lights should illuminate.

(c) Position arm switch at ON.

(d) Close both pneumatic values on pneumatic test panel or if a pressure regulator assemblies used, adjust regulator to zero pressure output. Apply air pressure to test panel or regulator.

(e) If the pneumatic test panel used, open the inlet valve, adjust pressure to 20 psi and open the outlet valve to the thrust reverser. If the regulator assembly is used, adjust outlet pressure to 20 psi.

(f) Check that test set ARM light illuminates.

NOTE: To adequately check all items in step (g), the thrust reverser must be deployed and stowed several times. To stow the reverser for another deploy cycle, select STOW on the test set.

(g) Select DEPLOY on the test set and observe the following indications:

- UNLOCK light illuminates.
- Deploy light illuminates within two seconds.
- Translating structure operates smoothly without bouncing or chattering.
- Blocker doors close smoothly and firmly with NO bounce.
- Doors do not move in any way after initial contact. They close and remain rock steady until stowed.

(h) Depress BLEED OFF button. ARM light should go out. Release BLEED OFF button. ARM light will illuminate.

NOTE: To adequately check all items in step (i), the thrust reverser must be deployed and stowed several times.

(i) Select STOW on the test set and observe the following indications:

- DEPLOY light goes out.
- UNLOCK light goes out within three seconds.
- Doors clear tailpipe cleanly (if installed on airplane).
- Translating structure operates smoothly without bouncing or chattering.
- Scoop approaches nacelle and stops firmly on both inboard and outboard sides. The scoop does not stop then relay away from nacelle on either or both sides. (Check follow-up cable if both sides relax. Check pinion gear housings, drive cables, and translating structure if only one side relaxes).

(j) Position toggle switch to STOW.

(k) Position toggle switch OFF.

(l) Open test set ARM switch. ARM light should go out.

(m) Position (pull) test set POWER switch to OFF.

(n) Remove air pressure and disconnect pneumatic test panel or pressure regulator from thrust reverser.

(o) Disconnect electrical test set from thrust reverser.

(p) Connect and safety (Wiggins fitting) thrust reverser engine air lines and electrical cable as applicable.

(3) To stow or deploy the thrust reversers using the aircraft controls, proceed as follows:

(a) If the nacelle aft fairing is installed, connect the engine to reverser electrical connector. If the nacelle aft fairing is removed, connect the engine electrical connector to the reverser electrical connector using a locally manufactured jumper cable.

(b) Disengage the applicable engine IGN & BOOST CONT and FUEL BOOST PUMP circuit breakers. Ensure that the LDG POS circuit breaker is engaged.

(c) Disengage gust lock.

(d) Apply external 28 volt DC power to aircraft.

(e) Position THRUST REVERSER CONTROL switch to ON.

(f) Position power lever to IDLE. ARM light will illumnate.

(g) Position power lever above IDLE. ARM light is not illuminated.

(h) Position power lever to IDLE. ARM light is illuminated.

NOTE: To adequately check all items in step (i), two men are required and the thrust reverser must be alternately deployed and stowed several times.

(i) Deploy thrust reverser and observe the following:

- UNLOCK light illuminates.
- DEPLOY light illuminates within two seconds.
- Translating structure operates smoothly with chattering or bouncing.
- Blocker doors close smoothly and firmly with NO bounce.
- Doors do not move in any way after initial contact. They close and remain rock steady until stowed.

(j) Move reverser lever forward beyond 80% RPM position. ARM light should go out. Move reverser lever aft to below 80% RPM position. ARM light should illuminate.

NOTE: To adequately check all items in step (k), two men are required and the thrust reverser must be alternately deployed and stowed several times.

(k) Stow thrust reverser and observe the following:

- DEPLOY light goes out.
- UNLOCK light goes out within three seconds.

- Doors clear tailpipe cleanly (if installed on airplane).
- Translating structure operates smoothly without bouncing or chattering.
- Scoop approaches nacelle and stops firmly on both inboard and outboard sides. The scoop does not stop then relax away from nacelle on either or both sides. (Check follow-up cable if both sides relax. Check pinion gear housings, drive cables, and translating structure if only one side relaxes).

(l) Position power lever to OFF. Check that ARM light goes out.

(m) Position THRUST REVERSER control switch to OFF.

(n) Disconnect external 28 volt DC utility power.

(o) Remove air pressure and disconnect flexible air line from thrust reverser system.

(p) Disconnect electrical jumper (if applicable).

(q) Connect and safety (Wiggins fitting) thrust reverser air line to engine.

3. Operational Test - Thrust Reverser System and Associated Controls

A. To operational test thrust reverser and associated controls, proceed as follows:

WARNING: ON NA380-5, -10, -12 AND LATER AIRPLANES, TO PREVENT AUTOSTOW OF THRUST REVERSER, ALWAYS PULL EMERGENCY STOW CIRCUIT BREAKER IF OPERATION IS INTERRUPTED OR AIRCRAFT IS LEFT UNATTENDED.

- ANY INTERRUPTION OF ELECTRICAL POWER TO THE LANDING GEAR DOWNLOCK RELAYS MAY CAUSE AN AUTOMATIC STOW OF THE REVERSER. DO NOT PERMIT MAINTENANCE ON OR IN THE MAIN LANDING GEAR WHEEL WELL DURING THIS CHECK OR IF REVERSERS LEFT IN OTHER THAN STOWED POSITION, REGARDLESS OF THRUST REVERSER SWITCH POSITION.

- IF ELECTRICAL POWER IS REMOVED AND/OR REAPPLIED WITH THE REVERSERS IN ANY OTHER POSITION THAN FULLY STOWED, AN AUTOMATIC STOW OF THE REVERSER WILL OCCUR.

- AN AUTOMATIC STOW CAN BE IDENTIFIED BY ILLUMINATION OF THE THRUST REVERSER UNSAFE LIGHT AND MASTER CAUTION LIGHT.

- THE AUTOMATIC STOW CIRCUIT INCORPORATES A HOLDING CIRCUIT. TO OPEN HOLDING CIRCUIT, OPEN AND CLOSE THE APPROPRIATE LH OR RH IND LTS AND LEVER CIRCUIT BREAKER.

(1) Open LH FUEL-BOOST PUMP, RH FUEL-BOOST PUMP, LH ENGINE-IGN & BOOST CONT, RH ENGINE-ING & BOOST CONT, RH STALL WARNING-INBD, OUTBD-VANE HEATHERS, LH STALL WARNING-INBD AND OUTBD-VANE HEATHERS, and on NA380-5, -10, -12 and later airplanes the EMER STOW circuit breakers.

(2) Close LDG GEAR-POS, and all (except EMER STOW on NA380-5, -10, -12 and later) thrust reverser circuit breakers.

(3) Open applicable engine access doors. Connect a flexible hose (one diameter) from the thrust reverser pneumatic test panel, Aeronca P/N M304522 **(figure 78-30-37)** or a locally manufactured pressure regulator assembly, to the forward end of the 234-22550-1 air line on the left engine or the 234-22550-3 air line on the right engine. Disconnect the line at it's forward end (first fitting out of engine) and slide the Wiggins fitting back and connect the air line.

NOTE: If a regulated air source cannot be applied to both reverser actuators simultaneously, the operational proceedures can be performed on each reverser individually.

(4) Regulate air pressure to 20 psig.

(5) Apply external 28 VDC electrical power to airplane.

(6) Position THRUST REVERSER control switch to ON. Check that all thrust reverser indicating lights are out.

(7) Position left and right power levers to IDLE. Check that the LH T/R ARM and RH T/R ARM indicator lights are illuminated.

(8) Advance left and right power levers to above the IDLE position. (over 80% engine RPM position). Check that the LH T/R ARM and RH T/R ARM indicator lights are out and that thrust reverser levers are locked in stowed position.

(9) Retard left and right power levers to the IDLE position.

CAUTION: MAKE SURE THAT ALL PERSONNEL ARE CLEAR OF TRANSLATING STRUCTURE (AFT FAIRING) BEFORE OPERATING SYSTEM.

(10) Raise reverser levers up and aft to the interlock stop. Check for the following items:

(a) LH T/R ARM light - Illuminated
(b) LH T/R UNLOCK light - Illuminated
(c) LH T/R DEPLOY light - Illuminated (after approximately 2 or 3 seconds).
(d) RH T/R ARM light - Illuminated
(e) RH T/R UNLOCK light - Illuminated

(f) RH T/R DEPLOY light - Illuminated (after approximately 2 or 3 seconds).

(g) Power Levers - Locked in IDLE position

(11) Move reverser levers aft beyond the 80% RPM position. Check that LH and RH T/R ARM light go out.

(12) Place reverser levers full down. Check for the following items:

(a) LH T/R ARM - Illuminated
(b) LH T/R UNLOCK - Out (after approximately 3 seconds)
(c) LH T/R DEPLOY - Out
(d) RH T/R ARM - Illuminated
(e) RH T/R UNLOCK - Out (after approximately 3 seconds)
(f) RH T/R DEPLOY - Out

(13) Move reverser levers to interlock position. Check that all thrust reverser indicator lights are illuminated.

(14) On NA380-5, -10, -12 and later, engage (push-in) the EMER STOW circuit breakers.

(15) Position THRUST REVERSER control switch at EMER STOW. Check that thrust reverser is stowed and the DEPLOY and UNLOCK indicator lights are out.

(16) Position THRUST REVERSER control switch at ON. Reversers should deploy. Check that all reverser indicator lights are on.

(17) Move reverser levers full down. Thrust reversers should stow.

(18) On NA380-5, -10, -12 and later, pull out (disengage) the EMER STOW circuit breakers.

(19) Position THRUST REVERSER control switch to OFF.

(20) Check that reverser levers do not move past the first detent (interlock stop). Check that power lever movement is blocked. Return reverser levers to the stow position.

(21) Move power levers to half power position. Check that reverser lever movement is blocked.

(22) Position power levers to IDLE position.

(23) Position left and right power levers to the CUT-OFF position.

(24) Position the THRUST REVERSER control switch of OFF.

(25) Remove external electrical power from airplane.

(26) Turn off regulated air source to thrust reverser actuators.

(27) Disconnect test panel from thrust reverser and reconnect air lines 234-22500-1 and/or 234-22500-3 as applicable.

(28) Close engine access doors.

(29) Close the LH FUEL-BOOST PUMP, RH FUEL-BOOST PUMP, LH ENGINE-IGN & BOOST CONT, RH ENGINE-IGN & BOOST CONT, RH STALL WARNING-INBD, OUTBD, VANE HEATHERS, LH STALL WARNING-INBD, OUTBD-VANE HEATHERS circuit breakers. Also the EMER STOW circuit breakers (if applicable).

4. Operational Checkout of Thrust Reverser Auto Stow System - NA380-5, -10, -12 and Later Airplanes

A. This check may be performed with an external air supply connected to the reverser or with the engine operating. If the engines are operated for this check, perform all procedures in step (1). If a pneumatic air source is used, perform all procedures in step (2).

(1) To perform operational check of auto stow system with the engine operating, proceed as follows:

(a) Close the engine cowling and install all access doors on nacelle aft fairing.

(b) Open two LH stall warning, INBD and OUTBD VANE HEATERS circuit breakers and the EPR-PITOT, STALL WARN HTR CONTROL circuit breaker and the two RH STALL WARNING, INBD and OUTBD VANE HEATERS circuit breakers.

(c) Apply external 28 V dc electrical power to airplane.

(d) Start engine(s).

(e) Position thrust reverser control switch at ON and deploy thrust reversers.

(f) Open (pull out) the LDG GEAR POS circuit breaker. Thrust reverser(s) should stow.

(g) Check that reverser ARM and UNLOCK lights illuminate. Check that the THRUST REVERSER UNSAFE light and the MASTER CAUTION light illuminates.

(h) Observing brilliance of the UNLOCK light, position

thrust reverser control switch to OFF. UNLOCK light brilliance should increase.

(i) Position reverser lever to stow.

(j) Push in LDG GEAR POS circuit breaker.

NOTE: Step (k) is required to deenergize an electrical holding circuit created by the Auto Stow System.

(k) Momentarily pull then push the IND LTS & LEVER circuit breaker in. Check that the THRUST REVERSER UNSAFE and UNLOCK lights go out.

(l) Check thrust reverser system for normal operation.

(m) Remove electrical power from airplane.

(n) Push in the two LH STALL WARNING, INBD and OUTBD VANE HEATERS circuit breakers and the EPR-PITOT, STALL WARN HTR CONTROL circuit breaker and the two RH STALL WARNING, INBD and OUTBD VANE HEATERS circuit breakers.

(2) To perform operational check of auto stow system using an external pneumatic power source, proceed as follows:

(a) Open RH engine access door. Connect a flexible hose (one in. diameter) from the thrust reverser pneumatic test panel (Aeronco P/N M304522), or a locally manufactured pressure regulator assembly, to the fwd end of the air line on the LH or RH engine. (See figure 78-30-37.) Disconnect the air line at fwd end (first fitting out of engine), slide Wiggins fitting back, and connect to external pneumatic power source.

(b) Pull two LH STALL WARNING, INBD and OUTBD VANE HEATERS circuit breakers, the EPR-PITOT, STALL WARN HTR CONTROL circuit breaker, the two RH STALL WARNING, INBD and OUTBD VANE HEATERS circuit breakers, the LH FUEL-BOOST PUMP, RH FUEL-BOOST PUMP, LH ENGINE-IGN & BOOST CONT, and the RH ENGINE-IGN & BOOST CONT circuit breakers.

(c) Apply external electrical 28 vdc power to airplane.

(d) Apply 20 psi air pressure to reverser.

(e) Set THRUST REVERSER CONTROL switch at ON and observe that ARM, UNLOCK, DEPLOY, THRUST REVERSER UNSAFE, and MASTER CAUTION lights off.

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(f) Set throttle levers at idle and observe the following:

- ARM light comes on as soon as T/R manifold pressurizes.
- UNLOCK, DEPLOY, THRUST REVERSER UNSAFE, AND MASTER CAUTION lights remain off.

(g) Deploy thrust reversers and observe the following:

- ARM light remains ON.
- UNLOCK light comes on as soon as either inbd or outbd pneumatic latch unlatches.
- DEPLOY light comes on when reversers deployed
- THRUST REVERSER UNSAFE and MASTER CAUTION lights remain off.

(h) Pull LDG GEAR POS circuit breaker and observe the following:

- MASTER CAUTION and THRUST REVERSER UNSAFE lights come on when circuit breaker is pulled.
- DEPLOY light goes off when thrust reverser starts to stow.
- UNLOCK light goes off when thrust reverser fully stowed.
- ARM light remains ON.

(i) Push LDG GEAR POS circuit breaker and observe the following:

- MASTER CAUTION light goes off immediately.
- THRUST REVERSER UNSAFE and ARM lights remain on.
- UNLOCK and DEPLOY lights remain off.
- Thrust reverser remains stowed.

(j) Pull IND LIGHTS AND LEVER ciruit breaker and observe the following:

- All thrust reverser lights and MASTER CAUTION light are off.
- Thrust reverser deploys.

(k) Push IND LIGHTS AND LEVER circuit breaker and observe the following:

- ARM, UNLOCK and DEPLOY lights come on immediately.
- THRUST REVERSER UNSAFE and MASTER CAUTION lights remain off.
- Thrust reverser remains deployed.

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(l) Set THRUST REVERSER CONTROL switch at OFF and observe the following:

- UNLOCK light becomes brighter.
- ARM light goes off.
- DEPLOY light remains ON.
- THRUST REVERSER UNSAFE and MASTER CAUTION lights remain off.
- Thrust reverser remains deployed.

(m) Set THRUST REVERSER CONTROL switch at EMER STOW and observe the following:

- UNLOCK light brightness decreases immediately but remains on at lower brightness until thrust reverser fully stows.
- ARM light comes on as soon as thrust reverser manifold pressurizes.
- DEPLOY light goes off when thrust reverser starts to stow.
- UNLOCK light goes off when thrust reverser fully stowed. THRUST REVERSER UNSAFE and MASTER CAUTION lights remain off.
- Thrust reverser stowed.

(n) Set THRUST REVERSER CONTROL switch at ON and observe the following:

- UNLOCK light comes on as soon as either inbd or outbd pneumatic latch unlatches.
- DEPLOY light comes on when reverser deploys.
- ARM light remains on.
- THRUST REVERSER UNSAFE and MASTER CAUTION light remain off.

(o) Set T/R levers at stow and observe the following:

- DEPLOY light goes off as soon as thrust reverser starts to stow.
- UNLOCK light goes off when thrust reverser fully stowed.
- ARM light remains on.
- THRUST REVERSER UNSAFE and MASTER CAUTION lights remain off.

(p) Set THRUST REVERSER CONTROL switch at OFF and observe the following:

- ARM light goes off as soon as pressure bleeds off thrust reverser manifold.
- UNLOCK, DEPLOY, THRUST REVERSER UNSAFE, and MASTER CAUTION lights remain off.

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(q) Remove electrical power from airplane.

(r) Push the two LH STALL WARNING, INBD and OUTBD VANE HEATERS circuit breakers and the EPR-PITOT, STALL WARN HTR CONTROL circuit breaker and the two RH STALL WARNING, INBD and OUTBD VANE HEATERS circuit breakers the LH FUEL-BOOST PUMP, RH FUEL-BOOST PUMP, LH ENGINE-IGN & BOOST CONT and the RH ENGINE-IGN & BOOST CONT circuit breakers.

(s) Remove air pressure and flexible hose from engine. Reconnect and safety engine air line (Wiggins fitting). Close engine access doors.

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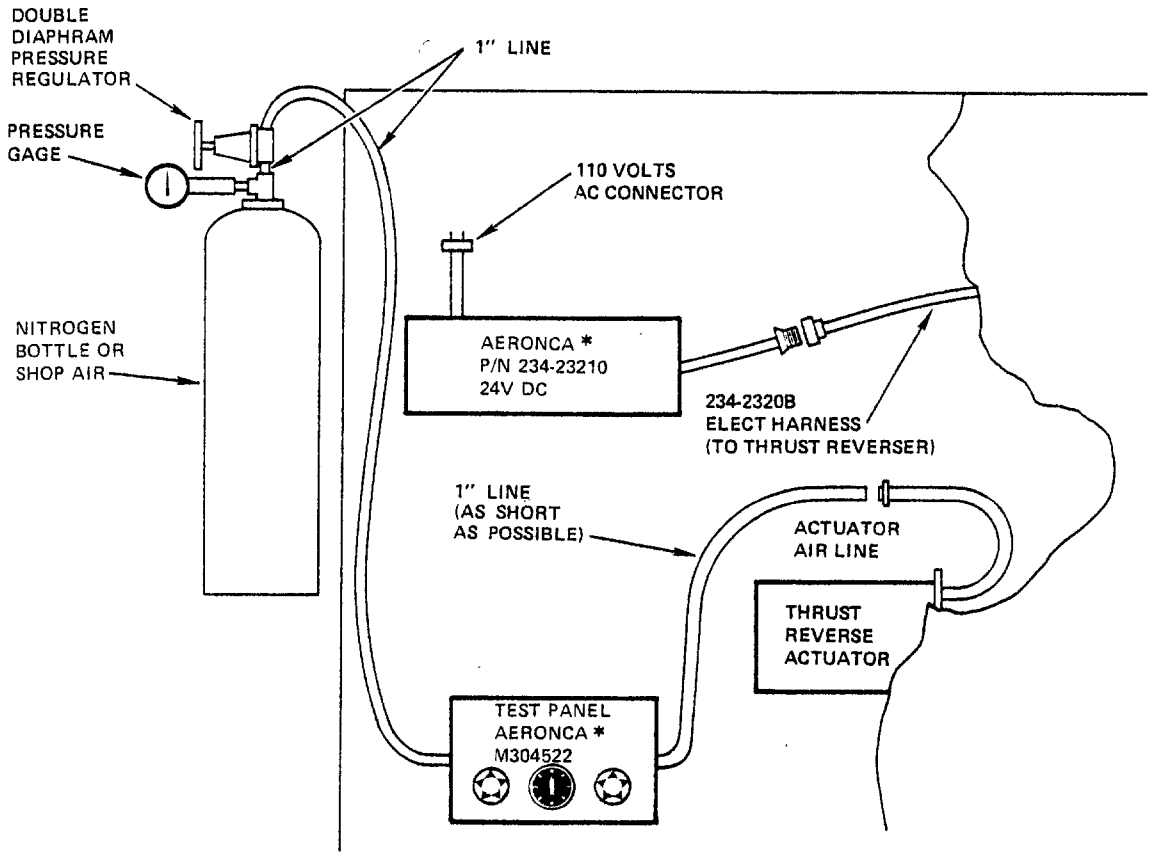
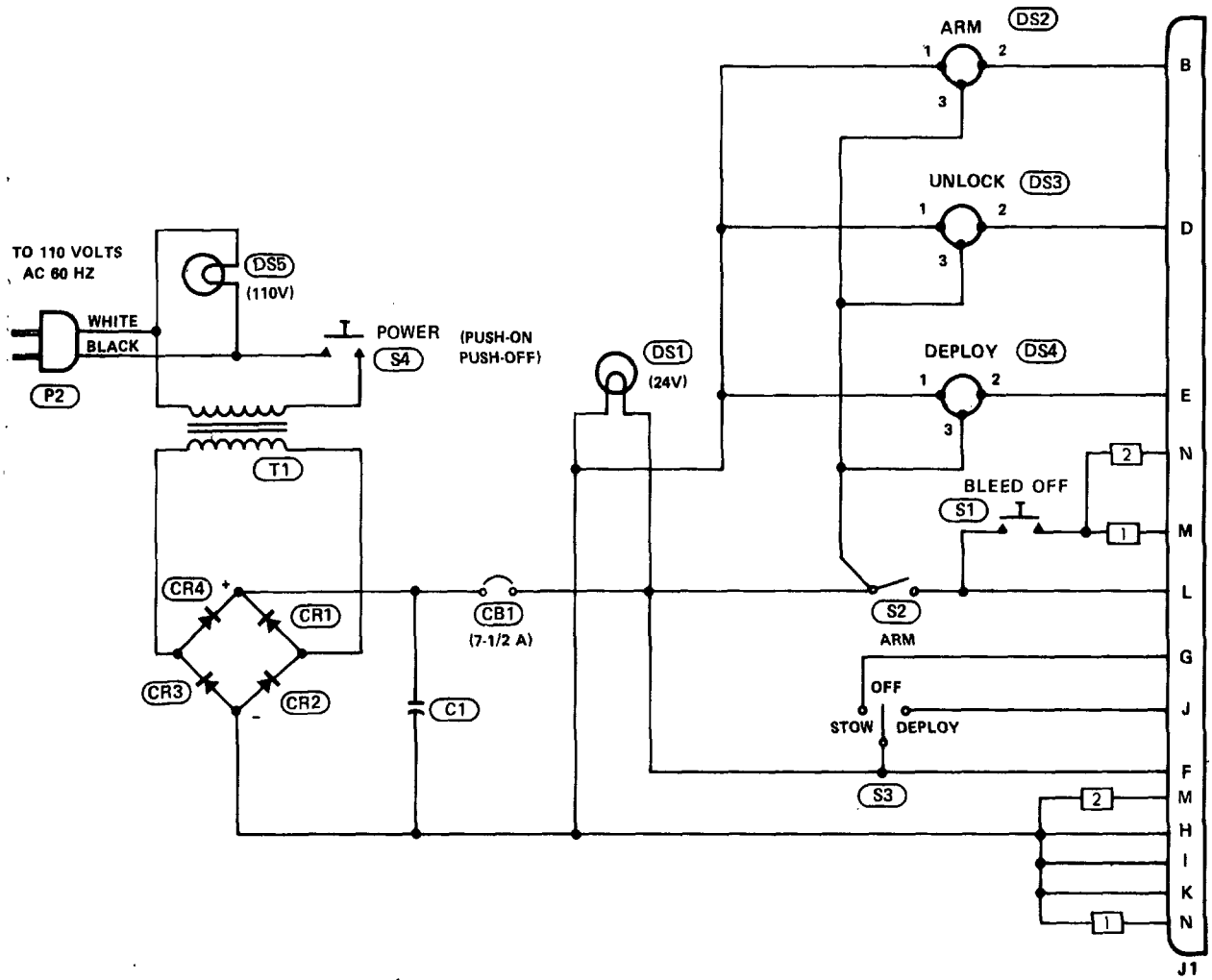


* THE PNEUMATIC REGULATOR AND ELECTRICAL TEST SET MAY BE LOCALLY MANUFACTURED.

S380-2-41-118

Thrust Reverser Test Set
Figure 78-30-37



C1 - CAPACITOR - SPRAGUE P/N 39D207F250JP4 (200 UF AT 150 VDC ELECTROLYTIC)
CB1 - CIRCUIT BREAKER - WOOD ELECTRIC, P/N 507-507-101 (7.5 AMP)
CR1 - DIODE - INTERNATIONAL RECTIFIER, P/N 50FB2L-C (5 AMP 200 PIV)
CR2 - DIODE - INTERNATIONAL RECTIFIER, P/N 50FB2L-C (5 AMP 200 PIV)
CR3 - DIODE - INTERNATIONAL RECTIFIER, P/N 50FB2L-C (5 AMP 200 PIV)
CR4 - DIODE - INTERNATIONAL RECTIFIER, P/N 50FB2L-C (5 AMP 200 PIV)
DS-1 - INDICATOR LIGHT - DIALCO, P/N 81-1059-01-102 (24 VDC BULB)
DS-2 - INDICATOR LIGHT - DIALCO, P/N 820-2701-01-502 (24 VDC BULBS; 820-0112-500 LENS [GREEN])
DS-3 - INDICATOR LIGHT - DIALCO, P/N 820-2701-01-502 (24 VDC BULBS; 820-0112-500 LENS [GREEN])
DS-4 - INDICATOR LIGHT - DIALCO, P/N 820-2701-01-502 (24 VDC BULBS; 820-0112-500 LENS [GREEN])
DS-5 - INDICATOR LIGHT - DIALCO, P/N 95-1308-09-241 (NEON BULB)
J1 - ELECTRICAL RECEPTACLE - COMMON, P/N CA3100KE-20-27S
P2 - ELECTRICAL OUTLET PLUG - LOCAL PURCHASE
S1 - SWITCH - NORMALLY OPEN - ARROW HART, P/N 3391-GL (SPST; N.O.)
S2 - SWITCH - ARROW HART, P/N 82600 (SPST)
S3 - SWITCH - ARROW HART, P/N 82602 (SPDT)
S4 - SWITCH - ARROW HART, P/N 82600 (SPST)
T1 - TRANSFORMER - 115 VAC PRIMARY, 24 VAC SECONDARY AT 5 AMPS

[1] AIRPLANES NA-380-1 THRU 3
[2] AIRPLANES NA-380-4 AND LATER

NOTE Equivalent parts with the same rating may be used.

- All connecting electrical wires should be 22 gage.
- Connector J1 Connects to P969 on LH engine and P971 on RH engine

S380-2-41-12B

Thrust Reverser Electrical Control Test Set - Local Manufacture
Figure 78-30-38

MAINTENANCE PRACTICES (Continued)

5. THRUST REVERSER INSPECTION
300 HOURS

A. Inspect the following thrust reverser components for security, chafing, galling (bright spots), cracks, hot spots, cleanliness, obstructions, freedom of movement, warping, loose components and applicable safeties (see figures 78-30-1, 78-30-3 and 78-30-7A for components identification). This inspection is to be performed with the access plate removed from the aft fairings and thrust reverser deployed to the point prior to movement of the blocker doors.

B. Remove aft fairing access panels, then deploy thrust reversers. Inspect the following components as listed.

(1) Actuator air lines.

(2) Electrical harness and deploy switch.

(3) Direction control valve, solenoid actuator and exhaust port.

(4) Pneumatic latch switches and rollers.

(5) Sequencing latch assembly.

(6) Translating beams.

(7) Cascade assembly.

(8) Visible portions of blocker door tension rods (4).

(9) Blocker doors.

(10) Visible portion of blocker doors operating rods (4).

(11) Yoke assemblies.

(12) Tension strap fittings.

(13) Turnbuckles (2) and rod ends.

(14) Visible portion of engine tail pipe and seal.

(15) Pneumatic air lines.

(16) Air motor, spur gear box and flex drive shafts.

(17) Visible portions of lever arms and levers (4 each).

(18) Push-pull follow-up cable stiffness (3.5 pounds max.).

C. Stow thrust reversers.

D. Install access doors on aft fairing and stow reversers.

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MAINTENANCE PRACTICES (Continued)

1000 HOUR/1000 CYCLE THRUST REVERSER INSPECTION

General

Inspection of the aft engine fairing and thrust reverser is required after each 1000 hours of engine operation or 1000 cycles of the thrust reverser, whichever comes first. This inspection should be done in conjunction with the engine hot section and torque translating inspections.

NOTE

If major discrepancies are found, contact Sabreliner Engineering in St. Louis, Repair Station No. VR2R250L (formerly 3919).

Inspection Procedures

1. Remove engine aft fairing from pylon per Sabreliner Maintenance Manual. Position in suitable stand.
2. Remove nacelle (engine) aft fairing (scoop) per Sabreliner Maintenance Manual.
3. Inspect aft nacelle fairing components per steps (a) thru (d).
   A. Inspect titanium inner skin and all visible internal framers for warping, buckling and distortion.
   B. Inspect complete aft nacelle fairing for nicks, dents and for loose or missing fasteners.
   C. Check for damaged or missing insulators.
   D. Inspect for cracks in No. 2 frame in area of 232-20265 longeron attachment. (See Figure 78-30-39.)
4. Check aft nacelle centering jacks.

   Replace all aft nacelle centering jacks that are binding or are difficult to adjust with new centering jacks that operate properly.
5. Inspect pinion gear boxes per steps (a) thru (e).
   A. Inspect pinion gear box bearing shoes (four each per gear box P/N 3233190-2) for wear. Remove and replace all damaged or worn bearing shoes. (See Figure 78-30-40.)
   B. Check pinion gear teeth for excessive wear, chipped teeth and general condition. Replace gearbox(es) if any one of these defects are found.
   C. Inspect deploy switch and switch brackets (outboard gear box) for cracks, security and general condition.
   D. Check gear box(es) for cracks and corrosion. If any cracks are found on a bolted-on item, the applicable item may be replaced.
   E. Inspect catch on forward end of gear box (Figure 78-30-40) for physical damage and cracks. Replace catch(es) as required.
6. Inspect feedback cable as follows.
   A. With spring scale as slider, make push-pull test (3.5 lbs. max.) using Chatillon Scale, Model DPPH-100 or equivalent.
   B. Check cable for exterior damage. Replace any cable that does not meet specifications.
7. Remove flex shafts in accordance with Maintenance Manual NA-69-420, chapter 78-30-3. Inspect and repair flex shaft assembly in accordance with Garrett AiResearch Overhaul Manual 78-30-11. Install flex shafts in accordance with Maintenance Manual NA-69-420, chapter 78-30-3.
8. Torque translating structure per chapter 78.
9. Check translating assembly for proper rig per steps (a) thru (e) and Figure 78-30-41.
   A. Deploy blocker doors, clamp at tab of blocker doors.
   B. Take measurement at center of clevis end of external arms. This dimension should be the same between top and bottom arms inbd and outbd within $\pm 1/8$ in.
   C. Place translating structure tail pipe down on plate and measure up to arm clevis. This dimension should be $3 1/4 \pm 1/8$ in.
10. Inspect rack per steps (a) thru (e) and Figure 78-30-42.
    A. Inspect leading edges of ramps on both racks for burrs and chipping of the spray coating. Replace if damaged.

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MAINTENANCE PRACTICES (Continued)

(b) Inspect rack teeth for heavy wear patterns, chipping and excessive wear. Replace if any one of these conditions exist.

(c) Inspect inserts (P/N MS521209 C0610 and P/N MS21269C0615) for damage and security. Replace as required.

(d) Check weld at base of pin for cracks. Check pins for distortion or outer surface damage. If damaged, replace rack assembly.

(e) Check pin P/N MS948650 on end of racks for wear and looseness. Replace rack as required.

(11) Inspect translating beams per steps (a) thru (e). (See figure 78-30-43 for corresponding steps to inspection areas.)

(a) Inspect beam surfaces for breaks in the hard coat (surface) areas as per figure 78-30-43. Breaks in hard coat areas will be visible by galling. Replace beam if galling is present.

(b) Check beam area where the yoke base travels. A total width of this area must not exceed 0.375 ±0.010 in. Replace beam if limits are exceeded.

(c) Inspect dimensions of beams per figure 78-30-43 (views A-A and C-C). Surfaces in the vertical plane and surfaces within the horizontal plane must be parallel to other bearing surfaces in the same plane within 0.001 in. Replace beams if dimensions are out of limits.

(d) Check beam ramps, guides and stops for wear and security. Replace any loose or damaged inserts (P/N MS21209-CO615). Ramp hardware may be secured by application of Nylock VC-3 or Loctite to threads.

(e) Perform the Dimensional Requirements of Replacing Pneumatic Latch(es) per Chapter 78.

(12) Inspect tail pipe seal for cuts, tears and general condition. Refer to "Tailpipe Seal Repairs," per Sabreliner Maintenance Manual.

(13) Perform the "Functional Check of Thrust Reverser Sequencing Latches" per Sabreliner Maintenance Manual.

(14) Lubricate thrust reversers. Refer to "Servicing Thrust Reverser System — Lubrication and Cleaning" in Sabreliner Maintenance Manual.

(15) Check complete thrust reverser assembly for component security, proper lock wire and lock pin installation, proper electrical cable routing, pneumatic line security, chafing between component parts and the freedom of all foreign objects.

(16) Install aft nacelle fairing (scoop) per Chapter 78.

(17) Run complete operational check per Chapter 78.

(18) Engine alignment must be verified before installation of aft nacelle fairing (thrust reverser assembly). Refer to "Aligning Engine" in Aeronca maintenance manual.

(19) Install aft engine fairing per Chapter 78.

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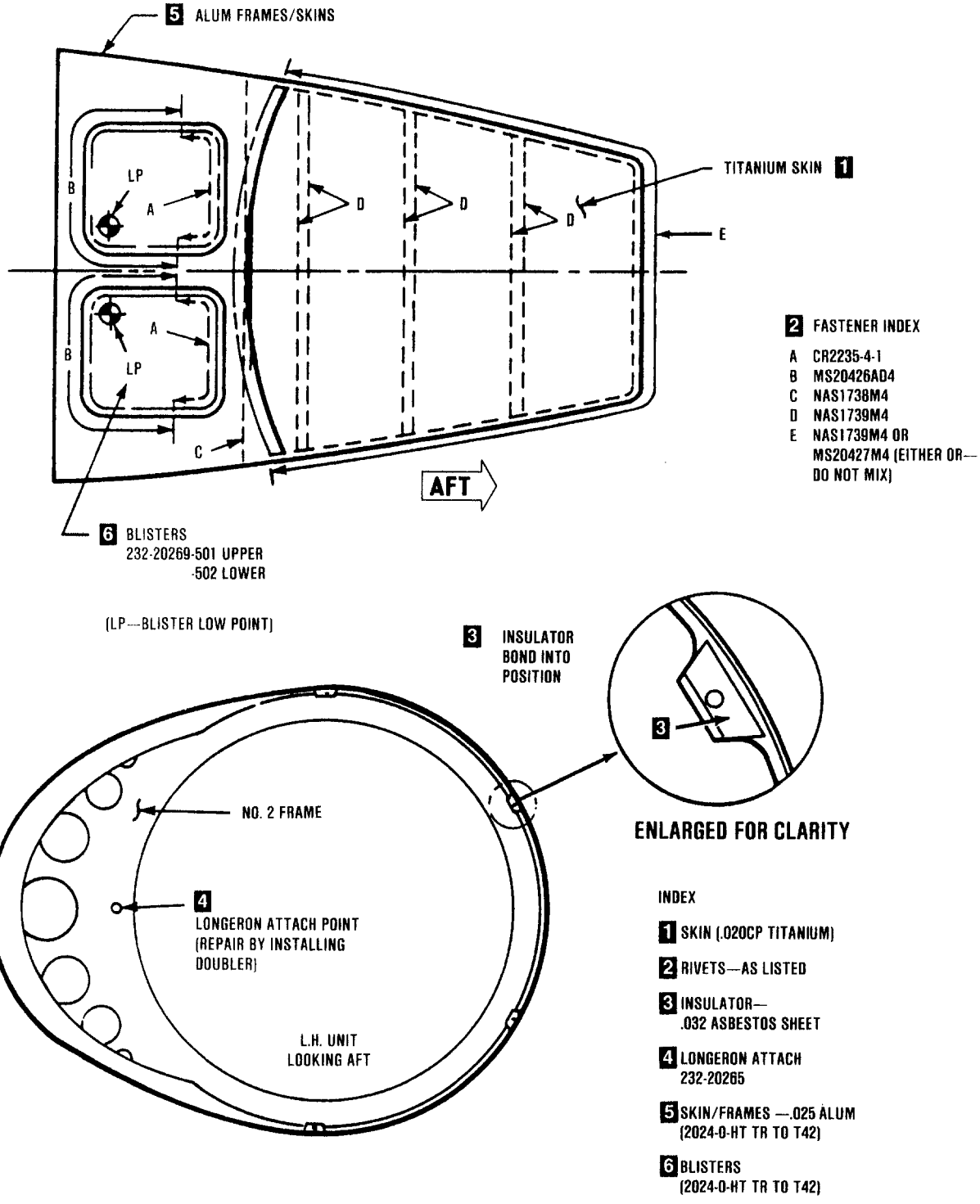

NEW

Aft Nacelle Fairing Component Location
Figure 78-30-39

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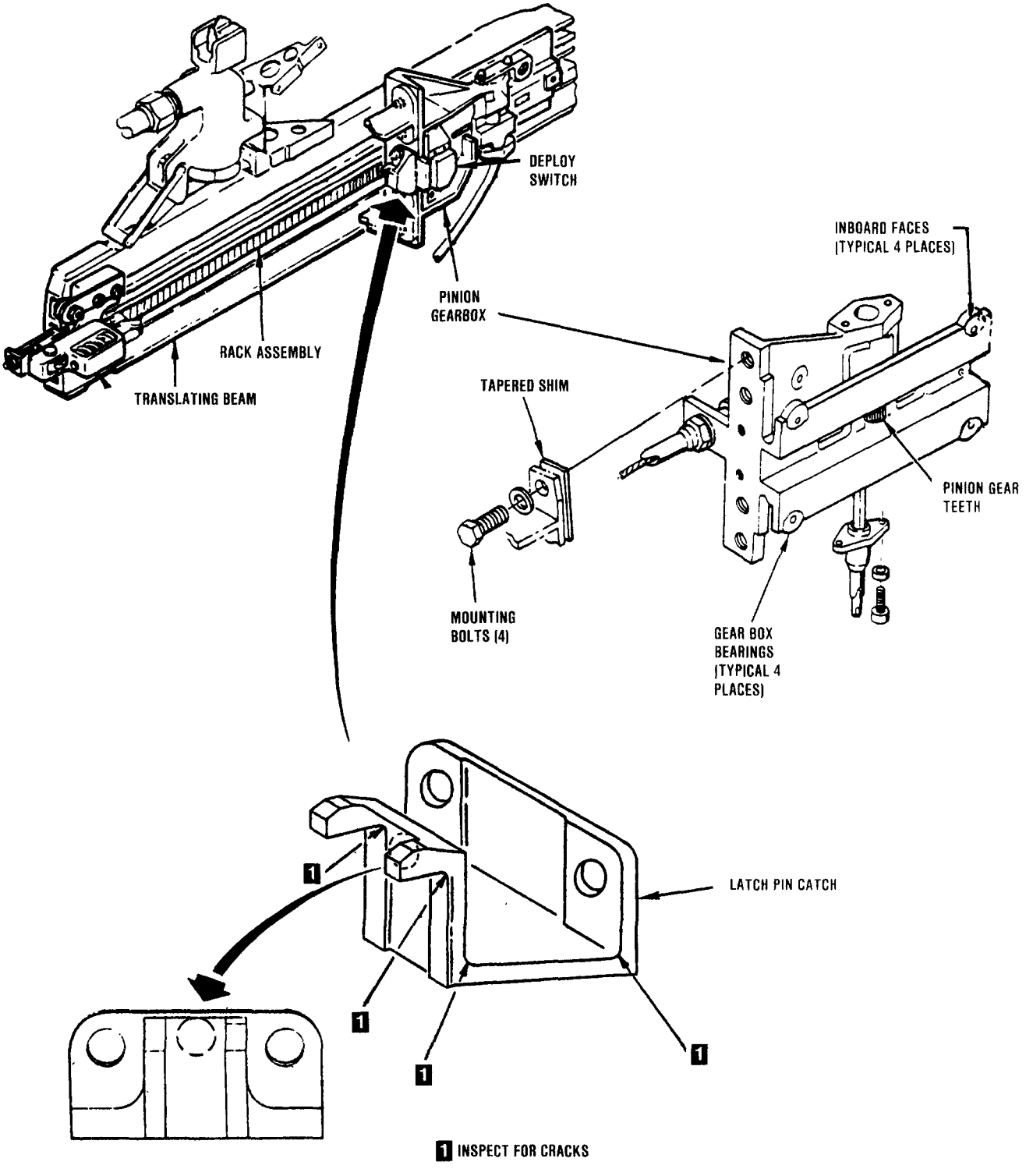

NEW

Pinion Gear Box Component Location
Figure 78-30-40

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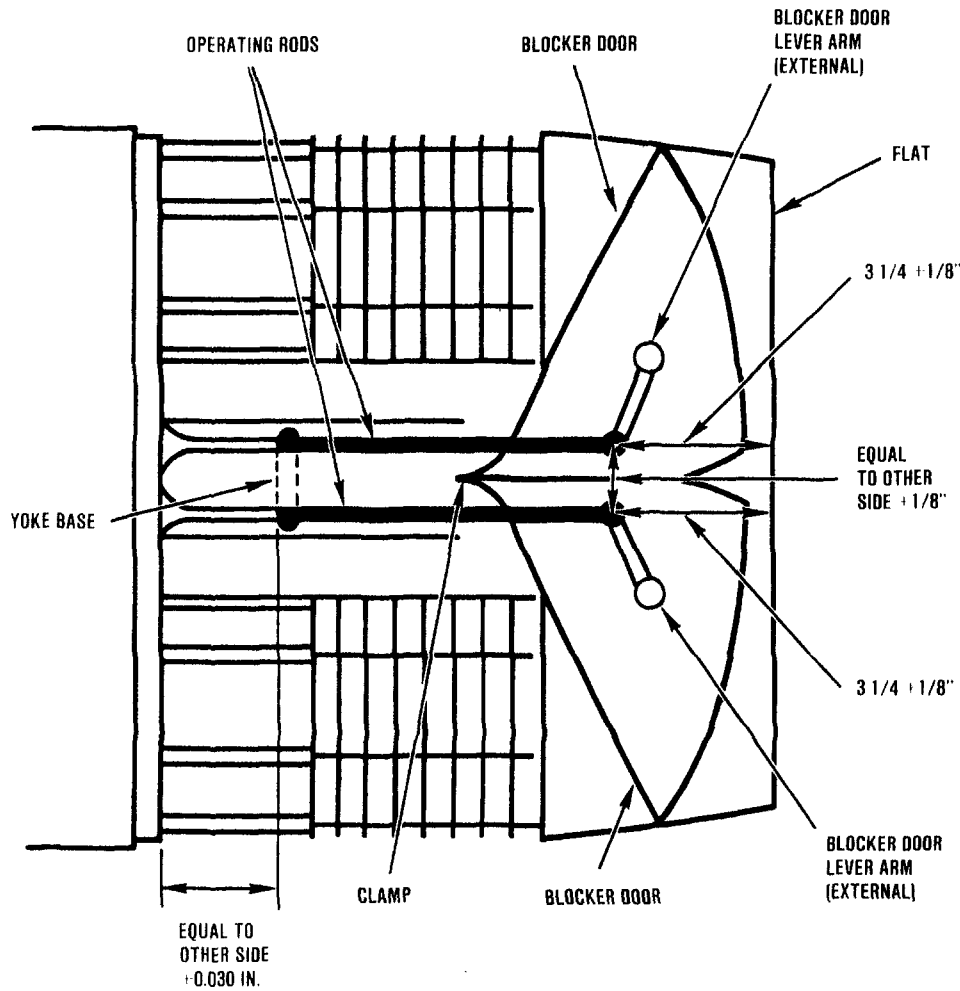

NOTE

ALL DIMENSIONS ARE WITH THRUST REVERSER DEPLOYED AND CLAMPED

NEW

[80] 78-30-5

Translating Structure Rigging Dimensions
Figure 78-30-41

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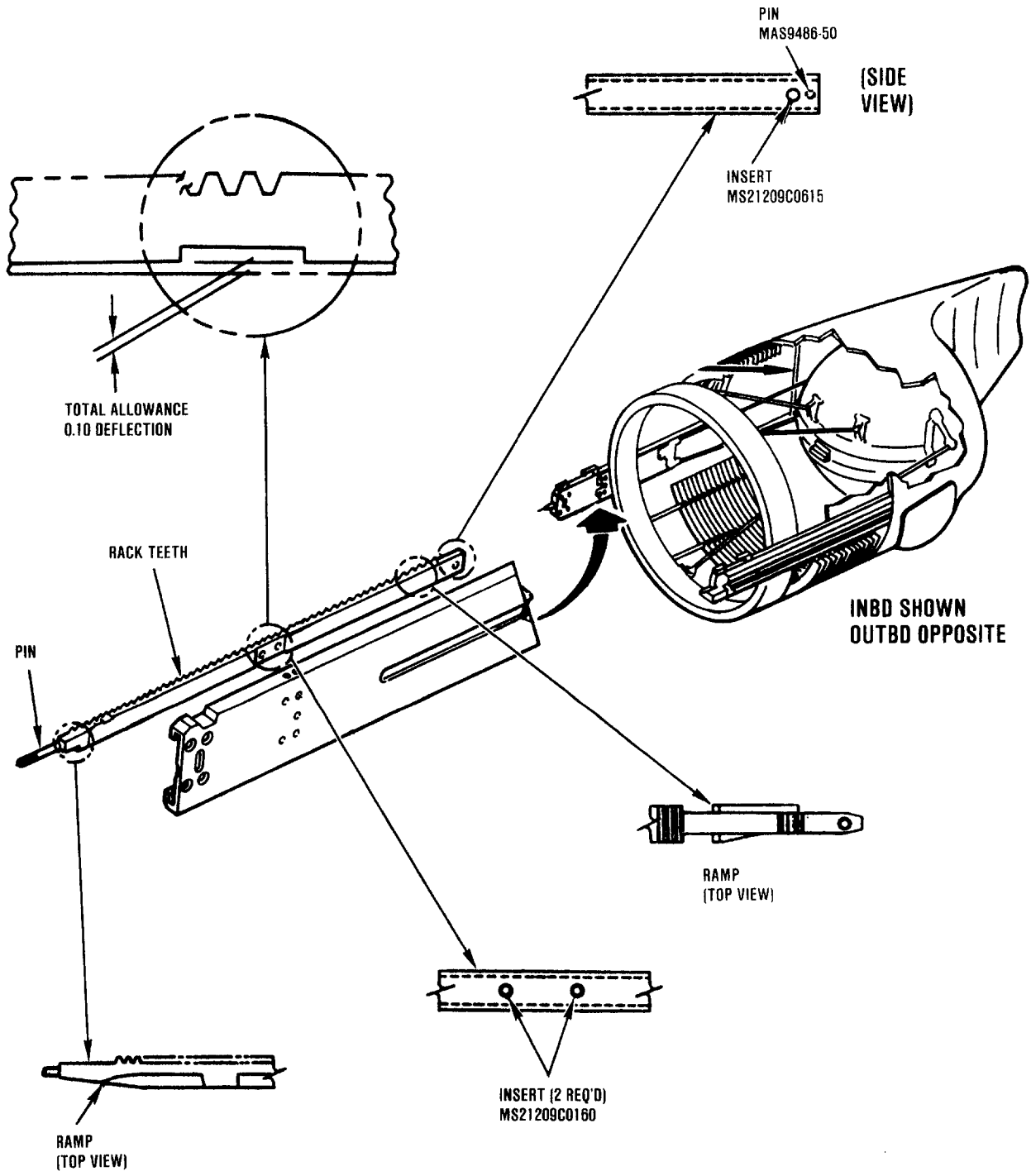

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Gear Rack Assembly Component Locations
Figure 78-30-42

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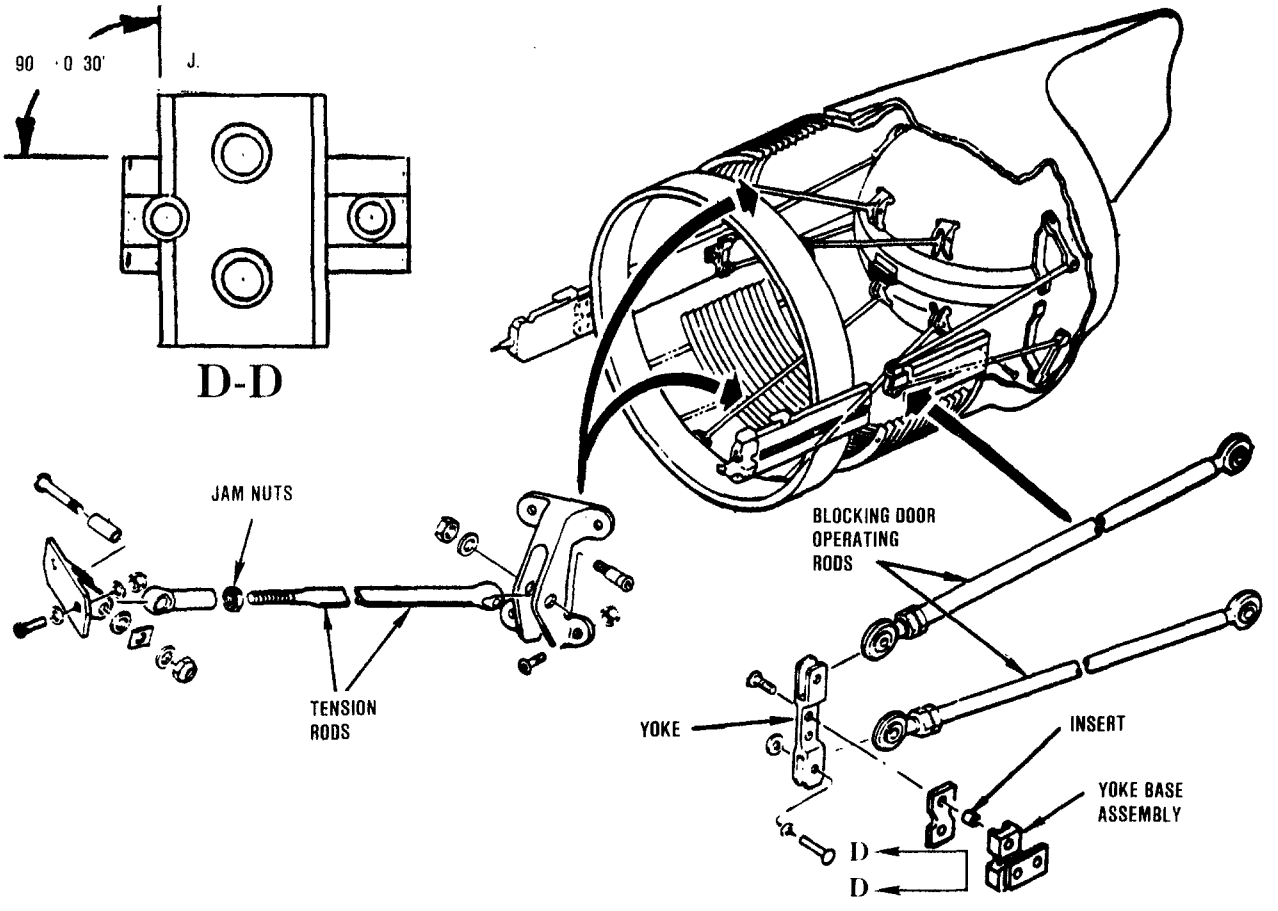
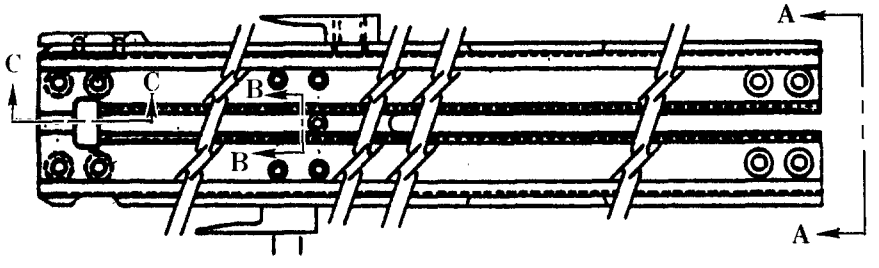
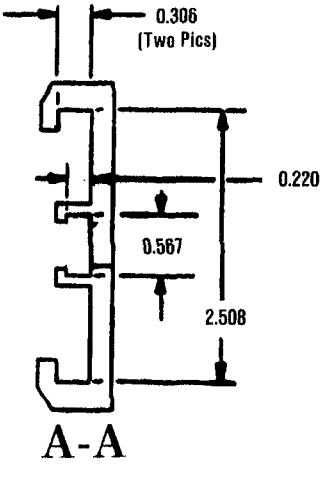


BEAM INSPECTION





NOTE

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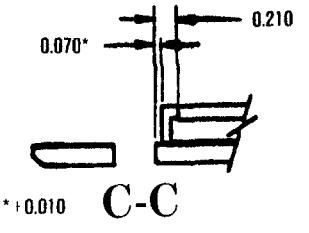

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[80] 78-30-5

Translating Beam Inspection
Figure 78-30-43

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CHAPTER

79

OIL

CHAPTER 79

OIL

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CHAPTER 79

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OIL - DESCRIPTION AND OPERATION

1. General

The oil system is an integral part of the engine. For information regarding this system, refer to Pratt & Whitney Engine Maintenance Manual, PWA 435107, on airplanes NA-370-2 through -6, -8 and -9. For airplanes NA-370-1, -7, NA-380-1 and later, the oil system description is provided in the General Electric CF700 Maintenance Manual, SEI-187.

OIL - MAINTENANCE PRACTICES

1. Checking Engine Oil Level

(Refer to Chapter 12 for engine oil servicing.)

2. Engine Oil Change - Aircraft NA-370-2 Through -6, -8 and -9

A. To change engine oil, proceed as follows:

(1) Remove lower engine access door.

(2) Remove drain plugs to drain oil tank and accessory drive case.

(3) Replace drain plugs using new packings when oil has drained. Torque plugs to 120-130 inch-pounds. Safety plug with lockwire.

(4) Fill oil tank with 5 quarts of approved oil.

(5) Perform engine run.

(6) Service engine oil as necessary.

(7) Install lower engine access door.

3. Engine Oil Change - Aircraft NA-370-1, -7, NA-380-1 and Later

A. To change engine oil, proceed as follows:

(1) Open engine access doors as required.

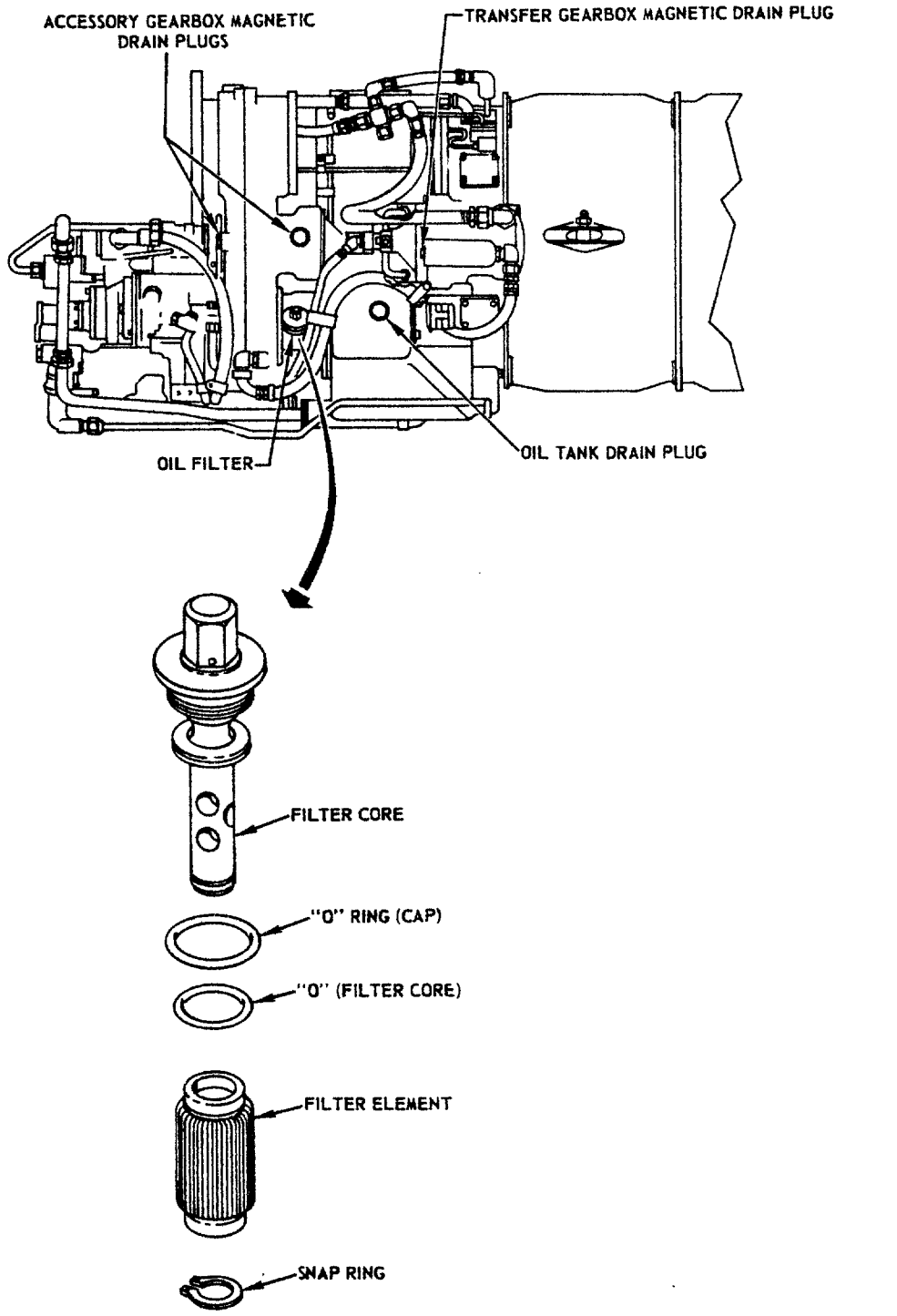
(2) Remove safety wire from oil tank filler cap and remove filler cap.

(3) Drain all possible oil from the engine oil system by removing drain plugs from the bottom of the oil tank, accessory gearbox and transfer gearbox. Drain oil into a clean container capable of holding at least four quarts.

(4) Remove and inspect main oil filter.

(5) Inspect the magnetic drain plugs for metallic particles. If magnetic particles are found, refer to procedures in "Removing and Inspecting Main Oil Filter - Aircraft NA-370-1, -7, NA-380-1 and Later".

(6) If magnetic particles are not found, install a new oil filter, using a new "o" ring. Torque filter cap to 40-60 inch-pounds and safety with lockwire.



Oil System - Drains and Filter Locations -
Aircraft NA-370-1, -7, NA-380-1 and Later
Figure 79-00-1

(7) Clean drain plugs in a suitable solvent, such as trichloroethylene, Stoddard Solvent (Federal Specification P-D-680) or equivalent.

WARNING: PERFORM THIS OPERATION IN A WELL VENTILATED AREA. TAKE CAUTION TO PREVENT INHALATION OF VAPORS AND TO MINIMIZE FIRE HAZARDS. REPEATED OR PROLONGED CONTACT OF SOME CLEANING MATERIALS WITH THE SKIN WILL REMOVE SKIN OILS AND MAY CAUSE SEVERE DERMATITIS.

(8) Reinstall all drain plugs using new "o" rings. Torque all drain plugs to 60-80 inch-pounds.

(9) Using same brand-and-type of oil as drained from system, add four quarts of lubricating oil to the engine-mounted oil tank.

(10) Reinstall engine oil tank filler cap and safety with lockwire.

NOTE: The engine will require some additional oil after the first run to bring the oil level to the FULL mark on the dipstick.

- Changing brands of oil within a type category or changing types of oil will require that the system be drained, filled, flushed, and refilled with the new brand or type of oil.
- Indiscriminate mixing of brands and/or types of oils must be avoided (refer to Servicing in Chapter 12 for a listing of approved engine lubricating oils).
- Engine oil consumption shall not exceed 0.085 gallons per hour (approximately three-fourths of a pint).

4. Removing and Inspecting the Main Oil Filter - (Type D, Disposable) - Aircraft NA-370-1, -7, NA-380-1 and Later

A. To remove the main oil filter, proceed as follows:

NOTE: To gain access to the engine oil filter it will be necessary to remove the engine lower inboard and outboard access doors.

(1) Remove lockwire securing oil filter cap to engine lubricating pump.

(2) Unscrew filter cap from the engine lubricating pump. Use a small container (about one pint capacity) to catch the small amount of oil that will drain from the filter housing in the pump. Discard the "o" ring sealing the filter cap to the pump housing.

(3) Disassemble the filter assembly by removing the snap ring at the base of the filter and sliding the filter element off the filter core.

(4) Inspect the filter element for accumulation of metal chips, synthetic rubber, lint, etc. Following inspection, discard filter.

(5) If no accumulation is found, the engine will be considered acceptable. Proceed as follows:

(a) Clean filter core by immersing in trichloroethylene, Stoddard Solvent (Federal Specification P-D-680) or equivalent. Swirl the filter core rapidly in the cleaning solution.

(b) Install a new filter element onto the filter core and secure in place using a new snap ring.

(c) Using new "o" rings install the filter assembly in engine lubrication pump housing. Torque filter cap to 40-60 inch-pounds and safety with lockwire.

NOTE: If the amount of oil lost during filter removal is less than one pint, and if the filter element is free of contaminants, complete draining and refilling of the engine oil system is not required; however, the oil level should be checked. (Refer to Chapter 12.)

(6) If accumulation is found and is greater than 10% of the total filter area, DO NOT operate the engine until the source of the particles is defined and corrected. If necessary, consult the engine manufacturer's representative for recommendations.

(7) If accumulation is found, but is less than 10% of the total filter element area, determine if the contamination (either in the oil or the filter) is metallic. If metallic particles are present, proceed as follows:

NOTE: A light deposit of minute silver particles may be found on the filter during the early stages of engine operation and is considered normal. Silver particles can be from normal seating of the engine bearing cages that have been silver plated during manufacture.

(a) Drain all oil from the engine, oil tank and gearboxes.

(b) Clean the filter core and drain plugs by immersing in trichloroethylene or Stoddard Solvent (Federal Specification P-D-680) or equivalent. Dry parts thoroughly.

(c) Reinstall and torque all drain plugs to 60-80 inch-pounds, then safety with lockwire.

(d) Install a new filter element on filter core using new "o" rings and snap ring. Safety with lockwire.

(e) Service the engine oil tank to the FULL mark on the dipstick using the same brand and type of oil as drained from the engine.

(f) Start and operate the engine for a minimum of 10 minutes at 85% rpm. Shutdown the engine. Remove and check the oil filter.

(g) If the filter is still significantly contaminated, DO NOT operate the engine until the cause of the contamination can be determined. Consult with the engine manufacturer's representative for appropriate action.

(h) If the filter is significantly cleaner, clean the filter core and install a new filter element, drain the oil from the engine, oil tank, and gearboxes, then reservice the oil tank to the FULL mark on the dipstick with new oil using the same brand and type as drained from engine.

(i) Start and operate the engine for two hours at various thrust levels including a total of 5 minutes at take-off thrust. During this time period, monitor oil pressure and oil temperature for any significant changes. Shut down the engine.

(j) Remove the main oil filter from the engine lubricating pump and examine for contaminants. If contaminants in excess of minute quantities are found, DO NOT operate the engine nor throw away the filter element until the cause of the contamination can be found.

(k) If no contaminants are found, reinstall the filter and operate the engine as required. Inspect the main oil filter every five hours of engine operation for a period of twenty hours. Record the main oil filter inspection findings. Normal filter inspections can be resumed when the filter element does not reveal the generation of any additional contaminants.

5. Flushing Engine Lubricating System - Aircraft NA-370-1, -7, NA-380-1 and Later

A. The following procedure is to be used if oil brands or types have been mixed, or if an unapproved oil has been used. Following the draining, the engine lubrication system must be flushed with the same brand and type of oil with which the engine is to be serviced. (Refer to Chapter 12 for listing of approved engine oils.) To flush the engine oil system, proceed as follows:

(1) Remove lockwire from engine oil tank filler cap and remove cap.

(2) Remove drain plugs from oil tank, accessory gearbox and transfer gearbox. Catch the draining oil in a clean container capable of holding at least four quarts.

(3) Remove the main oil filter from the engine lubricating pump housing. Inspect the oil filter element, drain plugs, and drained oil for magnetic particles. If magnetic particles are present, accomplish appropriate procedures in "Removing and Inspecting Main Oil Filter - Aircraft NA-370-1, -7, NA-380-1 and Later".

(4) If magnetic particles are not present, clean the filter core and drain plugs using a suitable solvent such as trichloroethylene, Stoddard Solvent (Federal Specification P-D-680), kerosene or equivalent.

WARNING: PERFORM THE CLEANING OPERATION IN A WELL VENTILATED AREA. TAKE PRECAUTION TO PREVENT INHALATION OF VAPORS AND TO MINIMIZE FIRE HAZARDS. REPEATED OR PROLONGED CONTACT OF SOME CLEANING MATERIALS WITH THE SKIN WILL REMOVE SKIN OILS AND MAY CAUSE SEVERE DERMATITIS.

(5) Install a new filter element on filter core using new "o" rings and snap ring.

(6) Install main oil filter and all drain plugs using new "o" rings. Torque oil filter to 40-60 inch-pounds and drain plugs to 60-80 inch-pounds.

(7) Fill the oil tank to FULL on the dipstick (approximately 4 quarts) using the same brand and type of oil that the engine will be serviced with. Install and lock filler cap.

(8) Start and operate engine at IDLE for three minutes, and at 75%-80% rpm for two minutes, then shut down engine.

(9) Drain oil from engine by removing the main oil filter and drain plugs from the oil tank, accessory gearbox and the transfer gearbox.

(10) After draining is completed, install all drain plugs using new "o" rings. Torque all drain plugs to 60-80 inch-pounds.

(11) Replace filter element using new "o" rings and snap ring.

(12) Install main oil filter using new "o" ring. Torque filter to 40-60 inch-pounds and safety with lockwire.

(13) Fill oil tank to FULL mark on dipstick with the same brand and type of oil that was used in the flushing procedure. Attach a metal tag or other suitable means of identifying the brand and type of oil used to oil tank filler cap.

(14) Install and lock oil tank filler cap. Safety with lockwire.

NOTE: The engine will require some additional oil after the first run to bring the oil level to the FULL mark on the dipstick because a small amount of oil will remain in the sumps and oil lines after shutdown. (Refer to Chapter 12 for servicing instructions.)

- After the engine has been operated in service for 15-35 hours, check the main oil filter, and replace if necessary.

CHAPTER

80

STARTING

NOTICE

The following information applies to NA 265-70 airplanes only.

CHAPTER 80

STARTING

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CHAPTER 80

STARTING

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*The asterisk indicates pages revised, added, or deleted by the current revision.

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STARTING - DESCRIPTION

1. General

The engine starting system is used to crank the engines before ignition and to motor the engines during maintenance. It also provides assistance after ignition until the engine rpm is sufficient to maintain self-sustained operation. Ignition is initiated by moving the throttle to the idle position. A red indicator light in the knob of the engine start switch indicates ignition system operation.

ENGINE STARTING SYSTEM - DESCRIPTION AND OPERATION

1. General

The engine starting system supplies dc power to the engine-mounted starter-generator for cranking the engine during engine starts. Provisions are included for making starts using an external power source, airplane batteries, and in-flight air starts. External power starts and battery starts are controlled by the corresponding engine start switch. (See figure 80-10-1). In-flight air starts are controlled by the "AIRSTART" switch on the left engine control panel. The starting cycle is controlled by the start controller relay after the starting sequence has been initiated. The starting cycle can be interrupted at any time by manually pulling up on the engine start switch. After the starting cycle has been completed, the starter-generator functions as a dc generator.

When the airplane batteries are used for engine start, the two 24-volt dc batteries are connected in series to provide a nominal 48-volt dc starting power.

To operate the engine starting system, power must be available from the starter bus, essential bus, and paralleling bus. The starter bus supplies power to the starter-generator. The essential bus supplies power to the starting system control circuitry.

When the external power source is connected to the external power start receptacle, the external start power monitor relay is energized from the paralleling bus through a switch actuated by the external power disconnect plug. The energized external power monitor relay opens the circuit to the battery start relay, preventing the batteries from energizing the starter bus. (See figure 80-10-2.)

NOTE: When an external power unit is used for starting the engines, the unit must be capable of providing not less than 24-volt dc and 800 amperes, and not more than 30-volt dc and 1200 amperes during the starting cycle.

When the airplane batteries are used for starting the engine, the battery start relay is energized through the engine starting system control circuitry. The energized battery start relay connects the batteries in series and connects this circuit to the starter bus.

NOTE: Only the right engine starting system is described. The left engine starting system is similar.

- Operation of the engine starter must be limited to a total of 120 seconds in any 20-minute period.

With the right engine master switch at ON, power from the essential bus is supplied to the right fuel shutoff valve (opening the valve), the right engine start switch (terminal No. 1), and the right fuel boost pump relay through the normally open contacts of the throttle-actuated fuel-ignition switch. The energized fuel boost pump relay opens the circuit to the fuel boost pump as long as the throttle is in closed position.

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Depressing the right engine start switch simultaneously energizes the left start-stop relay, the right start power monitor relay, the right generator field start relay, and the right ground start relay. The energized left start-stop relay opens a circuit which prevents operation of the left engine starting system while the right engine starting system is energized.

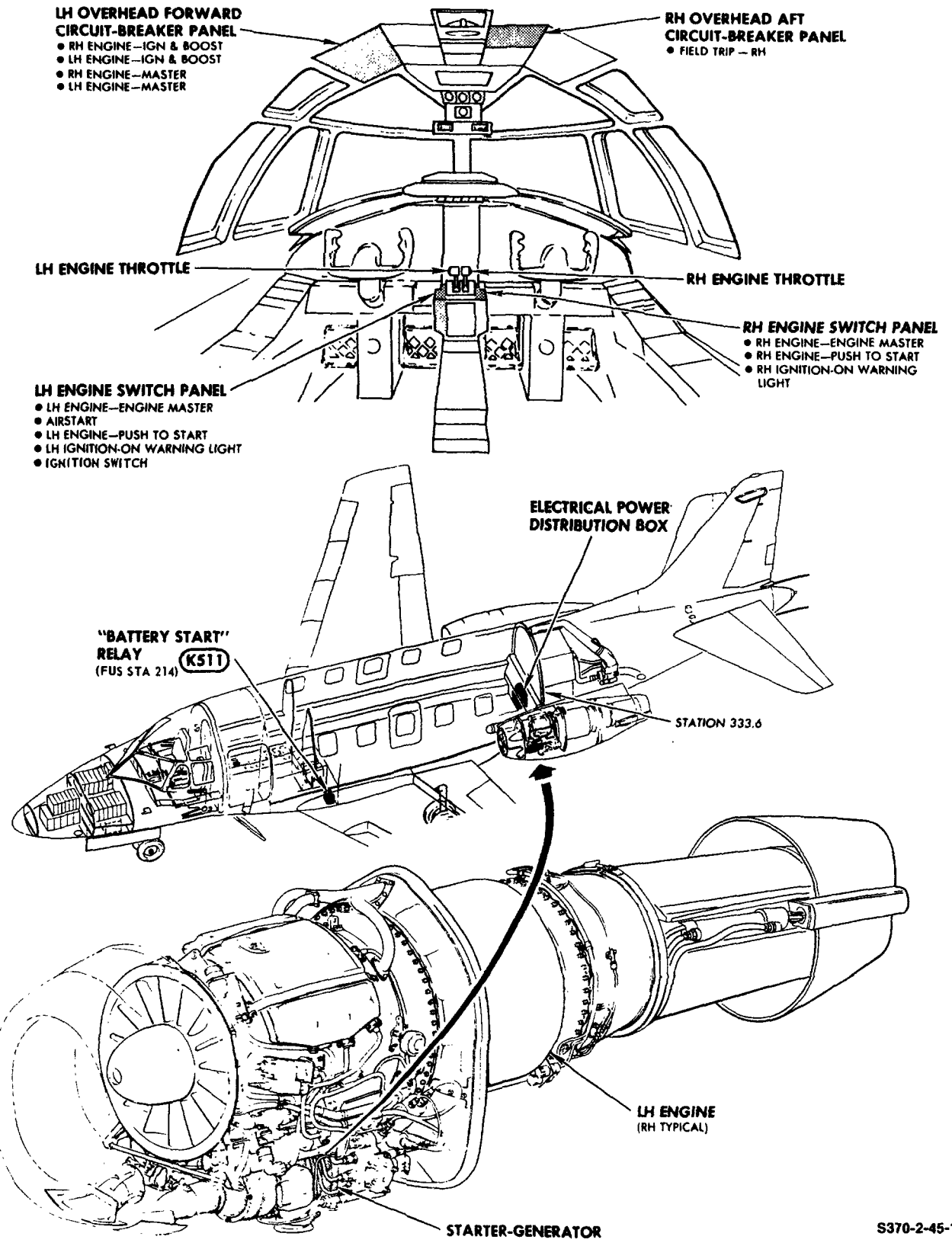

Figure 80-10-1 — Engine Starting System Unit Location

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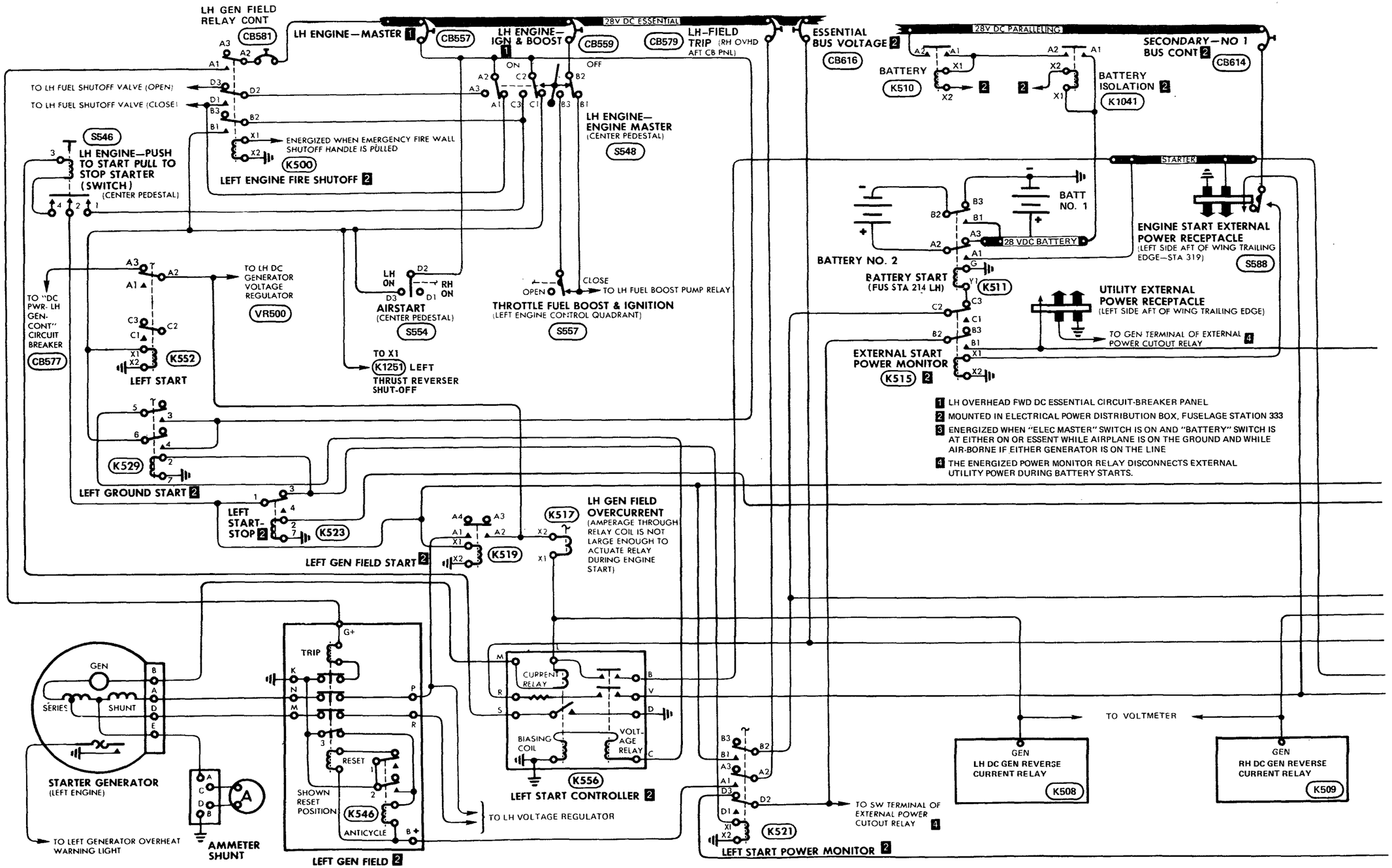

Figure 80-10-2 — Engine Starting System Electrical Schematic (Sheet 1 of 2)

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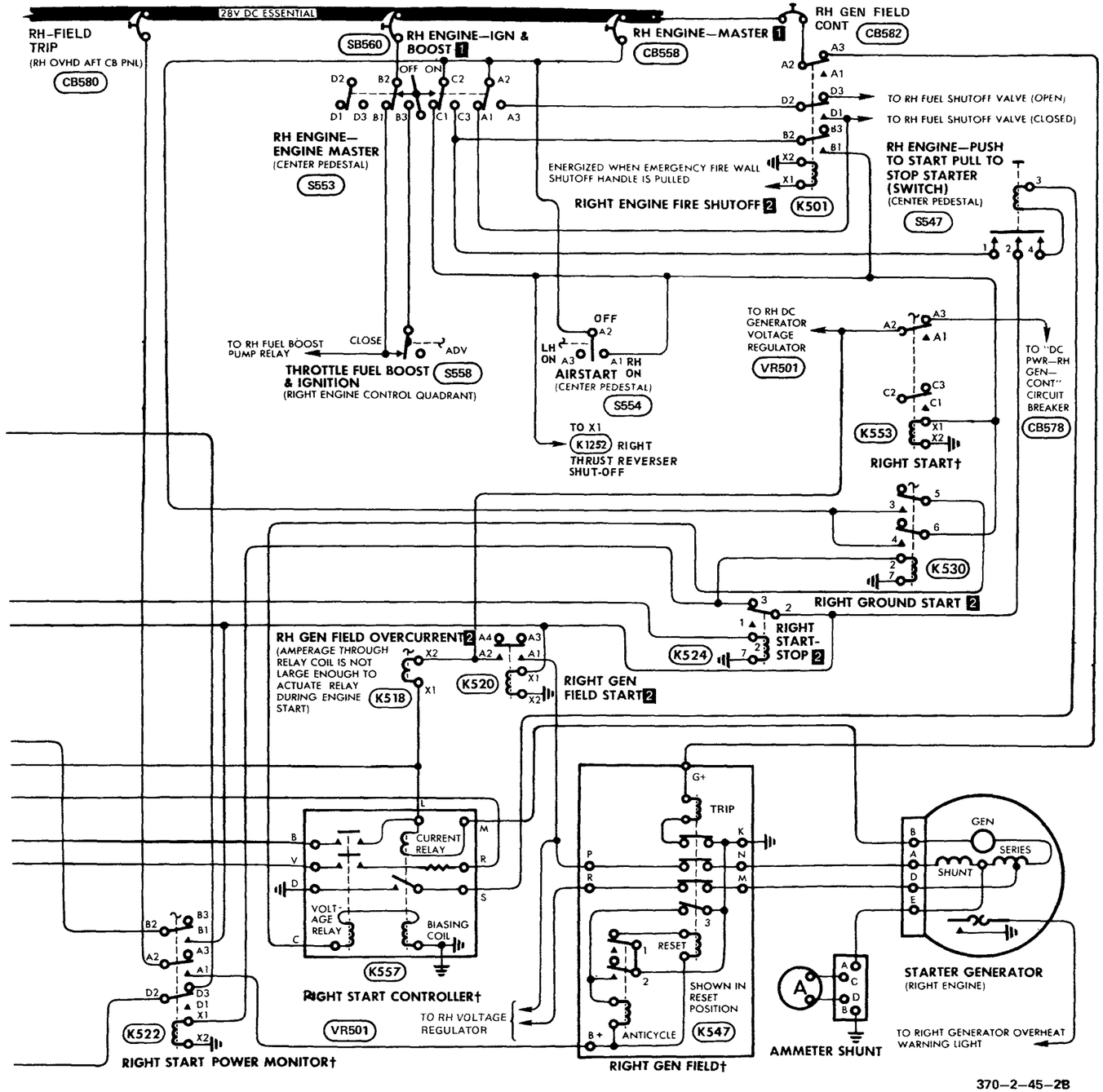

Figure 80-10-2 — Engine Starting System Electrical Schematic (Sheet 2 of 2)

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DESCRIPTION AND OPERATION (Continued)

31. GENERAL (Cont'd)

NOTE

The generator overcurrent relay contacts will not close during the starting cycle because the current flow through the low-resistance coil is not high enough. Normally, about 35 amps are required to actuate this relay.

When engine rpm reaches 10%, the throttle should be moved to idle. This opens the circuit through the throttle-actuated fuel ignition switch to the right fuel boost pump relay. The relay closes and turns on the fuel boost pump. With the throttle at idle position, a circuit is also compelted to the engine ignition system.

As the engine rpm increases, the amperage flow through the current coil of the start controller (between terminals "L" and "M") decreases from the initial 800 amps to about 200 amps when the engine rpm reaches about 35%. At this point, the current flow through the current coil in the start controller is not high enough to hold the engine start switch down. The start switch pops up, opening the switch contacts, which de-energizes the engine starting cycle.

When the engine rpm has accelerated to about 40%, the starter-generator begins to function as a generator.

NOTE

For information regarding air start of these engines, refer to chapter 74.

A split-bus (soft-start) type auxiliary power unit (APU) is recommended for starting the engine with an external power source. The soft-start type APU supplies 28 VDC constant-voltage power to the paralleling bus, while at the same time supplying 28 VDC, 1000-amp constant current to the starter bus. A signal from the paralleling bus is supplied to the APU control circuitry which splits the power output when the start controller is energized.

The soft-start type APU is equipped with one oval plug and one rectangular plug. The oval plug is inserted into the external power utility receptacle (oval-shelled) at left FS 319, which supplies power to the paralleling bus. The rectangular plug is inserted in the external power start receptacle (rectangular-shelled) at left FS 319, which supplies power to the starter bus.

32. STARTER-GENERATOR

A starter-generator is installed on the accessory drive of each engine. The starter-generator is a 400-amp, series-shunt-wound, Bendix combination starter and generator. This dual-purpose unit cranks the engine during the starting operation when the proper DC power source is connected, and it supplies DC power to the airplane DC electrical systems after the engine rpm has accelerated to about 40%. During the starting operation, both the series and shunt winding are energized to obtain maximum starter rpm.

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The starter-generator is equipped with a QAD (quick attach disconnect) type mounting. During installation, great care must be taken to ensure that pins on the QAD are engaged in the proper holes to prevent rotation of the starter-generator body within its attaching clamp ring when torque is developed during the starting operation.

3. Engine Start Switch

An engine start switch is provided for each engine starting system. The start switch controls power from the essential bus to the generator field start relay, the opposite starting system start-stop relay, the ground start relay (through the normally closed contacts of the start-stop relay), and terminal B1 of the start power monitor relay.

4. Start Controller Relays

A start controller relay is installed in the left and right engine starting systems. These relays are mounted in the electrical power distribution box at fuselage station 333. The start controller controls the engine starting cycle, and supplies a signal from the paralleling bus to the soft-start type auxiliary power unit. The start controller consists of a voltage relay, a biasing coil connected in series with the voltage relay coil, and a current relay. (See figure 80-10-2.)

5. Start Power Monitor Relays

A start power monitor relay is installed in the left and right engine starting systems. These relays are mounted in the electrical power distribution box at fuselage station 333. (See figure 80-10-1.) The start power monitor relays control power to the generator field control relay, battery start relay, external power cutout relay, and engine ignition system.

6. Ground Start Relays

A ground start relay is installed in the left and right engine starting systems. These relays are mounted in the electrical power distribution box at fuselage station 333. (See figure 80-10-1.) The ground start relay controls power to the start controller and to the start relay during engine starts. This relay is energized with power from the essential bus when the engine start switch is depressed.

7. Start-Stop Relays

A start-stop relay is installed in the left and right engine starting systems. These relays are mounted in the electrical power distribution box at fuselage station 333. The start-stop relay controls power to the ground start relay and the start power monitor relay. This relay is energized by the opposite engine starting system circuitry to prevent operation of both starting systems at the same time.

8. Start Relays

A start relay is installed in the left and right engine starting systems. These relays are mounted in the electrical power distribution box at fuselage station 333. The start relay is energized when the engine master switch is at OFF and during all starting operations when the engine master switch is at ON. (See figure 80-10-2.)

9. Generator Field Start Relays

A generator field start relay is installed in the left and right engine starting systems. These relays are mounted in the electrical power distribution box at fuselage station 333. The generator field start relay is energized from the essential bus through the engine start switch during normal external power starts and battery starts. (See figure 80-10-2.)

10. Generator Field Control Relays

A generator field control relay is installed in the left and right engine starting systems. These relays are mounted in the electrical power distribution box at fuselage station 333. This relay is utilized in the starting and the generator systems. For starting purposes, the generator field control relay controls the circuit to the starter-generator shunt field.

11. External Power Start Relay

The external power start relay controls power to the battery start relay. This relay is mounted in the electrical power distribution box at fuselage station 333. This relay is energized from the paralleling bus through a switch which is actuated when the auxiliary power unit plug is inserted into the rectangular external power receptacle. (See figure 80-10-2.)

12. Battery Start Relay

The battery start relay controls power from the batteries to the battery bus and starter bus. When de-energized, this relay connects both batteries in parallel to the battery bus. When energized, the battery start relay connects the batteries in series and connects this circuit to the starter bus to provide power to the starter-generator. The battery start relay is energized by power from the essential bus through the engine start switch, the energized start power monitor relay, and the de-energized external start power monitor relay during an engine start using battery power. Also, during a battery start, power for the engine ignition system and the fuel boost pumps is supplied by battery No. 1 through the battery start relay. The battery start relay is installed at fuselage station 214.

ENGINE STARTING SYSTEM - TROUBLE SHOOTING

1. Trouble Shooting Engine Starting System

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. ENGINE WILL NOT COME UP TO STARTING RPM - EXTERNAL POWER.		
(1) Power output of external power unit insufficient.	Check output of power unit: minimum, 24-volt dc and 800 amperes; maximum, 30-volt dc and 1200 amperes.	Readjust external power unit output to 800 amperes or replace power unit with an equivalent unit.
(2) No power to starter shunt field. Generator field start relay is not reset.	Check position of emergency fire wall shutoff handle.	If emergency fire wall shutoff handle is pulled, push handle in.
	Check for continuity between terminals "P" and "N" of generator field control relay. If there is no continuity between these terminals, check for 28-volt dc at terminal "B+" (reset coil) with starting system energized. If there is power at terminal "B+" of generator field control relay, check for power at terminal "G+" (trip coil). There should be no power at this terminal during engine start operation.	If there is power at terminal "G " of generator field control relay, repair generator circuit. (Refer to Chapter 24.)
(3) No power to starter shunt field. Defective generator field start relay or ground connection.	With starting system energized, check for 28-volt dc between X1 and ground. If power is available at X1, check for power at X2 of generator field start relay.	Repair ground connection and/or replace relay, as necessary.
B. ENGINE WILL NOT COME UP TO STARTING RPM - BATTERY POWER.		
(1) Batteries are not connected in series.	Check battery power supply circuit from batteries to starter bus.	Repair defective circuit or replace battery start relay as necessary.
(2) Batteries not fully charged.	Check charge on batteries. (Refer to Chapter 12.)	Recharge batteries.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
(3) Starter shunt field is not energized. Generator field control relay is not reset.	Refer to isolation procedure for probable cause (2) under trouble A.	
(4) Defective generator field start relay.	Refer to isolation procedure for probable cause (3) under trouble A.	
(5) Defective power supply circuit from starter bus to shunt field of starter-generator.	Check power supply circuit.	Repair defective circuit as necessary.
C. STARTER WILL NOT CRANK ENGINE.		
(1) Starter power leads loose or not connected to starter terminals.	Remove power. Visually check power lead connections.	Connect and/or tighten as required.
(2) Essential bus is not energized.	Refer to Chapter 24.	Energize essential bus.
(3) Starter drive shaft sheared.	Start switch will not stay down when depressed. Starter can be heard running when start switch is depressed.	Replace starter-generator.
(4) Defective start switch assembly.	Depress start switch and check for 28-volt dc at terminals "1" and "2." If power is available at terminal "1" but not at terminal "2," switch is defective.	Replace start switch assembly.
(5) Defective ground start relay.	Check power circuits to ground start relay.	Replace relay.
(6) Defective starter controller.	Check power circuit to the starter controller relay.	Replace starter controller.
D. ENGINE START SWITCH WILL NOT STAY DOWN WHEN DEPRESSED DURING STARTING CYCLE.		
(1) Defective engine start switch holding coil.	Check continuity between terminals "3" and "4" of engine start switch assembly.	Replace engine start switch assembly.

1. Replacement of Starting System Relays

NOTE: Refer to Chapter 24 for removal and installation of starter-generator.

The starting system relays are all mounted in the electrical power distribution box at fuselage station 333, except the battery start relay, which is mounted at fuselage station 214. To gain access to relays, remove lower trim panel from aft bulkhead in cabin compartment; then loosen fasteners securing close-out cover on electrical power distribution box and swing cover forward and down.

IGNITING

1. General

For information covering the igniting phase of the engine starting system, refer to Chapter 74.

NOTICE

The following information applies to NA 265-80 airplanes only.

CHAPTER 80

STARTING

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CHAPTER 80

STARTING

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ENGINE STARTING SYSTEM

DESCRIPTION AND OPERATION

GENERAL

The engine starting system supplies DC power to the engine-mounted starter-generator for cranking the engine during engine starts. Provisions are included for making starts using an external power source, airplane batteries, and in-flight air starts. External starts and battery starts are controlled by corresponding engine start switch. (See Figure 80-00-1.) In-flight air starts are controlled by the AIR START switch on the left engine control panel. The starting cycle is controlled by the DC generator control unit after the starting sequence has been initiated. The starting cycle can be interrupted at any time by manually pulling up on the engine start switch. After the starting cycle has been completed, the starter-generator functions as a DC generator.

OPERATION — S/N 380-1 THRU -72 NOT CHANGED BY SOK 17

To operate the engine starting system, power must be available from the starter bus and essential bus. The starter bus supplies the power to the starter only and the essential bus powers the starter control circuitry.

When external power is used, a switch in the external power start receptacle is actuated completing a circuit from the essential bus to the coil of the external power monitor relay. The energized external power monitor relay opens a circuit to the battery start relay, isolating the batteries from the starting circuit (refer to DC bus tie-in system).

When the batteries are used for starting, the battery start relay is energized which connects the batteries in series and ties them into the starter bus.

NOTE

Only the left engine starting operation is described, since left and right engine starting systems are similar.

When the left engine master switch is on, 28 VDC power from the essential bus is connected to the left engine start switch. To initiate a start cycle the left generator switch is placed at on and the left engine start switch is depressed. When the start switch is depressed a coil in the start switch is energized which holds the switch in the depressed position. The left-hand start contactor energizes completing a circuit from the starter bus through two feeder protectors to the starter-generator. The feeder protectors (if energized) complete a ground circuit for the generator control unit (GCU) pin L. After a preset time delay, a ground circuit is provided for the current trip coil in the generator switch which in turn causes the applicable generator switch to move to OFF if a current differential is sensed between the starter-generator feeder lines. The No. 1 protector monitors the current in the power and ground leads and the No. 2 protector monitors current through the two legs of the split power lead. A differential current sensed by either protector will cause the generator switch to move to OFF disabling both the generator and the starting circuits. As the engine rpm increases to 42%, the engine tachometer generator output voltage is routed through a biasing network in the GCU to the gate of a silicon controlled rectifier (SCR). This biases the SCR into conduction which in turn provides a ground circuit for relay K1. Relay K1 is then energized and opens the circuit to the left engine start control relays No. 1 and No. 2 which break the starter contactor circuit and the start switch holding coil circuit de-energizing the starter. Relay K1 also provides a ground circuit for the GCU which in turn trips the generator switch to off. This prevents overloading the engine below idle speed when the starter begins to function as a generator. The gound circuit for start control relays No. 1 and No. 2 is made through normally closed contacts in the opposite engine start control relay No. 2. This prevents one engine start system from being energized when the other start system is in operation.

NOTE

For information regarding air start of these engines, refer to chapter 74.

A split-bus (soft-start) type auxiliary power unit (APU) is recommended for starting the engine with an external power source. The soft-start type APU supplies 28 VDC constant-voltage power to the essential bus while at the same time supplying 28 VDC, 800 amp constant current to the starter bus. A signal from the essential bus is supplied to the APU control circuitry which splits the power output when the start controller is energized.

The soft-start type APU is equipped with one oval plug and one rectangular plug. The oval plug is inserted into the external power utility receptacle (oval-shelled) at left FS 319 [1], or right FS 146 [2], which supplies power to the paralleling [2] or essential [1] bus. The rectangular plug is inserted in the external power start receptacle (rectangular-shelled) which supplies power to the starter bus.

[1] S/N 380-1 and Subs
[2] S/N 370-1 and -7

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DESCRIPTION AND OPERATION (Continued)

OPERATION S/N 380-1 AND SUBS CHANGED BY SOK 17

To operate the engine starting system, power must be available from the starter bus and essential bus. The starter bus supplies the power to the starter and the essential bus powers the starter control circuitry.

When external power is used, a switch in the external power start receptacle is actuated completing a circuit from the essential bus to the coil of the external power monitor relay. The energized external power monitor relay opens a circuit to the battery start relay, isolating the batteries from the starting circuit (refer to DC bus tie-in system).

The batteries are connected in series or parallel dependent on the position of the engine start switch (S1230) in the cockpit. With the engine start switch in the normal position, the batteries are connected in parallel for engine starting. With the switch at cold, the batteries are connected in series for engine starting.

NOTE

Only the left engine starting operation is described, since left and right engine starting systems are similar.

During a starting cycle with the APU operating, 28 VDC power is supplied to the essential bus by the APU to initiate a starting cycle (APU operating), the engine and electrical master, left hand generator and both battery switches must be positioned at ON. With the engine start switch (S1230) positioned at normal, depressing the left engine start switch energizes a coil within the start switch which holds the start switch in the depressed position (See Figure 80-00-3). At the same time, 28 VDC power from the essential bus is routed through the start switch, generator switch and through the generator control unit (K1 normally closed contacts) to the engine start control relays (K1057 and K521). Relay K521 contacts 3 and 5, completes a circuit to the start contactor (K1045). Relay K1057, contacts 11 and 13 completes a circuit through the external start power monitor relay (K515), through the engine start switch (K1230) to both battery monitor relays (K841 and K842). With both battery monitor relays energized, electrical power is routed to the coil of the parallel start relay (K502). When the parallel start relay is energized, the battery start relay is deenergized which in turn allows both batteries to remain connected in parallel. Power is then routed to the starter bus through the isolation relays and the energized closed contacts (A1 and A2) of relay K502. At the same time, 28 VDC is routed through the energized APU monitor relay (K4014) to the coil of the battery start APU disable relay (K23). With relay K23 energized, the batteries are isolated from the essential bus by de-energizing the battery relay K510. During this condition the batteries provide power to the starting circuit only and the APU powers all items connected to the essential bus. With the starter contactor (K1045) energized, a circuit from the starter bus through two feeder protectors to the starter-generator is completed. The feeder protectors (if energized) complete a ground circuit for the generator control unit (GCU) pin L. After a preset time delay, a ground circuit is provided for the current trip coil in the generator switch which in turn causes the applicable generator switch to move to OFF if a current differential is sensed between the starter-generator feeder lines. The No. 1 protector monitors the current in the power and ground leads and the No. 2 protector monitors current through the two legs of the split power lead. A differential current sensed by either protector will cause the generator switch to move to OFF disabling both the generator and the starting circuits. As the engine rpm increases to 42%, the engine tachometer generator output voltage is routed through a biasing network in the GCU to the gate of a silicon controlled rectifier (SCR). This biases the SCR into conduction which in turn provides a ground circuit for relay K1. Relay K1 is then energized and opens the circuit to the left engine start control relays No. 1 and No. 2 which break the starter contractor circuit and the start switch holding coil circuit de-energizing the starter. Relay K1 also provides a ground circuit for the GCU which in turn trips the generator switch to OFF. This prevents overloading the engine below idle speed when the starter begins to function as a generator. The ground circuit for start control relays No. 1 and No. 2 is made through normally closed contacts in the opposite engine start control relay No. 2. This prevents one engine start system from being energized when the other start system is in operation.

At low outside ambient temperatures (below 15°C), the engine start switch (S1230) should be placed at COLD during a start cycle. With the switch at cold, and subsequent to depressing the engine start switch (S546), the batteries are series connected through the battery start relay (K511) and are connected directly to the starter bus.

STARTER-GENERATOR

The starter-generator is a General Electric 400-amp, 30 VDC, combination starter and generator. The starter-generator starts as a shunt motor energized by the airplane batteries or an external 28 VDC power source. When the engine speed reaches about 42% rpm, the starter-generator begins to function as a generator. The starter-generator is equipped with a quick-attach-disconnect (QAD) mounting for installation on the engine accessory section.

ENGINE START SWITCH

The engine start switches are push-pull switches, each containing a holding coil to hold the switch in during the start cycle. Each switch controls power from the essential bus to the respective generator control unit and starting control circuitry.

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DESCRIPTION AND OPERATION (Continued)

START CONTROL RELAYS

There are two start control relays in each engine starting system. The No. 1 start control relay controls the starter contactor and the generator disable relay. The No. 2 start control relay controls the start switch holding circuit and the battery start relay and provides the interlock to prevent the opposite starter system from being operated simultaneously. The start control relays are located on shelf 125 in electrical equipment rack at FS 150.

STARTER CONTACTORS

There is one starter contactor in each engine start system. The starter contactor is energized from the essential bus through the No. 1 engine start control relay. When energized, the contactor completes the circuit from the starter bus to the starter-generator. The contactors are located on shelf 102 in the electrical equipment rack at FS 150.

GENERATOR CONTROL UNIT

Each engine starter-generator is controlled by a generator control unit. Both control units are in stalled in the electrical equipment rack at FS 150 on shelf 104. The generator control unit provides voltage regulation, overvoltage protection, equalization, speed sensing, and starting control. The starting control consists of a field weakening circuit, a start relay and a speed sensing circuit. The field weak ening circuit, by current and voltage control provides for high torque capabilities even at high starter rpm. The speed sensing circuit uses the output from an engine-driven tachometer generator to energize the start relay in the unit to de-energize the starter at 42% engine rpm.

EXTERNAL POWER START RELAY

The external power start relay controls power to the battery start relay. This relay is located on shelf 115 in the electrical equipment rack, at FS 150. The relay is energized from the essential bus through a switch which is actuated when the auxiliary power unit plug is inserted into the rectangular external power receptacle. (Refer to chapter 24).

BATTERY START RELAY

On S/N 380-1 thru -72 airplanes not changed by SOK 17, the battery start relay controls power from the batteries to the battery bus and starter bus. When de-energized, this relay connects both batteries in parallel to the battery bus. When energized, the battery start relay connects the batteries in series and connects this circuit to the starter bus to provide power to the starter-generator. The battery start relay is energized by power from the essential bus through the engine master, the engine start switch, the energized No. 2 engine start control relay, and the de-energized external start power monitor relay during an engine start using battery power. Also, during a battery start, power for the engine ignition system and the fuel boost pumps is supplied by battery No. 1 or the APU which supplies the 28 VDC essential bus during the starting cycle. The battery start relay is installed at FS 50. On S/N 380-1 and Subs changed by SOK 17, the battery start relay is energized during the starting cycle when the engine start switch (S1230) is positioned at cold. When energized, the batteries are series con nected to the starter bus. The battery start relay is de-energized during the starting cycle when the engine start switch (S1230) is positioned at normal. In this condition, the batteries are connected in parallel to the starter bus and with the APU op erating, the batteries are isolated from the essential bus.

PARALLEL START AND BATTERY MONITOR RELAYS — S/N 380-1 AND SUBS, CHANGED BY SOK 17

The parallel start relay (K502) is energized (through the battery monitor relays) by the essential bus when the engine start switch (S1230) is positioned at normal. When energized, a circuit is completed from the isolation relays to the starter bus and the circuit to the battery start relay (K511) is opened. The No. 1 and 2 battery monitor relays (K842 and K841) are energized by the essential bus through the engine start switch (S1230) when positioned at normal. These relays when energized, complete the power circuit to the parallel start relay (K502). The battery monitor relays are mounted on the electrical equipment rack at FS 150. The parallel start relay is mounted at FS 50 RH.

ENGINE START SWITCH — S/N 380-1 AND SUBS CHANGED BY SOK 17

The engine start switch (S1230) is a two position switch mounted adjacent to the anti-skid switch in the cockpit overhead panel. With the switch at normal, the batteries are connected in parallel during battery starts. When in cold (15°C or lower) position the batteries are connected in series during a battery start.

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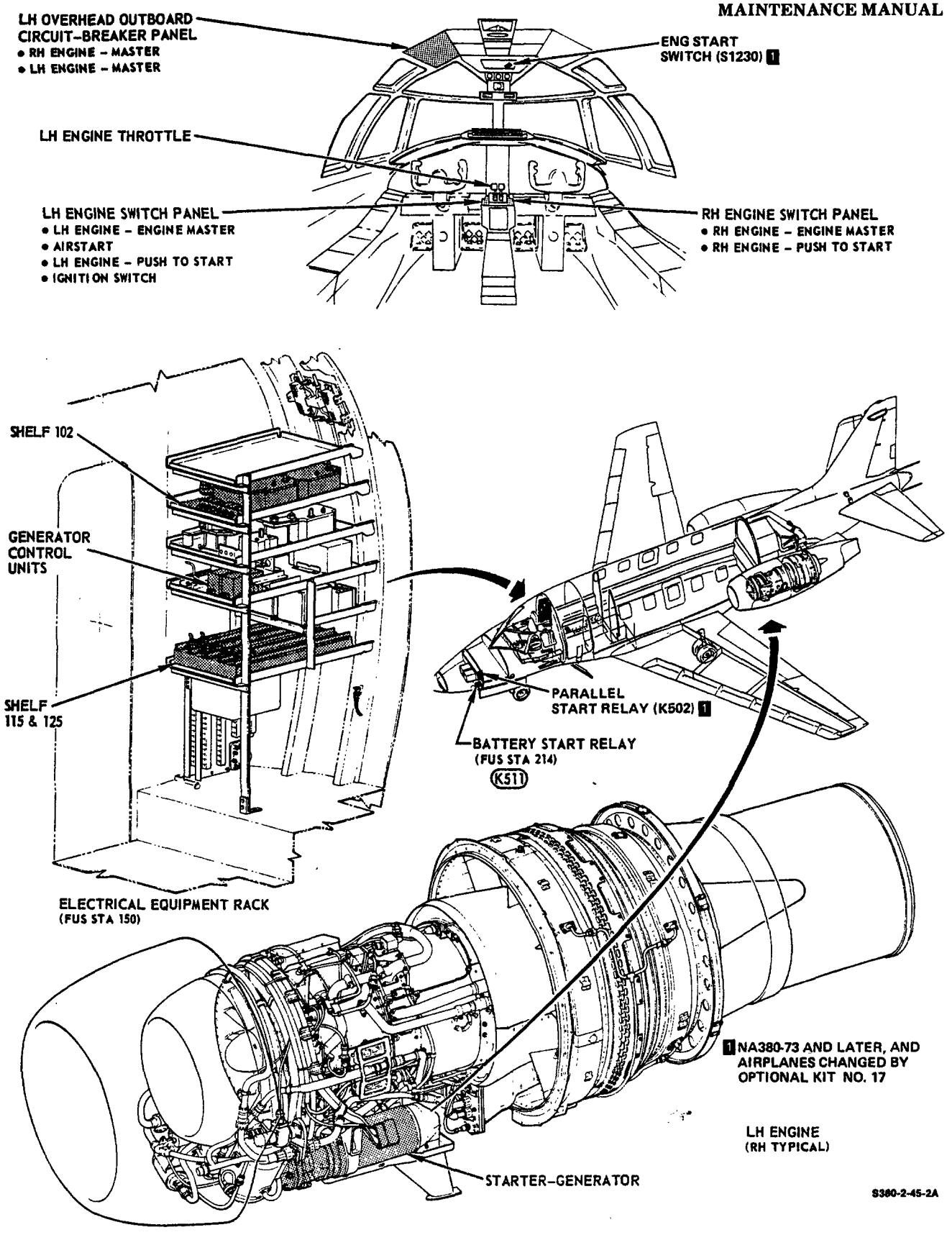
LH OVERHEAD OUTBOARD CIRCUIT-BREAKER PANEL
• RH ENGINE – MASTER
• LH ENGINE – MASTER
ENG START SWITCH (S1230) 1
LH ENGINE THROTTLE
LH ENGINE SWITCH PANEL
• LH ENGINE – ENGINE MASTER
• AIRSTART
• LH ENGINE – PUSH TO START
• IGNITION SWITCH
RH ENGINE SWITCH PANEL
• RH ENGINE – ENGINE MASTER
• RH ENGINE – PUSH TO START
SHELF 102
GENERATOR CONTROL UNITS
SHELF 115 & 125
ELECTRICAL EQUIPMENT RACK (FUS STA 150)
PARALLEL START RELAY (K502) 1
BATTERY START RELAY (FUS STA 214) (K511)
1 NA380-73 AND LATER, AND AIRPLANES CHANGED BY OPTIONAL KIT NO. 17
LH ENGINE (RH TYPICAL)
STARTER-GENERATOR
S380-2-45-2A

Figure 80-00-1 — Engine Starting System Component Location

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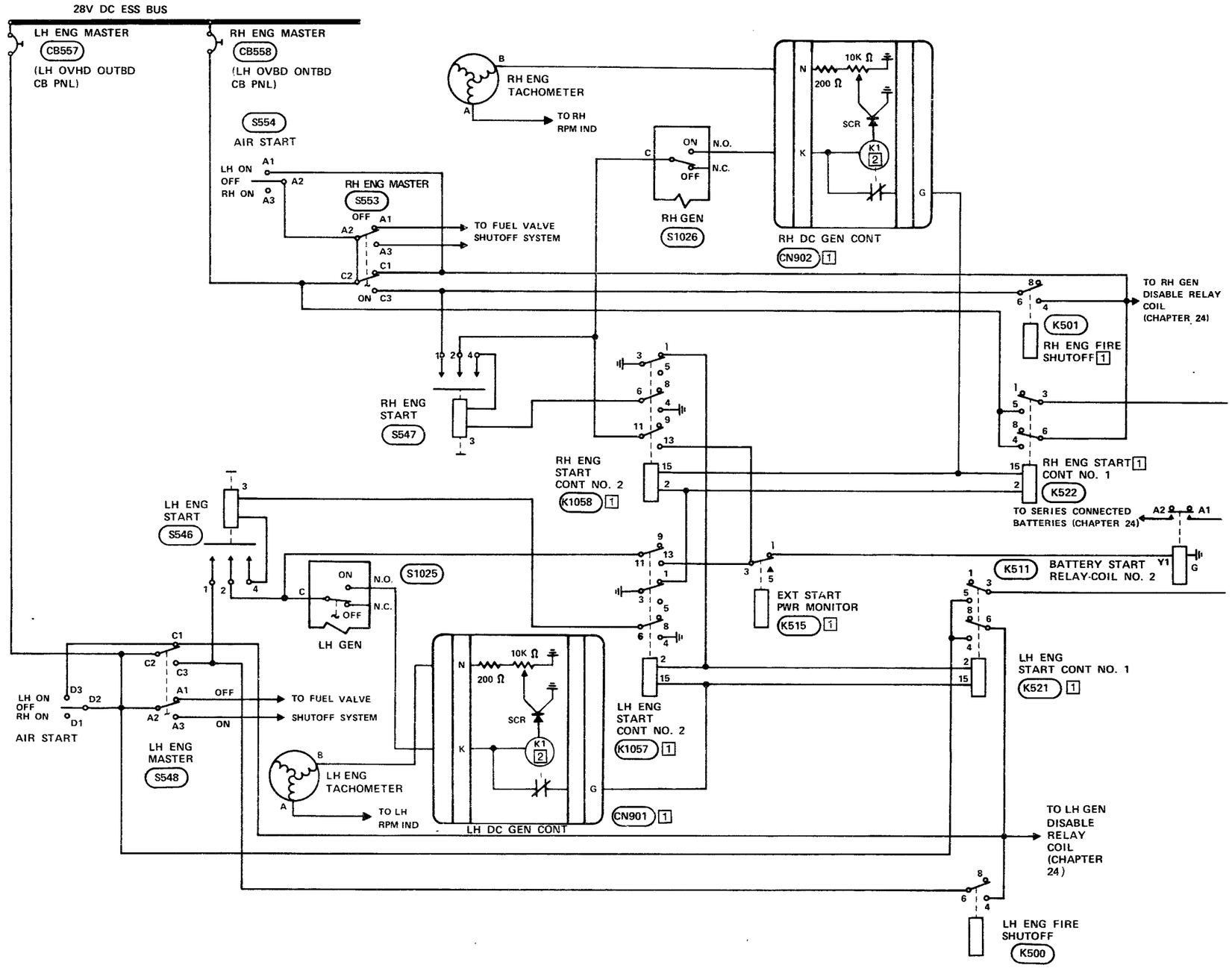

1 ELECTRICAL EQUIPMENT RACK-FUSELAGE STATION 150 RH.

2 RELAY K1 IS DE-ENERGIZED DURING THE STARTING CYCLE, AT APPROXIMATELY 42% RPM, RELAY K1 ENERGIZES, WHICH DE-ENERGIZES THE ENGINE START CONTROL RELAYS. K1 IS A FUNCTION OF THE SPEED SENSING CIRCUIT WITHIN THE GCU.

Figure 80-00-2 — Engine Starting System Electrical Schematic S/N 380-1 thru -72 Not Changed by SOK 17 (Sheet 1 of 2)

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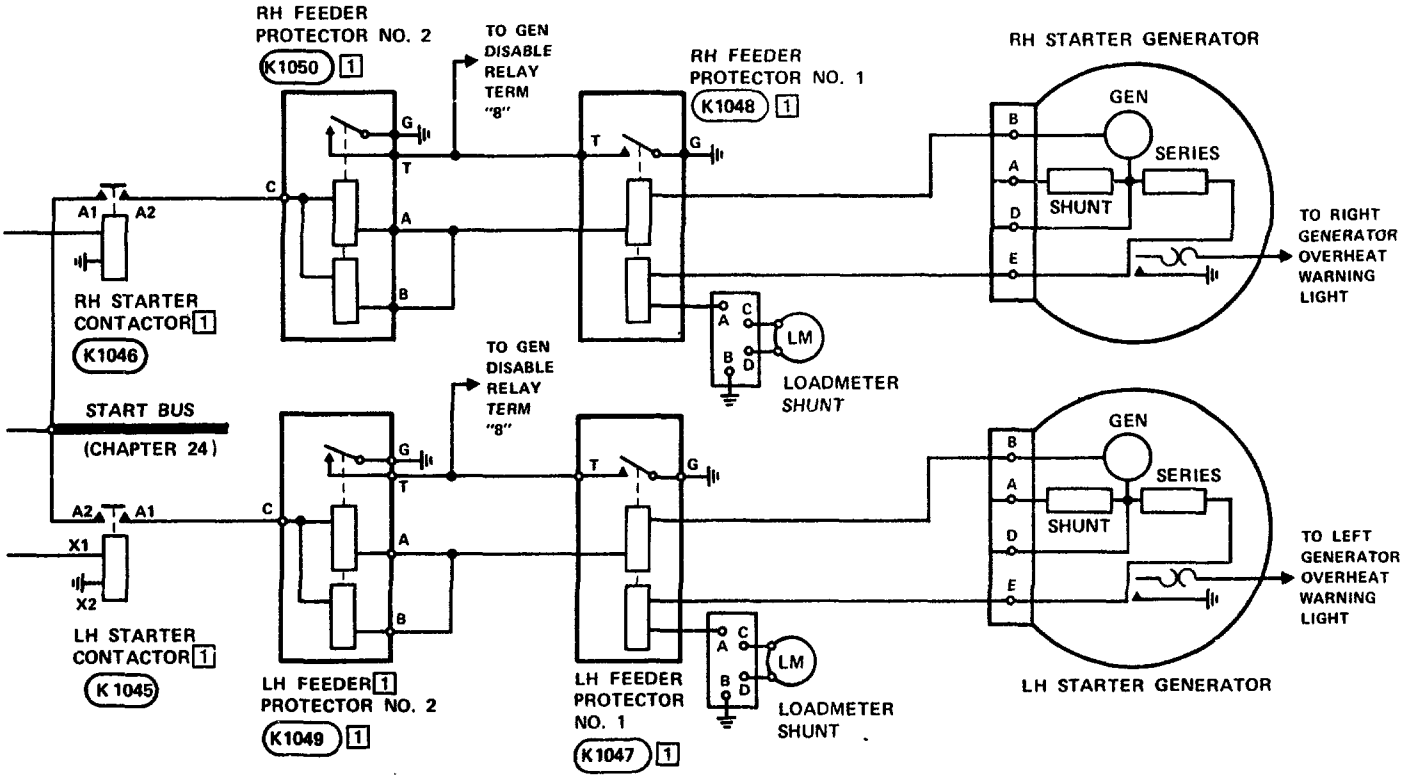

NOTE: STARTER LIMITS:

30 SEC ON, 2 MIN OFF
30 SEC ON, 5 MIN OFF
30 SEC ON, 20 MIN OFF
30 SEC ON, 90 MIN OFF

1 ELECTRICAL EQUIPMENT RACK-FUSELAGE STATION 150 RH.

2 RELAY K1 IS DE-ENERGIZED DURING THE STARTING CYCLE, AT APPROXIMATELY 42% RPM, RELAY K1 ENERGIZES, WHICH DE-ENERGIZES THE ENGINE START CONTROL RELAYS. K1 IS A FUNCTION OF THE SPEED SENSING CIRCUIT WITHIN THE GCU.

S380-2-45-1A

Figure 80-00-2 — Engine Starting System Electrical Schematic S/N 380-1 thru -72 Not Changed by SOK 17 (Sheet 1 of 2)

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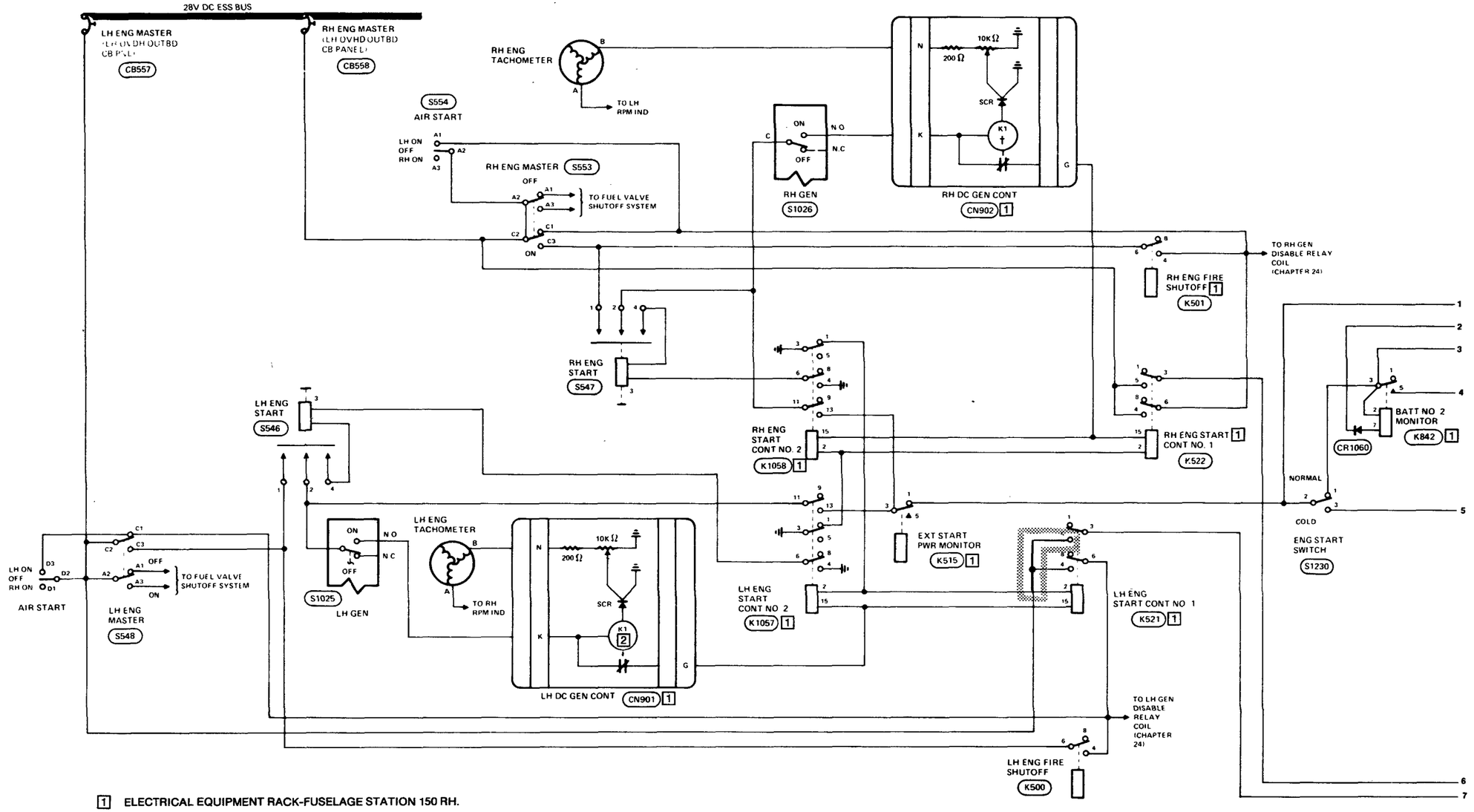

1 ELECTRICAL EQUIPMENT RACK-FUSELAGE STATION 150 RH.

2 RELAY K1 IS DE-ENERGIZED DURING THE STARTING CYCLE, AT APPROXIMATELY 42% RPM, RELAY K1 ENERGIZES, WHICH DE-ENERGIZES THE ENGINE START CONTROL RELAYS. K1 IS A FUNCTION OF THE SPEED SENSING CIRCUIT WITHIN THE GCU.

3 FUSELAGE STATION 50 RH

Figure 80-00-3 — Engine Starting System Electrical Schematic S/N 380-1 and Subs Changed by SOK 17 (Sheet 1 of 2)

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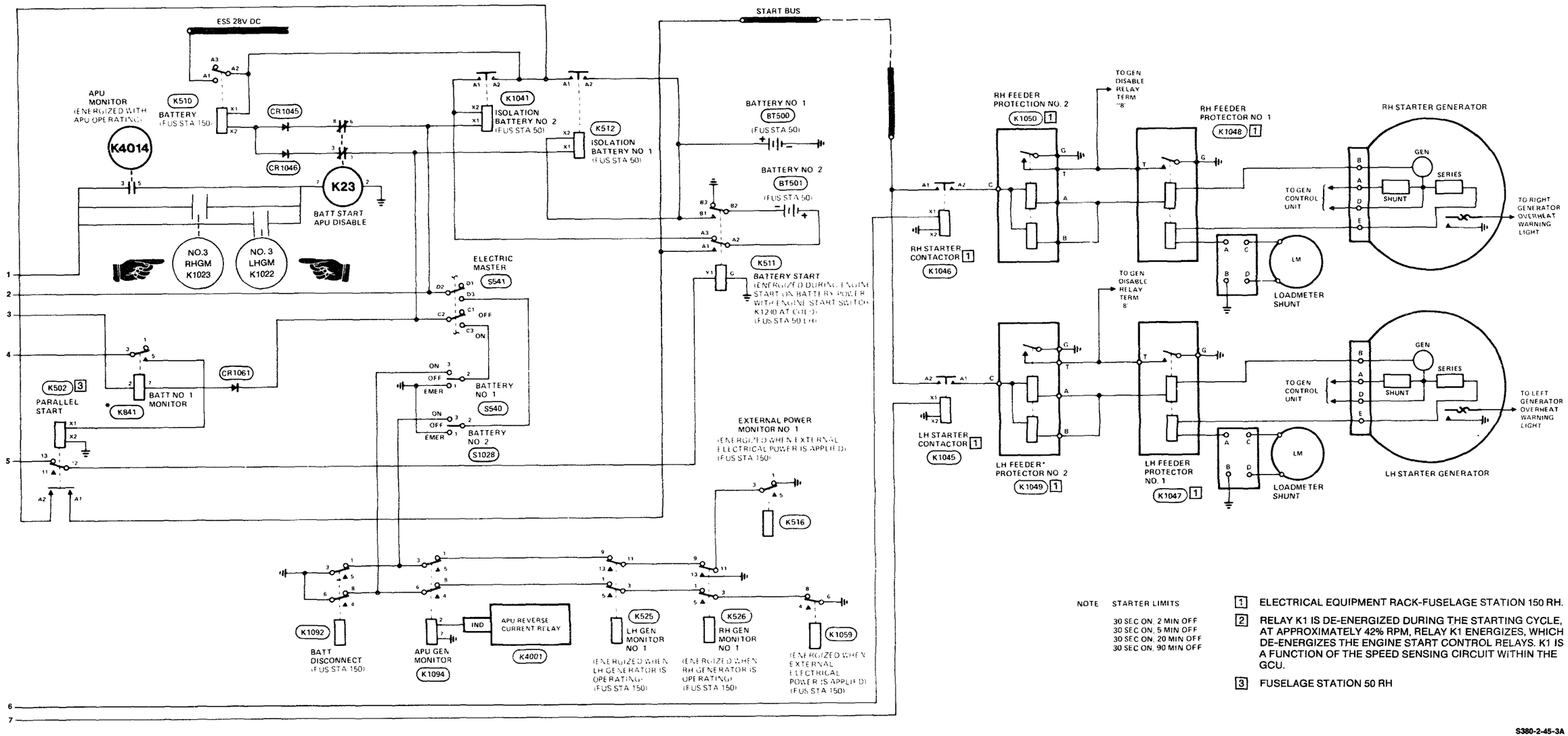
ESS 28V DC
START BUS
APU MONITOR (ENERGIZED WITH APU OPERATING)
K4014
K510 BATTERY (FUS STA 150)
CR1045
CR1046
K23 BATT START APU DISABLE
NO.3 RHGM K1023
NO. 3 LHGM K1022
K1041 ISOLATION BATTERY NO 2 (FUS STA 50)
K512 ISOLATION BATTERY NO 1 (FUS STA 50)
BATTERY NO 1 BT500 (FUS STA 50)
BATTERY NO 2 BT501 (FUS STA 50)
K511 BATTERY START (ENERGIZED DURING ENGINE START ON BATTERY POWER WITH ENGINE START SWITCH K128 AT COLD) (FUS STA 50 LH)
ELECTRIC MASTER S541
OFF
ON
BATTERY NO 1 S540
BATTERY NO 2 S1028
CR1061
K502 3 PARALLEL START
K841 BATT NO 1 MONITOR
EXTERNAL POWER MONITOR NO 1 (ENERGIZED WHEN EXTERNAL ELECTRICAL POWER IS APPLIED) (FUS STA 150)
K516
K1092 BATT DISCONNECT (FUS STA 150)
K1094 APU GEN MONITOR
IND
APU REVERSE CURRENT RELAY
K4001
K525 LH GEN MONITOR NO 1 (ENERGIZED WHEN LH GENERATOR IS OPERATING) (FUS STA 150)
K526 RH GEN MONITOR NO 1 (ENERGIZED WHEN RH GENERATOR IS OPERATING) (FUS STA 150)
K1059 (ENERGIZED WHEN EXTERNAL ELECTRICAL POWER IS APPLIED) (FUS STA 150)
RH FEEDER PROTECTION NO. 2 K1050 1
TO GEN DISABLE RELAY TERM "8"
RH FEEDER PROTECTOR NO 1 K1048 1
RH STARTER GENERATOR
RH STARTER CONTACTOR 1 K1046
TO GEN CONTROL UNIT
GEN
SERIES
SHUNT
LM
LOADMETER SHUNT
TO RIGHT GENERATOR OVERHEAT WARNING LIGHT
LH STARTER CONTACTOR 1 K1045
LH FEEDER PROTECTOR NO 2 K1049 1
LH FEEDER PROTECTOR NO. 1 K1047 1
LH STARTER GENERATOR
TO LEFT GENERATOR OVERHEAT WARNING LIGHT

NOTE STARTER LIMITS

30 SEC ON, 2 MIN OFF
30 SEC ON, 5 MIN OFF
30 SEC ON, 20 MIN OFF
30 SEC ON, 90 MIN OFF

[1] ELECTRICAL EQUIPMENT RACK-FUSELAGE STATION 150 RH.

[2] RELAY K1 IS DE-ENERGIZED DURING THE STARTING CYCLE, AT APPROXIMATELY 42% RPM, RELAY K1 ENERGIZES, WHICH DE-ENERGIZES THE ENGINE START CONTROL RELAYS. K1 IS A FUNCTION OF THE SPEED SENSING CIRCUIT WITHIN THE GCU.

[3] FUSELAGE STATION 50 RH

S380-2-45-3A

Figure 80-00-3 — Engine Starting System Electrical Schematic S/N 380-1 and Subs Changed by SOK 17 (Sheet 2 of 2)

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ENGINE STARTING SYSTEM - TROUBLE SHOOTING

1. Trouble Shooting Engine Starting System

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
A. ENGINE WILL NOT COME UP TO STARTING RPM - EXTERNAL POWER.		
(1) Power output of external power unit insufficient.	Check output of unit: minimum, 24 volts dc at 600 amps; maximum 28.5 volts dc at 1000 amps.	Adjust external power unit output or replace unit.
(2) Starter-generator faulty.	Check for 28 volts dc at terminal "B" of starter-generator with start system energized.	If voltage present is 24 to 28 volts, check starter per manufacturer's instructions.
(3) Generator control unit speed sensing circuit faulty.	If engine begins to rotate but start system de-energizes before starting rpm is reached, replace control unit with one known to be good and re-check start system.	Replace generator control unit.
(4) Starter contactor faulty.	Energize start system and check for 24 - 28 volts dc at pins "A1" and "A2" of starter contactor.	Replace contactor if "A1" is 24 - 28 volts and "A2" is much less.
B. ENGINE WILL NOT COME UP TO STARTING RPM - BATTERY POWER.		
(1) Batteries not fully charged.	Check charge on batteries. (Refer to Chapter 12.)	Recharge batteries.
(2) Starter contactor faulty.	Energize start system and check for 24 or 48 volts at pins "A1" and "A2" of starter contactor.	Replace contactor if either is 24 or 48 volts and the other is much less.
(3) Battery start relay faulty.	Energize start system and check for 24 or 48 volts dc at pins "A1" and "A2" of relay. If voltage is present at "A1" go to probable cause (2). If voltage is present at "A1" but not at "A2", relay is faulty.	Replace relay.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
(4) Starter-generator faulty.	Check for 24 or 48 volts dc at terminal "B" of starter-generator with start system energized.	If voltage is present, check starter-generator in accordance with manufacturer's instructions.
(5) Generator control unit speed sensing circuit faulty.	If engine begins to rotate but start system de-energizes before starting rpm is reached, replace control unit with one known to be good and re-check start system.	Replace generator control unit.
C. STARTER WILL NOT CRANK ENGINE.		
(1) Starter power leads loose or not connected to starter.	Remove power and visually check power lead connections.	Connect and/or tighten leads as required.
(2) Essential bus not energized.	Refer to Chapter 24.	Energize essential bus.
(3) Start switch faulty.	Depress start switch and check for 28 volts dc at terminals "1" and "2". If power is available at terminal "1" but not at terminal "2", switch is defective.	Replace switch.
(4) No. 1 start control relay faulty.	Perform relay circuit check. (Refer to Wiring Diagram Manual.)	Replace relay.
(5) No. 2 start control relay in opposite engine start system faulty.	With power off, check for continuity between contacts "1" and "3". If no continuity exists, relay is faulty.	Replace relay.
(6) Starter contactor faulty.	Energize start system and check for 24 or 48 volts dc at pins "A1" and "A2" of starter contactor.	Replace contactor if voltage is present at one contact but not at the other.

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
D. ENGINE START SWITCH WILL NOT STAY DEPRESSED DURING START CYCLE.		
(1) Start switch holding coil faulty.	Check for continuity between terminals "3" and "4" of switch.	Replace switch assembly.
(2) No. 2 start control relay faulty.	Perform relay circuit check. (Refer to Wiring Diagram Manual.)	Replace relay.
(3) Generator control unit speed sensing circuit faulty.	Replace control unit with one known to be operating properly and recheck start system.	Replace defective generator control unit.
E. STARTER WILL NOT CRANK ENGINE WHEN ENGINE START SWITCH AT NORMAL-NA380-73 AND LATER, AND AIRPLANES CHANGED BY O.K. #17		
(1) Parallel start relay (K502) faulty.	Check for 28 volts dc at terminals XI with start button depressed.	If voltage is present, replace relay.
(2) Battery monitor relay(s) (K841 & K842) faulty.	Check for 28 volts dc at terminal 2 of both relays with start button depressed.	If voltage is present and the ground circuit is known to be complete, replace applicable relay.
F. BATTERIES NOT ISOLATED FROM ESSENTIAL BUS DURING START CYCLE (APU OPERATING) NA380-73 AND LATER AND AIRPLANES CHANGED BY O. K. #17		
(1) Faulty battery start APU disable relay (K23).	Check for 28 volts dc at terminal 7 of relay when start button is depressed.	If voltage is present, replace relay.
(2) APU monitor relay faulty.	With APU operating, check for 28 volts dc at terminal 7 of APU monitor relay.	If voltage is present, replace relay.

ENGINES STARTING SYSTEM - MAINTENANCE PRACTICES

NOTE: Removal and installation of the starter-generator and generator control units is covered in Chapter 24.

1. Replacement of Engine Starting System Relays

The engine starting system relays and contactors are installed on trays in the electrical equipment rack at the bulkhead behind the co-pilot at fuselage station 150 or at fuselage station 50 adjacent to the batteries.